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FOREWORD

These Proceedings include the reports delivered during the First International Symposium "Experimental Facilities and Aircraft Certification". The Symposium was organized by Russian governmental bodies: State Committee on Defense Industry Branches, Aviation Register of Interstate Aviation Committee, Air Transport Department, and leading Russian research centers: the Central Aero-Hydrodynamic Institute (TsAGI), the Flight Research Institute (LII), the Central Institute of Aviation Motors (CIAM), the All-Russian Institute of Aviation Materials (VIAM), the Research Institute of Aviation Equipment (NIIAO).

The Symposium took place in summer 22-25 August, 1995 in Zhukovsky, the picturesque proximity of Moscow, where TsAGI, LII, NIIAO were situated. As it got into practice the Symposium was held within the scope of Moscow International Aero-Space Salon-95. Over three hundred Russian participants and forty participants from the USA, Canada, the UK, France, Germany, India, Ukraine attended the event. 88 reports and over 80 posters were presented during the Symposium. Topics of interest included experimental facilities and certification testing for aircraft and helicopters.

The Symposium was aimed at stimulating international activities of specialists and encouraging free exchange of information and views on today's standing of experimental facilities and aircraft certification. In addition to the scientific sittings the participants had a chance to get acquainted with experimental facilities of the Russian scientific centers, to observe the exhibition at the Salon and to enjoy a spectacular aeroshow given by famous pilots.

The working languages of the Symposium were Russian and English with providing simultaneous translation at the plenary sessions. The Symposium Proceedings are published in English to give specialists from different countries an opportunity to get better acquainted with Russian research and development activities in certification of facilities and aircraft.

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PLENARY SESSION

PROSPECTS OF CIVIL AVIATION DEVELOPMENT AND AIRCRAFT CERTIFICATION IN RUSSIA

A.G.Bratukhin
State Committee on Defense Industries Branches, Moscow, Russia

Aviation Industry in Russia is a big machine-building branch with a high scientific and technological potential affecting progressively many allied industries-electronics, communications means, radio and metallurgy industries, etc.

Aviation industry enterprises embracing over 80% of the former USSR's Minaviaprom potential are located actually in every economic region of Russia and provide over 3 million jobs, with the related industries being taken into account.

It is significant that Russia together with other CIS countries form one of the three world aviation centers (besides the USA and United Europe) capable of performing the entire cycle of modern airliners development and manufacture, these being super complex technical systems based on advanced technologies.

To secure the national interests of Russia in civil aviation area responsible to a great extent for a dynamic social-economic development of the country the State Committee for Defense Industries has elaborated a national scientific and engineering program called "Program for civil aircraft development in Russia up to 2000". The Program has won the Russian Federation Government's Approval and has been granted the Federal Program Status.

An unprecedented conversion process, involving the aviation industry reorienting towards development and manufacture of new civil aircraft and helicopters with high performance, as per the goals set forth in the Federal Program, will allow the Russian airlines' fleets to be re-equipped as regards every aircraft type before 2000—2005. Alongside the re-equipping of airliners with most advanced, both from the technical and commercial points of view, aviation equipment, and the solution of most acute problems of the Russian transport infrastructure, the Program would prevent a loss of the aviation industry's unique scientific and engineering potential, known for its world-level achievements, advanced technological equipment and highly skilled people.

The Program in question is an integrated one: it covers all the stages of developing Russian aircraft and helicopters which are not inferior, from the technical point of view, to the best aircraft types of the world. The Program includes coordinated questions of lead research and development to be carried out in areas of aviation science, engineering, technology, materials, certification of aircraft types ensuring their reliability and safety, aircraft production certification and setting up and mastering of full-scale production.

The major goals of the Program are:

1. Development by 2000 of new generation civil aircraft conforming to the forecast technical and economic levels of perfection, which include:
   — fuel efficiency of all airplane types to be brought up to a level of 15—18 g/pass-km; of commuter airliners — 15—25 g/pass-km depending upon the passenger capacity, and 0.5—0.7 kg/t-km of helicopters;
   — full compliance with the international requirements of flight safety, airplane ecological influence upon the environment and passenger cabins comfort;
   — technical provisions for the flying hours number to be increased to 3500—5000 for big airliners of various types: to 2400—3000 for commuter airplanes and to 1800—2400 for medium and large helicopters due to their structure serviceability;
2. Retaining of the scientific and production potential of the national aviation industry as a major factor of its competitiveness in the world market;
3. Expansion of the range of full-scale production aircraft types to be employed for various purposes (in agriculture, forest-fire fighting, ice and geological reconnaissance, fishery reconnaissance, map-making, patrolling and nature protection);
4. A considerable enhancement of ergonomic characteristic and serviceability of civil aviation products by implementing new technologies in avionics, digital electronic equipment and advanced structural materials, non-metals included;
5. Forming a scientific and engineering reserve for aerodynamics, gas dynamics and aircraft strength investigations; for achieving great (supersonic as well) flight speeds; hypersonic technologies (material, power plants and structures); study of prospects of developing aerospace vehicles and airliners of super passenger capacity. The Program forces production of up-to-date comfortable airplanes with low fuel consumption, such as the long-haul IL-96-300 aircraft and IL-114 commuter airplane designed by the S.V.Ilyushin bureau; medium Tu-204 and short-haul Tu-334 of the Tupolev bureau; commuter, business, agricultural, ambulance, tourist and taxi airplanes of general designers R.A.Belyakov, M.P.Simonov, G.V.Novozhilov, A.N.Dondukov, G.E.Losino-Losinsky and V.K.Novikov.

Within the framework of conversion the Beriev ANTK (general designer G.S.Panatov) is developing a multipurpose amphibian aircraft in search-and-rescue, cargo-passenger, anti-fire and ecology monitoring versions.

Helicopter design bureaus (general design designers S.V.Mikheev and M.V.Vainberg) are working to create and start full-scale production, in collaboration with the related plants, of highly efficient Ka-62 and Mi-38 helicopters to be made of polymer composite materials and have high fuel efficiency and flight performance.

In the sphere of international cooperation the A.N.Tupolev ANTK and S.V.Ilyushin design bureau are implementing development programs for a second-generation supersonic airliner and wide-body airplanes to carry 500—600 passengers. Despite the difficulties encountered the work carried out under the federal "Program for civil aviation development in Russia up to 2000" permitted to have almost everything planned for the 1992—1994 period completed. The principal results attained:

— the IL-96-300 long-distance wide-body started passenger services in 1993;

— an airworthiness certificate has been obtained for the Tu-204 medium haul airplane; passenger carriage is scheduled for 1995;

— the An-124 cargo airplane has been certificated;

— it is for the first time in our history of aviation that certificates have been received by the Ka-32A, Mi-34 and Mi-26 helicopters; the Ka-32A-1 helicopter has also been certified and entered service;

— certification tests are under way for the IL-98M, Tu-204-120, IL-114, IL-103 and An-38 aircraft, Tu-204 cargo version, Ka-26 and Mi-172 helicopters, etc.;

— integrated research and development work has been performed on new air-vehicles: Tu-334, Be-200, Tu-330, IL-76MF airplanes and Mi-38, Ka-62 helicopters, etc.

There has been launched work aimed at studying and creating the required scientific and engineering reserve in science, engineering technology, materials, integrated quality support systems for aviation industry products, their production certification, etc.

Among these developments there should be emphasized the following ones:

— new wing configurations by TsAGI designed for greater cruise speeds (M=0.78 for short-hauls airplanes and M=0.85 for long-distance ones) and this will allow their cost effectiveness to be raised considerably;

— research carried out by TsAGI to work out recommendations for lowering the structure's weight, with the airplane long service life being provided at the same time (60000—90000

flights) suggestions in collaboration with CIAM on a wide use of modern composite materials and aluminum-lithium alloys being made as well as proposals on active systems usage to decrease loads and ideas to increase the flutter critical speeds;
- CIAM development of scientific base for creating an engine with a super high by-pass ratio, that has been implemented in the NK-93 engine development;
- experimental bench, and flight testing by CIAM doing research into the ramjet engine intended for future air vehicles of super high speeds;
- to master satellite technologies and participate in introducing the global communications navigation and observation system based on these technologies, the LII has conducted flight tests to evaluate its accuracy and measures to increase it, and for the first time in Europe;
- there has been performed flight research to finalize automatic landing with satellite systems using a correcting station;
- LII's investigation to substantiate the use of standard gaseous fuels in aviation.

Of top priority for the aviation research institute are subjects of developing advanced technologies, specialized technological equipment and highly effective materials such as low-density aluminum alloys, polymer composites and high-strength corrosion resistant alloys, etc., which determine, in many respects, the scientific and engineering level of aircraft.

An important component of civil aircraft flight safety provision is their certification, that is establishment of their compliance with the airworthiness requirements, which determines a transportation system safety level depending upon an aircraft regarded as a link in this system.

The first airworthiness requirements of civil aircraft in our country have come into being in 1967. Since that time they have been evolving. In 1974 there appeared NLGS-2 meeting the world standards and they played a significant role in developing and certifying the IL-96, Yak-42 and An-28 airplanes, as well as the Tu-144 supersonic transport. Subsequent activities to improve the Requirements ended in preparing more advanced NLGS-3 made use of in developing and certification the new-generation IL-96-300, Tu-204 and IL-114 aircraft. Similar trends have been observed with developments. For the last period of time Russia has done much work to harmonize the national and western requirements. Airworthiness requirements have been developed for the large and light AP-25 and AP-23 aircraft, the AP-29 helicopter, AP-33 engine, etc., and these requirements are analogous of western ones by their structure and similar to them by their contents, though having certain differences due to our specific flight practices. This is a big step forward since Russian civil aircraft certification by the new requirements makes them automatically comply with the western regulations.

The level of civil aviation equipment airworthiness is greatly influenced by its certification system. Established in 1973 Aviaregister was not an independent competent body as it functioned under the Civil Aviation Ministry. An Independent Aviaregister came to life in 1986 when it was made directly subordinate to the Council of Ministers. After the USSR disintegration the Aviaregister has become a part of the Interstate Aviation Committee of CIS and on the instructions of the Russian government it has powers of a federal body and this provides its independence.

The certification system of Russia is based on the full amount of certification evaluations to be conducted at the general designer's stage with participation as Aviaregister experts of certification centers at the Minaviaprom and civil aviation research institutes, then come check evaluations at the Aviaregister stage. Hence, the aircraft industry makes a great contribution to aviation products certification as it carries the main burden of performing all requirements compliance evaluations.

Of significance for civil aircraft certification is a question of their production certification. Aviaregister in collaboration with Aviation Industry NIAT has elaborated a system and normative documentation to the effect, and as a result, the Voronezh and Ulyanovsk mass production plants have been given certificates for their productions by now.

Of no less significance and new, at the same time, to us is a problem of design bureaus certification in order to allow them to create new types of air vehicles. While we do not have any doubts concerning large design bureaus engaged in traditional civil aircraft development, but as to a great number of design bureaus (about 50) that have been set up recently and which deal with business aircraft it is doubtful that they are capable of designing, manufacturing and certifying a new air vehicle and providing support for its operation. So Aviaregister together with the aircraft industry have worked out requirements to such design bureaus, have set up a committee of aircraft industry's and research institutes' representatives, which is to consider the work of such design bureaus and issues conclusions as to the possibility of their certification, several dozens of such design bureaus have already been certificated.

Serious questions question are posed by environmental protection against aircraft engine noise and emissions. The most grave problem is that of engine noise.

At present all the civil aircraft and the civil aircraft and helicopters now in operation in Russia (56 all in all, versions included) have been certified for compliance with the noise requirements of ICAO, Annex 16.

Now certification tests are well under way of the Tu-204 and IL-96 versions powered by foreign engines.

Thus, practically speaking, the whole aircraft fleet formally meets the international community noise requirements.

Unfortunately, the aircraft forming the base of the CIS countries' airlines (Tu-134, Tu-154, IL-96TD and IL-86 are certified by ICAO, Annex 16, Chapter 2, dealing with aircraft of old construction. ICAO has recommended staged restrictions be introduced to these aircraft flights starting in 1995 and a complete ban on them in 2002.

Now already the local authorities of many airport regions enjoying the support of people have banned such aircraft flights.

Besides, the ICAO environment protection against aircraft, has elaborated proposals on making community noise requirements to subsonic jet airliners more strict (Chapter 3).

According to one of the variants they are supposed to be applied to new aircraft starting in 1998. Drastic measures will have to be taken to satisfy these more strict requirements.

A good example of how this task is solved is IL-76M development — an IL-96 version with modern engines installed.

Aviation industry in Russia has vast experiment facilities in research institutes and design bureaus for certification work to be done at.

For example, while developing the Tu-204 there were built 70 test benches. 20 of them being employed for carrying out certification evaluation.

Experimental facilities for strength investigation play a very important role in certification process. The aerodynamic complex of TsAGI consisting of over 50 sets permits research to be done into aircraft aerodynamic characteristics and those of aeroelasticity using models and small life-size objects for a whole possible range of flight speeds, from small subsonic ones to hypersonic, space flight speeds included. Nearly 50 industry's laboratories and centers, engaged in strength problems studies and to be found at a number of institutes, design bureaus and at mass production plants, are conducting vast investigations of static and endurance characteristics of constructions, beginning with simple objects tests to determine the materials' properties and finishing with certification tests of life-size structures (including the vehicle as a whole). In addition all the loads at various flight modes are reproduced, including the limiting ones, as well as external factors (temperature and humidity) affecting the object's strength characteristics.

The importance of engine development facilities should not be underestimated. NIT's CIAM is the only complex is a key component of a system of developing new engines. It is next to impossible to create a competitive engine without conducting final development work on the altitude and special stands of this complex. All the engines designed in our country have been put to experimental tests at the stands of NITs CIAM.

At present NITs CIAM disposes of power generating equipment with total electric power of 500 MW, and has over 50 stands of various applications.

NITs CIAM is compulsory participant in state tests of new aviation products and certification tests.

The LII's facilities are of great significance for the aviation industry.

The IL is a unique aviation center provided with all the necessary means to conduct flight certification tests, these means including flight navigation systems, micro-wave landing aids, equipment for satellite technologies to be developed, outer trajectory measurement systems and those of flight experiment management.

The LII possesses several dozens of flying testbeds allowing lead development of new aircraft systems to be fulfilled, as well as a portion of certification evaluations. The Institute also has a big experimental complex to conduct certification evaluation of seats dynamic characteristics, lightning strike resistance, influence of external electromagnetic fields, etc.

In conclusion it must be pointed out, that under the difficult conditions found in our country, the "Program", which gained support of the government as a fundamental long-term project, has become a real instrument of the state's technical policy pursued in such a high-technology and competitive area as aviation industry.

CERTIFICATION OF THE CIVIL AERONAUTICAL ENGINEERING. EXPERIENCE AND RESULTS OF CERTIFICATION CONDUCTED IAC AVIATION REGISTER

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1. LEGAL PRINCIPLES OF CERTIFICATION.

After the disintegration of the USSR, 12 independent states, former republics of the USSR concluded in December of 1991 the Agreement on civil aviation and on use of air space (the Minsk Agreement). These states were as follows:

Republic of Azerbaijan
Republic of Armenia
Republic of Byelorussia
Republic of Georgia
Republic of Kazakhstan
Republic of Kirgizia
Republic of Moldova
Republic of Tadzhikistan
Republic of Uzbekistan
Russian Federation (Russian Soviet Federative Socialist Republic at that time)
Turkmenistan
Ukraine

Later on, Lithuania and Estonia joined the Minsk Agreement as observers.

The Minsk Agreement specifies that the spheres of joint conducting and regulation as regards the Contracting States involve:

- elaboration of interstate normative acts and standards on flight safety including airworthiness regulations of aircraft and rules of their certification;
- aircraft certification.

The above agreement stipulates that for the Agreement's purposes to be realized the Contracting States shall establish:

— Council for Aviation and Use of Air Space consisting of representatives of the Contracting States;
— Interstate Aviation Committee (IAC).

The Contracting states have admitted the IAC as a successor of Gosavianadzor of the USSR as concerns the functions agreed upon by these states. The states have confirmed their support of the functions of the IAC in respective decrees of the President and/or acts of the Government.

In the Russian Federation, this is formalized, in particular, by the Decree of the President of May 5,1992 N 439 and by the Act of the Government of April 23, 1994 N 367. Paragraph 1 of this act runs as follows:

"To delegate to the IAC the power and responsibility of a Federal executive authority in the field of the formation of aircraft airworthiness regulations and to entrust the IAC with the elaboration and improvement of Aviation rules".

According to the paragraph 2 of the Act, the IAC is entrusted with certification of aircraft and their components, production certification of aeronautical engineering with a subsequent issue of respective certificates and equivalent documents. These powers of the IAC are also confirmed at the international level on behalf of the Russian Federation in the Memorandum of Understanding between the Government of the Russian Federation and the Government of the USA in the field of technical cooperation aimed at concluding a bilateral agreement on airworthiness. The Memorandum is signed on June 30, 1995 by V.S. Chernomyrdin, Chairman of the Government of the Russian Federation and A. Gore, US Vice-President.

It should also be noted that the law "On certification of products and services" was adopted on June 10, 1993 in the Russian Federation. The system of certification developed by the IAC is in full conformity to this law. The system is registered by the authorized Russian body Russian Federation Committee of Standardization, Metrology and Certification (Certificate of registration N POCCRU 0001/01AT 00 of July 21, 1994).

The IAC deals with aircraft certification in the interests and on behalf of all Contracting states; each state is to admit the validity of certificates issued by the committee. Aviaregister, one of the commissions of the IAC, is responsible for functions of certification.

The system of Aviation Rules (ARs, in Russian -) is an important part of the legal field of certification. One of the functions of Aviation register implies the organization of elaboration of ARs.

At the present time, a package of ARs has been already worked out which forms the core of the system of ARs required to certify aeronautical engineering, its production and maintenance of airworthiness in service.

— AR-21	Certification procedures
— AR-23	Airworthiness regulations of normal, utility, acrobatic and commuter category aircraft
— AR-25	Airworthiness regulations of transport category aircraft
— AR -29	Airworthiness regulations of propeller-driven transport category vehicles
— AR -31	Airworthiness regulations of air balloons
— AR-33	Airworthiness regulations of aircraft engines
— AR -35	Airworthiness regulations of propellers
— AR-36	Community noise standards
— AR-39	Airworthiness directives
— AR-145	Repair stations
— AR-183	Aviaregister's representatives.

The AR listed below are at different stages of elaboration:

— AR-27 Airworthiness regulations of propeller-driven normal category aircraft
— AR- Airworthiness regulations of super light aircraft
— AR- Airworthiness regulations of powered deltaplans
— AR-34 Engine exhaust emission requirements
— AR- Airworthiness regulations of auxiliary engines
— AR- Requirements for airship airworthiness.

The procedure of elaboration of Aviation rules implies their approval at the Council for aviation and use of air space and subsequent putting into force by each Contracting state.

Proceeding from certification results, the AR is to issue:
— aircraft, aircraft engine and propeller type certificates;
— production certificates;
— product fitness certificates;
— noise certificate;
— experimental aircraft certificates.

2. MAIN RESULTS OF AERONAUTICAL ENGINEERING CERTIFICATION

2.1. Aeronautical engineering certification

Over the period of 1992-95, more than 100 types and modifications of civil aeronautical engineering (aircraft, engines, auxiliary propulsion systems and propellers) were certified at different stages.

Based on the certification results, the AR of the IAC issued more that 50 type certificates to manufacturers of aeronautical engineering.

The organization of certification and certification procedure of advanced domestic civil aeronautical engineering is a priority activity of the IAC AR. First of all, it concerns Il-96-300, Tu-204, An-124, Su-29 aircraft, Ka-32A, Mi-26TC, Mi-34C helicopters, PC-90A, D-36, D-136, M-14 engines. Their certification has been completed with success and respective certificates are issued for them.

At the present time, the activities are undertaken on certification of modifications of these aircraft: Il-96M, Il-96T, Tu-204-100, 120, 200, 220, as well as on certification of a number of new projects: Il-114, Il-103, Tu-330, Tu-334, An-38, Be-200 aircraft, Mi-171, Mi-172 helicopters etc.

Permanent support of certified aircraft in service in order to maintain their airworthiness and further extension of possible applications is a very important and labour-intensive activity line of the AR.

Within this activity line, major changes in aircraft design and Service documents are being certified with a subsequent issue of respective supplements to type certificates. In total, the AR has issued about 70 of such supplements.

Typical major changes being implemented in numerous certified aircraft during their service are as follows:
— increased positive and negative temperature range;
— increased take-off and landing weight;
— increased flight altitude and range;
— installation of novel navigation equipment;
— increased specified service life;
— reduced meteo minimum;
— conversion of passenger version into cargo one and vice verse;
— increased passenger capacity.

Since the time of putting Il-86 aircraft into service, 27 major changes have been implemented, the scope of certification tests required for them comparable with that of initial certification. Respectively, 27 Supplements to the type certificate are issued for the aircraft. Similar situation takes place for Yak-42 aircraft: 22 Supplements to the type certificate are issued. During 2 years of the Il-96-300 aircraft operation, 5 major changes have already been implemented (increased passenger capacity, novel inertial system, increased take-off mass etc.). Five supplements to the certificate are awarded for PS-90A engine. Major changes are also being introduced into other aircraft and their engines. As a result, the aircraft characteristics become improved, the possibilities of their applications rise and the cost-effectiveness and competitiveness enhance with obligatory compliance with effective airworthiness regulations. In some cases, it is followed by improved flight safety, for example, it concerns the certification of Il-96-300 and Il-86 aircraft equipped with up-to-date collision warning system TCAS-II which makes it possible to continue scheduled flights of the aircraft in the air space over the USA.

In the last few years, the scope of aircraft noise certification has increased considerably. Based on the results obtained, noise certificates are issued essentially for all aircraft having the type certificate and for a number of aircraft awarded previously by airworthiness certificates (Il-86, Il-96-300, Tu-204, An-124, An-74, An-32P, Su-29, Ka-32A, Mi-172, Mi-34C, Il-18, An-24, An-12, Yak-40 etc.), as well as for some foreign aircraft types. In total, about 30 noise certificates are issued.

Up to 1995, the aircraft were certified in noise in conformity with the ICAO requirements (Supplement 16). In March of 1995, the Council for aviation and use of air space approved Aviation rules -36 which are harmonized with the requirements of both ICAO and FAR-36. The first certificate on the compliance with -36 is issued for An-32P aircraft.

In the interests of airlines of member-countries of the Minsk agreement, some types of foreign aircraft (Boeing-737, 747, 757, 767, A-320, BAe-125 etc.) are also certified.

2.2. AERONAUTICAL ENGINEERING PRODUCTION CERTIFICATION.

This work was started in 1992 after Production Certification Rules -21 (subparts F and G) had been worked out and put into practice. Then IAC AR issued directive letters and standard program on production certification, which are the basis of programmes elaboration and concrete aircraft types production certification.

According to the results of the work, performed by IAC AR, production approving certificates are given to 11 types of aeronautical engineering, including:

- Il-86 and Il-96-300 airplanes (Voronezh city);
- An-124-100 and Tu-204 airplanes for "Aviastar" stock company (Ulyanovsk city);
- D-36, D-136, TVZ-117 engines for "Motor Sitch" (Zaporozhye city).

At present time AR specialists perform certificative works at machine-building plant (Lukhovitsi town), "Permskie Motori" stock company and "Vpered" plant. Irkutsk and Novosibirsk Aviation Industry Unions, Kazan and Ulan-Ude helicopter plants, instrument-making plant in Ramenskoye town and Pratt & Whitney/Klimov enterprise (St.-Petersburg) are on the preparation stage for certification.

Jointly with Department of Air Transport (DAT) RF preparation for civil aeronautical engineering overhaul establishments certification has been started.

Essential attention was paid to further development of normative and methodic documentation on production certification, in particular:

- Requirements to quality check systems and aviation industry establishments' reports and accounts.

Editorial changes have been inserted into subparts F and G of "Production certification" of ARs -21 and methods of productions estimation.

- Projects of subparts O, K and L ("Aircraft spare parts production approval" and "Airworthiness export certificates issuing") of ARs -21 have been worked out.
- Requirements to major producers's work with import equipment suppliers have been formed.
- Appropriate documents on activities organization of AR Regional representations have been prepared.
- Regulations on AR and Independent Inspection cooperation at industry establishments have been submitted.

3. INTERNATIONAL ASPECTS OF CERTIFICATION.

Three systems of aeronautical engineering certification have been formed in aviation practice:

- The USA (FAA)
- West-European
- former USSR and CMEA countries.

Interests of world aviation market's development conditioned objective tendency to the integration of these three schemes, this tendency has greatly heighten for last period. The intensity of contacts has increased between USA Federal Aviation Administration (FAA) and Joint Aviation Administration of Western Europe countries (JAA), directed to full harmonize of systems and requirements of ARs, FAR and JAR and performance of joint certification.

The problem of mutual acknowledgement of certification systems of the USA and the USSR was officially raised at the intergovernmental level in 1990. Then practical works were started in the trend of obtaining of bilateral agreement on airworthiness. First, this work was headed by Gosavianadzor of the USSR, and since 1992 - by IAC.

The content of the work is mutual study of certification systems on these main trends: laws, sets of aviation rules, competent bodies, supervision of concrete products certification ("shadow" certification). Such products are Il-96T and Il-103 airplanes of Ilyushin plant.

More than 50 meetings were conducted on the level of administrations, managers and technical specialists, where a lot of problems, concerning laws, norms agreement, additional requirements to the subjects of "shadow" certification, were discussed.

The work on mutual estimation of aeronautical engineering certification systems is close to completion. Preliminary results allow to make conclusion about mutual principal acceptability of these systems.

As a result of already performed works qualitatively new level of mutual understanding is obtained for today. It can be said that the longest and the most difficult part of the way to obtaining the agreement on airworthiness has been covered.

Great significance of this work for both sides and the necessity of its completion acceleration were emphasized in the interstate memorandum, signed on 30 June 1995 by V.S. Chernomyrdin and A.Gore, which was mentioned in the first part of this report.

Simultaneously with this contacts with Joint Aviation Administration (JAA) were conducted. As a result of conducting several meetings has been worked out the Manual for JAA specialists in certification of aircraft, produced in CIS countries; the Manual comes to an agreement with IAC AR. Thus conditions for starting practical certification of home aeronautical engineering in Europe were prepared.

The development of Russian aviation industry establishments' links with western partners demanded:

— Preparation of Working Agreements on production of aeronautical engineering, supplied to CIS countries, with Aviation authorities of some countries. A sort of Agreement was signed by Canada Transport Ministry on supervision of spare parts for 6A-65, engine production at Pratt & Whitney (Canada) company. Similar Agreements are being prepared with Aviation authorities of Germany, England, France.

Within the framework of measures on conclusion of bilateral agreement between Russia and the USA, joint FAA/AR audits of aviation productions were performed in the USA (General Electric, Golfstream) and in Russia ("Aviastar" stock company), and also seminar of specialists in production certification.

For last several years IAC AR also took part in all important fora (forums) and meetings of Aviation authorities, industries and air carriers, devoted to the working out of politics on reapproachement and integration of requirements of airworthiness national norms. Conducted discussions and exchange of opinions, confirmed the correctness of chosen strategy of creation of the ARs system, agreed with acting worldwide rules systems.

At present time IAC AR undertakes efforts to increase leading home specialists' participation in the work of international meetings, devoted to agreement of airworthiness requirements in specialized fields, i.e. avionics, strength, materials etc. The purpose of these efforts is working out unified requirements for the RF, taking into consideration our positive experience in norming airworthiness.

AIRWORTHINESS CONTINUATION OF AVAILABLE AIRCRAFT FLEET

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Air transport is a part of the common transport system of Russia. It solves problems of passenger, mail, goods transportation and also other problems of national economy. At present air transport of RF numbers about 8100 aircraft.

These aircraft are operated by 421 independent aviation establishments and companies. 201 of them were created on the basis of the former state detachments, 14 were created on the basis of the former Aviation Industry Ministry establishments and 178 ones are new.

The main part of aircraft, operated now, were produced in 1960/70s. They have practically exhausted their specified service life. Only several specimens of the new generation airplanes, i.e. Il-96-300, Tu-204, An-124-100, are being operated now.

More advanced aircraft, i.e. Tu-334, Tu-334-200, An-38, Yak-242 airplanes, Ka-226 and Mi-38 helicopters, are being tested or are still being designed.

Most aviation enterprises and companies can not buy the new generation aircraft because of their high prices. Due to this fact they have to carry out large-scale and labour-consuming work to continue available aircraft fleet airworthiness state. In this case meet serious troubles, associated with purchasing appliances, fuel, oil, etc. whole and aviation enterprises taken separately. For example total productivity of all types of aircraft has decreased; average year flying time of all aircraft has also decreased. Effectiveness of jet fuel using has worsened for 6.2%; the portion of aircraft standing idle has increased.

The main reasons for aircraft's demurrage are the following:

- — absence of spare parts; due to this reason mainly Mi-2 and L -410, produced abroad, stand idle;
- — absence of engines; due to this reason mainly An-28 (25.8%), IL-62 (12.2%), IL-76T (11.5%) and IL-86 (18.4%) airplanes stand idle;
- — insolvency of aviation establishments, whose aircraft are at overhaul plants;

The main part of aircraft, as aforenamed, have almost exhausted their specified service life. Due to this fact 48% main airline airplanes, 59% local-service and domestic airlines airplanes, 63% transport airplanes will have been written off by 2001.

All aforenamed makes Department of Air Transport (DAT) solve the problem of maintenance of proper airworthiness level of aircraft fleet in the shortest time.

It is considered that airworthiness continuation of an aircraft is a complex of measures, due to which the aircraft fits the corresponding requirements of airworthiness at any moment of its time of work; technical condition of such aircraft ensures its safe operation.

This complex of measures is a complicated system, which, first, includes the program of airplanes maintenance. All standard types of works on aircraft and dates of the works' fulfilment are listed in the program. Special program is worked out for each aircraft. All programs must be worked out on the basis of regulation documents, similar for all aviation enterprises. All aviation enterprises, operating aircraft, must fulfil the requirements, stated in these documents. When these programs are worked out, they are examined by authorized state department and are to be approved by it.

At present time all maintenance works are regulated by "Civil Aviation Aircraft Engineering Operation and Overhaul Direction", published in 1993 (in Russian: НТЭРАТ ГА-93).

Second, the system must contain a list of enterprises, which have an appropriate licenses to perform works on maintenance, given by an authorized state body.

Such licenses were given by Civil Aviation Ministry in the USSR, (CAM; in Russian: МГА). Before the license had been given, authorized bodies of Ministry organized and performed the preparation of the organization, which wanted to get the license. Then CAM regional organizations checked the preparation and servicing quality on every maintenance type at the initial stage of the organization activities. Only after successful completion of all these measures the organization could get appropriate certificate on aircraft maintenance.

At present time, as most aviation enterprises are not controlled by the State, the above described system practically doesn't work. In 1991 we started the work on performing certification of air carrier enterprises and aircraft maintenance and overhaul enterprises. We also take into consideration international experience, which aviation enterprise establishments and airlines have got in this field.

Main stages of this work are:

— 1992/93 — working out main normative and methodical documents, containing certification requirements to enterprises, performing aircraft maintenance and overhaul to personnel working at these establishments and quality checking system organization of aircraft maintenance at these enterprises. During the same period "Certification Manual" and "Certification Methods" were worked out.

- The beginning of 1994 — aircraft certification organization structure of aircraft maintenance was established.
- The end of 1994 — practical certification was started.

RF Government instructed DAT of RF Transport Ministry to certify Civil Aviation. Concrete work on certification of aircraft maintenance and overhaul is performed by Engineering Operation and Overhaul Brunch of DAT.

The purposes of RF Civil Aviation certification are the following:

a. The subject of certification must confirm, that it observes the requirements on securing flight safety, determined for it by the state bodies; it must also confirm its right to posses "The Correspondence Certificate", which is given by an authorized state body.
b. State bodies perform supervision and regulation of organizations and specialists, providing aircraft maintenance and overhaul. This purpose is very important because at present time a lot of new organizations, operating aircrafts, appear.
c. Rise of the role and responsibility of organizations and specialists, providing aircraft maintenance and overhaul, in the field of securing flight safety and continuation aircraft worthiness.
d. Agreement of norms and rules, which are in force in RF Civil Aviation, with appropriate international documents.

Air transport is a type of transport, which must be better adopted to constantly developing engineering rather than other types. This development must be taken into consideration when preparing personnel, which performs aircraft maintenance.

Every aviation enterprise must have a program of personnel developing, which serves aeronautical engineering. It implies theoretical teaching, work on probation at a work place and an examination. Then head of the enterprise (plant, factory etc.) issues an order, in accordance with which the employees, who have passed the examination, get a personal document, confirming their right to perform aircraft maintenance at this enterprise. This program is still in force in RF aircraft industry.

New system of certification replaces the program, described above, which is typical for every enterprise, with a new one, identical for all RF enterprises and organizations, which perform aircraft maintenance and overhaul. In accordance with it, such organizations will get state powers after they are certified.

DAT has worked out the following documents, defining the certification procedure of such organizations:

- certification requirements for engineering staff;
- personnel certification manual;

— certification requirements for quality checking system;
— quality checking systems manual;
— certification requirements for organizations, which perform aircraft maintenance and overhaul.
— organizations certification manual, performing aircraft maintenance and overhaul;

This documents are in agreement with the same FAA's documents in general.

Certification elements are listed below:

1. Object of certification
   1.1. Aircraft maintenance and overhaul organizations
   1.2. Aircraft maintenance and overhaul linear stations
   1.3. Maintenance quality checking systems
   1.4. Staff
2. Concrete spheres of activities of maintenance and overhaul organizations, which must be certified.
   2.1. Aircraft routine (linear) maintenance
   2.2. Aircraft periodical, labour-intensive and special maintenance; complex maintenance and overhaul.
3. The fields of activities of maintenance and overhaul organizations, which must be periodically inspected.
   3.1. Production management and organization of production
   3.2. Maintenance information provision
   3.3. Skilled manpower hiring
   3.4. Production base of organizations
   3.5. Maintenance and overhaul process (general, special, avionics etc.)
   3.6. Quality checking systems
   3.7. Technical resources and reserves of organizations
   3.8. Organizations financial provision

Successful work of air transport in many respects depends on the work of aircraft maintenance and overhaul system.

Flight operating safety, regularity of operations and aircraft reliability directly depend on qualitative work of this system. Besides, well-working maintenance system allows to improve upon use of aeronautical engineering. The qualitative work of the system, in its turn, depends on following internal parameters of the system: level of production organization; engineering level of production; concentration and specialization of production etc.

At present time aircraft maintenance system consists of two self-dependent sub-systems which are loosely held to each other. They are: sub-system of Aircraft Maintenance Bases (AMB) and sub-system of Aircraft Repair Plants (ARP).

As a rule, AMBs are incorporated into plant engineering and maintenance service of air carrier enterprises. The AMBs perform maintenance of the enterprises' own aircraft fleet, transit aircraft and aircraft, which are at served by the enterprises at present time. The role of AMBs in provision of high level aircraft reliability and serviceability is very important and also in securing maximum effectivity of airplanes and helicopters operation. The performance of aircraft directly depends on annual hour's flying time; in its turn, the less repair time of aircraft, the more flying time.

Unfortunately, afore mentioned internal parameters are, as a rule, on poor level. The maintenance organization's engineering resource and reserve base must be developed and improved for the parameters to correspond to demands of successful operation of new generation aircraft.

The most important factors, that allow to develop maintenance and overhaul bases, are the following:

— consistent and steadfast integration of aircraft maintenance and overhaul;
— production concentration;
— making up of large-scale centers of maintenance and overhaul, both branch and regional;
— effective exploitation of existing working areas;

Such development of material-engineering resources in the first place must be oriented to new generation airplanes maintenance, i.e. IL-96, Tu-204, Tu-334, IL-114.

It should be mentioned, that organizational and structural changes in the of RF Civil Aviation, being carried out at present time, are bound up with the increase of enterprises types, which perform aircraft maintenance and overhaul. All changes are carried out on the basis of complex programs of aircraft maintenance and overhaul system development, which we have worked out.

At present time the development of engineering resources and reserve base of aircraft maintenance and overhaul is carried out mainly by means of building large enterprises. These enterprises, called Aviation-Engineering Bases (AEB), will be supplied with modern equipment and will have large size hangars. In future, such AEBs will form the basis of branch and regional aircraft maintenance and overhaul centers, which will have the independent organization status. Thus, after the development period completion, four types of organizations, performing aircraft maintenance and overhaul, will be created:

— Already existing, standard AEBs, which are not independent juridical and/or financial persons;
— New type Aviation-Technical Bases, which are independent juridical and/or financial persons;
— Aircraft maintenance and overhaul centers, working as members of integrated system of aircraft maintenance; aircraft general overhaul isn't performed at these centers;

— Cooperative, lease and stock companies, specializing in performing separate types of works on aircraft maintenance.

It should be taken into consideration, that establishments of one type can be called, depending on the situation, either "Aviation-Engineering Base" or "Aircraft Maintenance and Overhaul Organization".

To continue aircraft worthiness at high level, it is necessary to perform periodical validity period prolongation of "Airworthiness certificate". This requires observing the following requirements:

— concrete aircraft design must meet the requirements of this type aircraft design;
— technical documentation of every concrete aircraft must correspond to physical design and really maintained equipment;
— all changes of equipment components and/or aircraft design must be done in time; all tests and inspections must be done in time; appropriate notes must be made in log book;
— equipment and systems operating time registration must be carried out correctly; equipment, which has exhausted its specified life, must be replaced in time.

Besides, it is necessary to gather information about faults, failures and other defects of aviation engineering, analyze it and save in databases. These databases must be certified too. As one of the bases of securing flight safety is integrity of aircraft framework, measures on framework troubleshooting are of great importance. Among these measures, first, introduction of nondestructive inspection procedures of aviation engineering condition must be named.

At present time the number of such procedures in the system of aircraft maintenance and overhaul is insufficient for proper securing flight safety. The number of such procedures must be increased, and their use must be made more effective.

In conclusion, it should be said, that we have worked out theoretical and methodical bases of aircraft resources and service life prolongation, to be able to solve the problems of continuation of aircraft worthiness at high level. Mathematical models of operating and engineering characteristics of aircraft flight and navigation systems and complexes and engineer methods of aircraft resources increase have also been worked out. The use of these methods allows to start such kind of operation of great amount of different types of aircraft, that it allows to shorten their demurrage, entailed with performance of maintenance and/or overhaul.

TSAGI WORK ON CERTIFICATION OF CIVIL AVIATION ENGINEERING

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In the first years after the foundation TsAGI began to deal with all problems of aviation science and engineering from basic research to design and building of their own aircraft.

Later a number of research institutes (NIAT, VIAM, CIAM, LII in 1940) as well as a number of design bureaus, for example, the Tupolev Design Bureau which is familiar to the whole world, were formed mainly on the basis of TsAGI. This has resulted in narrowing the scope of the problems on which TsAGI continues working in the field of aircraft technology and at present the range of work is still rather large, namely:

— aero - and hydrodynamics,
— strength, service life and aeroelasticity,
— aerothermodynamics and gas dynamics,
— aeroacoustics,
— advanced aircraft
— unique experimental facilities.

Thus, a list of civil research aircraft objects, namely: airplanes (hydroplanes) and helicopters as a whole and specific assemblies, for example, such as control systems, aircraft propellers and helicopter rotors, air inlets and exhaust assemblies of power plants and at last structure components (load carrying elements, thermal protection and sound absorption and so on) was determined.

At the same time the directions of work were also determined:
— basic research,
— advanced research,
— research in specific aircraft types
— certification.

Before we get down to description of certification work performed by TsAGI it should be noted that as far back as 1924 a decree under which no one aircraft for the first time could fly without a corresponding permission of TsAGI, was issued. In essence, this meant the beginning of TsAGI's work on certification of aeronautical engineering and at present this work is being proceeded.

Sometimes people think that certification is a final stage of the aircraft development. Indeed, the type certificate is given when the whole investigation cycle is accomplished. However, work on certification is performed at all stages of the aircraft development beginning with the initial concept and some final results can be obtained or to say more properly they must have been already obtained at the earliest stages. For example, it is impossible to carry out design work on the aircraft strength without establishing the values of the design velocities V_C and V_D in accordance with requirements of the Aviation Rules. It should be also noted that the methodology adopted in Russia for providing the structure service life allows for establishing the prescribed service life stage-by-stage (in parts: 20 — 25% of design value). That is why, by the moment when the first airplanes (helicopters) approach to the established service life an additional work cycle including analysis of the operation experience (real operation conditions, detected defects) and additional calculation and experimental study, has been performed and on their basis the conditions for providing the operation safety with respect to the service life for the next stage are specified more exactly. This procedure is repeated many times (as a rule, over the designed service life) until further increase in service life is economically reasonable or safe. Every extension of service life is considered as a main change in operation conditions and allows for granting a supplement to the Certificate of the type. Thus, the certification procedure covers all stages of the aircraft development and tests and as to structural problems it extends to the mass operation stage.

The investigation methods used at TsAGI for conducting the certification work involve:

- calculations,
- testing models,
- testing structural materials,
- testing structure elements and parts,
- testing full-scale assemblies and aircraft as a whole.

As seen from the above-mentioned, tests are one of the main components of certification work, they complement and make specific a calculational study. That is why, TsAGI always paid a great attention to development of their test facilities, many of them are unique. At present there are 7 test centers at TsAGI (including branch), such as:

- Aerodynamics,
- Aerothermodynamics,
- Dynamics,
- Strength,
- Aeroacoustics,
- Hydrodynamics,
- Metrology.

At this conference the chairman of the Aviation Register of the Interstate Aviation Committee Mr. V.V.Sushko on behalf of the Register and the State Standard Committee of the Russian Federation officially presented the accreditation Certificates to the first six test centers (Metrology Test Center will receive the Certificate somewhat later). So, it would be reasonable to dwell on description of these centers, their brief features are presented in tables (fig. 1—7).

While considering these tables one should pay attention to the following: the tables (fig. 1 and 2) show that the TsAGI test facilities allow us to perform aerodynamic model investigations (and some wind tunnels provide full-scale tests of small aircraft and their specific assemblies) throughout all flight speeds which will be operational for modern and future aircraft of all types and purposes. And the interesting thing is that the test centers include only those aerodynamic facilities where so called industrial tests are performed and their results may be used for certification of the specific aircraft types but such centers have no facilities designed for scientific and research purposes (30 facilities).

The center of Dynamics (fig. 3) is designed in the first turn for forming ideology and developing structure, parameters and specific elements of modern automatic aircraft control systems. Naturally, the knowledge of the nonstationary aerodynamic aircraft characteristics is one of the basic components of such work.

The center of Strength (fig. 4) provides performing the whole complex of experimental work, beginning with the material certification and model tests and completing with the structural certification tests of full-scale structures with regard to environmental, acoustic and other effects. Along with determination of static and fatigue characteristics of structure investigations of dynamic aircraft characteristics are performed taking into account the influence of automatic flight control systems.

The center of Aeroacoustics (fig. 5) performs a cycle of work on investigation and suppression of noise produced in flight both for the purpose of meeting the strict ICAO certification community noise requirements and providing the passenger comfort in the passenger compartment.

The center of Hydrodynamics (fig. 6) provides investigation of loading conditions and stability and controllability characteristics of hydroplanes and amphibious aeroplanes when flying over the sea surface and also performs certification tests of aircraft and their models during emergency water landing.

At last, the center of Metrology (fig. 7) provides the unity and accuracy of measurements performed in the above-mentioned TsAGI test centers by means of the periodical metrological certification of all test benches possessing for this purpose a necessary set of primary standards and reference devices.

Naturally, all certification work of the test centers is done by TsAGI scientists and engineers. Furthermore, this work is carried out under the leadership or control of the auditing experts appointed by the Aviation Register in accordance with the AP-183 requirement. The same specialists make up not only reports on calculation and experiment results and also methods of performing this work, Regulations and Instructions on organization and order of their performance and also proposals of drafts (or amendments) of the Aviation Rules chapters which are within the TsAGI competence (AP -23, AP-25, AP-27, AP-29, AP-33) and methods of correspondence to these rules. Along with these documents a set of conclusions on aerohydrodynamics, dynamics, strength and aeroacoustics, summarizing the results of work at some aircraft development stages: at the stage of the preliminary design, before flight tests, at the stage of the state and operational tests, is prepared. As to the main certification documents, namely: a conclusion on meeting the strength requirements with respect to the aircraft (and a number of its assemblies) in accordance with the Aviation rules (for preparing the certificate of the type) and conclusions concerning the maintenance of airworthiness during operation when increasing the service life step-by-step, these documents are granted by the "TsAGI-TEST" Certification center formed by the Aviation Register in accordance with the AP-21 Aviation rules. The same center performs coordination of the certification bases of the civil aircraft.

Finishing this presentation it should be noted that the absence of the severe accidents with the domestic civil aircraft because of the structural failure for more than 20 years testifies to the high quality of the TsAGI work on the certification investigations of the domestic civil aircraft and in particular with respect to the strength conditions where TsAGI is a main expert of the Aviation Register for all these years.

We hope we shall be able to keep this safety level for the next years in spite of division of the common Aeroflot into a large number of middle and small airlines and following from this a natural reduction in the level of the aircraft maintenance.

"Aerodynamics" Test Center performs tests of:

1. Full-scale aircraft and their large-scale models.
2. Aircraft models in the take-off/landing regimes and at the low flight speeds.
3. Models of aircraft and their elements at sub- trans- and supersonic speeds.
4. Aircraft models for spin.
5. Engine intakes/nozzles.
6. Propellers and rotors.
7. Conveyance models, buildings and facilities.

The test center has **20** wind tunnels.
Performs tests at Mach numbers from **0.1** to **4**.
Dimensions of objects tested are from **0.4** to **16** m.

Figure 1.

"Aerothermodynamics" Test Center performs:

1. Aerodynamic/thermal tests of models at hypersonic speeds:
   Mach numbers $\mathrm{M} = 4\text{—}20.5,\ T_{max} = 2600$ K.
2. Aerodynamic/thermal tests of models and structure components in upper atmosphere flight conditions
   $V_{max} = 8$ km/s, $T_{max} = 6000$ K.
3. Determination of ionospheric aerodynamic and plasmadynamic characteristics of aircraft
   $V_{max} = 10$ km/s.
4. Tests of jet — hypersonic aircraft components interference
   $\mathrm{M}_{max} = 20,\ T_{max} = 2000$ K.

The test center has **7** wind tunnels and facilities.

Figure 2.

"Dynamics" Test Center performs:

1. Half-scale simulation of the aircraft flight on piloted simulators with the pilot participation in the control loop.
2. Determination of dynamic characteristics of control systems and their components on simulators.
3. Determination of unsteady aerodynamic characteristics of aircraft and their component models at sub/supersonic speeds in TsAGI's wind tunnels.

The test center has 4 piloted simulators and a simulator complex for testing hydraulic actuators and control systems.

Figure 3.

"Strength" Test Center performs:

1. Structural integrity/endurance tests of specimens, structure components and full-scale units, including the tests with the simulation of acoustic, thermal and climatic actions.
2. Static strength, endurance and survivability tests of full-scale aircraft as a whole.
3. Determination of dynamic characteristics of aircraft and control systems on models and full-scale objects.

The test center has:

— a complex of testing machines,
— a number of facilities for testing structure components and units, including the facilities for simulation of acoustic, thermal and climatic actions;
— halls for structural tests of full-scale objects,
— mobile laboratories for dynamic tests.

Figure 4.

"Aeroacoustics" Test Center performs:

1. Noise measurement on terrain and in passenger airplane and helicopter compartments.
2. Airplane structures sound insulation/absorption effectiveness determination.
3. Calibrations and certification of acoustic measuring equipment for aeroacoustic tests of models and full-scale objects.
4. Determination of aeroacoustic characteristics of air blowers.

The test center has:
— anechoic/reverberation chambers with air flow;
— special-purpose instrumentation complexes;
— facilities for determination of sound insulation/absorption;
— characteristics;
— calibration stands;
— stands for aerodynamic tests of air blowers.

Figure 5.

"Hydrodynamics" Test Center performs:

1. Testing the amphibious airplane, ground-effect airplane and high-speed ship models on the still water and under disturbance conditions.
2. Testing vehicles and their models at emergency landing on water, models of high-speed underwater vehicles and vehicles crossing the water surface.
3. Full-scale hydrodynamic/flight tests of hydroplanes up to 10 tons take-off weight.
4. Testing ground-effect airplanes and their models.

The test center has:
— a test tank with systems of recording/processing equipment;
— a water base at Moscow Sea for performing tests under full-scale wind -wave conditions;
— a seadrome,
— a number of specialized stands providing the simulation of hydrodynamic processes at speeds of moving up to 200 m/s

Figure 6.

"Metrology" Test Center

provides traceability and accuracy of measurements by metrological certification of test centers' simulators:

Aerodynamics	Strength
Aerothermodynamics	Hydrodynamics
Dynamics	Aeroacoustics

The test center has a base of a system of standards and reference means for measuring:

Mass	Temperature
Force	Linear-angular quantities
Pressure	Optical quantities
Strain	Air flow velocity

Figure 7.

LII EXPERIMENTAL FACILITIES FOR TESTING AND CERTIFICATION

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The Flight Research Institute (LII) experimental base for testing and certification of aircraft and its systems is formed in accordance with the Institute is tasks, which are formulated in fig. 1.

In the first place, it is a test center, which is the best complex of unique test airdrome in Europe, equipped with all required experimental base for aircraft testing: system of trajectory and radio telemetry measurements, complex of all modern radio engineering and satellite navigation and landing means, flight experiment control and automated data processing systems, special flying testbeds for certification of prototypes and airdrome facilities.

Secondly, it is the experimental base (flying testbeds and benches) for development of prototype components on the subject, presented in fig. 1.

Thirdly, it is flying testbeds and benches to carry out fundamental researches.

The effectiveness of combined approach, which was worked out by LII, for experimental base using in case of development and flight testing of prototypes is the most clearly revealed on the example of the Tu-204 and IL-96-300 airplane development (fig. 2).

In this case this base was used:

— to support flight testing of these airplanes;
— for advanced system development on flying testbeds;
— to carry out a number of tests on certification assessments by LII specialists with using of the LII base.

The characteristic example of flying testbeds using in works which are carried out on the Tu-154 flying testbed in the interest of Tu-204 certification.

The Tu-204 airplane is the first civil aviation airplane with the mini-control wheel, so-called integral digital control low (which provides an automatic limitation of flight parameters and a number of other principal features of airplane controllability) as well as a glass cockpit.

In this conditions the flying testbed had become the effective tool which allowed to simulate dynamics. Controls and display system of a designed airplane in advance, to provide its qualitative and quantitative assessment in real flight conditions within certification requirements, and in a case of necessity to carry out an optimization of airplane and its systems characteristics.

This enabled to reduce a number of modifications during the Tu-204 airplane certification. The diagram of flying simulation complex on the base of Tu-154M airplane and its cabin interior is shown in fig. 3,4.

The glass cockpit and experimental control stick are shown on fig. 4.

Fragments of similar researches were carried out in the interest of Il-96 and An-70 airplanes.

Specialized flying testbeds of LII provide advanced flight tests of new engines and power plants which are an integral part of a new engine and new aircraft development. Especially, the significance of flight test on flying testbeds was increased for by-pass turbojet engines and prop-fan engines with high air flows, altitude test rigs of which are absent, and for power plants of maneuverable airplanes, power plants and engine characteristics of which greatly depend on design features of an aircraft.

At present time LII has flying testbeds on the base of IL-76(4), Tu-22, An-12, Tu-16(2), Tu-134 airplanes and MiL-17 helicopters, which are designed for flight researches of any dimension engines, auxiliary power units.

The characteristic example of flight test structure and volume on flying testbeds is advanced and accompanying flight tests of D-18T engine on IL-76 flying testbed (fig. 5), which were carried out without tests of this engine on the altitude test bench (see fig. 6).

A complex of works was carried out on the flying testbed to support the maiden flight and the first phase of flight tests of the RUSLAN prototype, as well as flight tests were carried out completely: to define basic data (100%), flight endurance tests (100%), strain gauging (100%), automatic compressor control system (80%), gas-dynamic flow stability (80%), starting (80%), oil system (60%).

Now flight tests of the D-27 propane engine on the flying testbed IL-76 are being carried out, which is intended to be installed on An-70 airplane.

To test an auxiliary power unit (APU) in the airplane configuration it is also used universal flying testbed on the basis of the An-12 airplane, which allows to carry out flight tests of auxiliary gas-turbine engines and their systems flight tests with almost full simulation of limit environmental conditions.

The most full using of flying testbeds in case of power plant certification can be illustrated on the example of PC-90A engine flight tests.

To support the certification of PC-90A engine which is a part of IL-96-300 ant Tu-204 airplane power plant, 105 flights were performed, including 61 flights to prove compliance with requirements of NLGS-3, § 6.5.6, chapter 6 "before installation in airplane" and 44 flights for NLGS-3 § 6.6, chapter 6.

As flight test type programs of power plants on flying testbeds have been perfected for all turbine engine modes of operations in expected operating conditions, in our opinion, using of flying testbeds during turbine engine certification allows to transfer 40% of the work volume in accordance with flight test type program from the basic airplane to flying testbed.

For development and certification of flight and navigation equipment the following flying testbeds were used:

a) Il-62 flying testbed for development and certification of the basic navigation complex (BNK-2P) and inertial navigation system I-11 in the interest of the IL-86 prototype;
b) An-26 flying testbed for development and certification of BNK-11 in the interest of Yak-42 and An-74 airplanes;
c) Tu-154 flying testbed for development and certification of ground proximity warning system for IL-86, Yak-42, An-74 airplanes, development of the windshear detection system algorithm, fuel management system, take-off monitoring in interest of navigation and control complexes for IL-96-300 and Tu-204 airplanes;
d) Tu-104 flying testbed for development and certification of navigation complex in subsonic area in the interest of the Tu-144 prototype;
e) Tu-144 flying testbed for development and certification of flight instrument complex for Tu-144 airplane and for reduction of its weather minima up to ICAO Cat. II. and a number of other flying testbeds, which are used to this extent or otherwise in the interest of the flight and navigation equipment system development of certified airplanes. A hundred of flights performed on these flying testbeds, including flights, results of which were accepted as certification works which must be carried out on prototypes on the phase of development and certification. These works on flying testbeds considerably reduced certification volume and duration of the these flight and navigation equipment complex systems on prototypes. For illustration (as an example) the table is shown (fig. 7).

Besides the above mentioned flying testbeds it is necessary to note the flying testbeds on the basis of An-26 and IL-76MD airplanes, on which satellite technology development were performed. In fig. 8. as an example, results of the curved approach on ICAO Cat. I are presented of An-26 flying testbed for satellite navigation system GPS in differential mode.

LII is constantly improving the capability of the test center in the interest of effective carrying out different aircraft, special complexes and aeronautical product systems testing. In fig. 9 the list of LII flight test center means is presented to support test flights.

Flying testbeds for various subjects are developed and used in LII during flight researches of aeronautical products.

One example of works using flying testbed and LII experimental base in the interest of the aircraft flight safety improvement is researches of active flight safety system development for many airplanes.

Main tasks of the system are as follows:

- preventing violation of operating limitation on flight speed and altitude;
- improving airplane resistibility to stall and spin.

In addition to the above mentioned flying testbeds it is necessary to note flying testbeds on the basis of An-26 and IL-76MD airplanes, on which the development of satellite technologies was carried out.

As an example it is shown in fig. 8 results of the ICAO Cat. I. curved approach which was performed on the An-26 flying testbed with the satellite navigation system of GPS type in differential mode.

LII is constantly improving the test center capabilities for the purpose of effective tests of various aircraft types, special complexes and aeronautical product systems. The list of LII test center means is presented in fig. 9.

Flying testbeds for various tasks are developed by LII and used during flight researches and tests of aeronautical products. One of work examples using the flying testbeds and LII experimental base in the interest of aircraft flight safety improvement is researches of an active safety providing system of the high-maneuverable airplane flight.

The main tasks of the system are the following:

- to prevent exceeding of operating limitations on flight speed and altitude;
- to increase the aircraft resistance to stall and spin.

The researches into the system development are performed by means of flight and simulation complex, which is shown in fig. 10, consisting of:

- flying testbed on the basis of Su-27 airplane with experimental control systems, flight information display,

exchange of telemetric and TV information with the ground system of flight experiment control point;
- flight experiment control system, which provides input of telemetric information and trajectory measurement data, their processing and flight information forming for information display system and command signals for ACS with subsequent transmission on TV channel to aircraft;
- ground test and development bench of hardware-in-the-loop simulation.

By means of this base the great volume of researches had been carried out, which allowed to develop the system conception, rational control algorithms, to assess its functioning and get the flight personnel assessment.

Flying testbeds for ecological researches are developed on the basis of Mil-17 helicopter and Tu-134 and An-12 airplanes (fig. 11).

Mil-17 and Tu-134 flying testbeds are equipped with bleed air systems and systems for concentration determination of CO, NO_x, H_xC_y, O_2, O_3 and aerosols. These flying testbeds are intended to assess an atmosphere pollution, for full-scale researches of physical and chemical processes in an airplane wake vortex, transboundary transition researches.

The flying testbed on the basis of An-12 airplane is equipped with the special equipment which provides to scan the ground surface within the wide band of radiation from the radar band to ultra-violet band.

These flying testbeds provide to carry out the ecological monitoring of land and water surfaces, atmosphere practically in any region of the Russian Federation.

The flying testbeds were successfully used for a number of works in the RF and abroad.

LII has also the powerful ground experimental base, including:
- ground base station of satellite navigation, which provide the high accuracy of navigation and landing systems in differential mode by means of correction determination and transmission of such data to aircraft;
- bench for both pilots ejection. Development of simultaneous two seat ejection for space planes with simulation of different cockpit attitudes;
- bench for seat tests of passenger airplane up to 30 g;
- base for preparation of flight strength tests. Calibration of load forces for helicopters and light airplanes;
- flight simulator for researches of passenger airplane control system dynamic and other benches and facilities.

An example of using the LII experimental base and developed special means is the program on assessment of the static and dynamic aircraft loading.

The main task of LII in the area of flight strength tests is to determine actual static and dynamic loads, which effect on airplane in flight, and to specify objective laws of the structure loading.

On the basis of flight test data design bureaus and TsAGI clarify design data and experimental materials.

The LII complex of special means (fig. 12) which is developed and produced by LII, is used for the assessment of dynamic and static aircraft loading. The determination of aeroelastic stability margin and flutter characteristics is carried out by means of aerodynamic or electrodynamic aircraft structure excitation system as well as systems using aircraft control surface deflection by specially formed signal through the automatic control system. The means of shock and mechanic excitation of electroimpulse eddy current excitation are used to assess vibroacoustic loading of aircraft structure elements.

The base was established in LII to support aircraft tests on static loading assessment of aircraft structure elements during multicomponent calibrated loading of strain-gauging airplane parts and other aircraft during its preparation for flight strength tests. Another example of the LII ground base using are tests of the airplane system resistance to effect of external electromagnetic fields which are mandatory in accordance with effective foreign airworthiness requirements. Experimental facilities developed by LII in combination with the appropriate measurement equipment provide tests on the effect of external electromagnetic fields in accordance with FAA requirements or certification bases, In particular the IL-96, Tu-204, IL-114 airplanes were tested on such environmental effects. The working episode of tests on the electronic engine control system resistance to the effect of external electromagnetic fields is shown in fig. 13.

The technology of aircraft and its system tests on the lightning and electrostatic protection which provides to carry out a work cycle to support the development of actions on the protection, its checking and issuing appropriate certification materials.

The technology provides tests on the lightning strike selectivity, determination of the current pattern, assessment of induced electromotive forces, checking of the actual structure element resistance to the full-scale lightning strike, and the final assessment of lightning protection on jet aircraft by means of the PIK-URAL simulation complex.

The working moment of airplane "Finist" model tests on the selectivity strike by means of the high voltage facility is shown in fig. 14.

Some words about radio links of digital information transmission.

ICAO decision on necessity to use in the system of information exchange "aircraft-airtraffic control service" the radio link of automated data exchange required to develop the new technology of researches and flight tests of digital information transmission radio link.

As main parameters of the data exchange radio link are specified by probability values, it is necessary to have a large volume of material to assess, obtaining of which in the flight experiment requires great expenses and time.

The combined assessment method of main parameters of the data exchange radio link provides work carrying out on flying testbeds and simulators (hardware-in-the-loop and software-in-the-loop simulation). The block diagram of the data exchange radio links is shown in fig. 15.

Works on aircraft noise certification have been being carried out in LII from the beginning of 1970s. As a result, 26 certificates have been obtained, which confirm the aircraft compliance with international requirements of the ICAO Annex 16. Aircraft certification tests in our Institute have a number of advantages:

- long runway (more than 6 km) provides to carry out tests in one place practically of all types of aircraft from very light aircraft to super large transport category airplanes;
- experimental base of the Institute is equipped with high accuracy trajectory measurement means that it is very important for such types of testing;
- practically, all new types of aircraft are tested on the base of our Institute. So there is the possibility to carry out noise tests along with the aircraft airworthiness certification that reduces the cost of these works.

Main Tasks of the Instutute

THE SUPPORT OF FLIGHT TESTING WITH THE EXPERIMENTAL FACILITIES	• Allocation of the Experimental Aerocomplexes and Bases • Traning of the Test-Pilots • Working out of the Flight Testing Techniques • Development of the Component Parts on Flying Testbeds and Stands • Conducting of the Most Complicated Types of Flight Tests • Participation in Flight Testing
LEADING RESEARCH	• Development of Flying Testbed • In-Flight Investigations of New Conseps • Development of New Flight Testing Techniques
PERFORMING THE FUNCTIONS OF THE LEADING INSTITUTE IN THIS BRANCH OF INDUSTRY	• Aeroplane Certification (MOS - the Technique for the Determination of Compliance with NLG-3; Participation in the Development of Rules and Certification Work, Community Nois) • Measurements and Processing During Flight Testing • Reliability and Maintainability • Decreasing of Meteominimum • Lightning Protection • Electromagnetic Compatability and Protection Against External Electromagnetic Fields • Antiicing Protection • Fire Protection • Stelthness • Systems (Fuel, In-Flight Refueling, Emergency Escape and Rescue, Operational Recoders)

Figure 1.

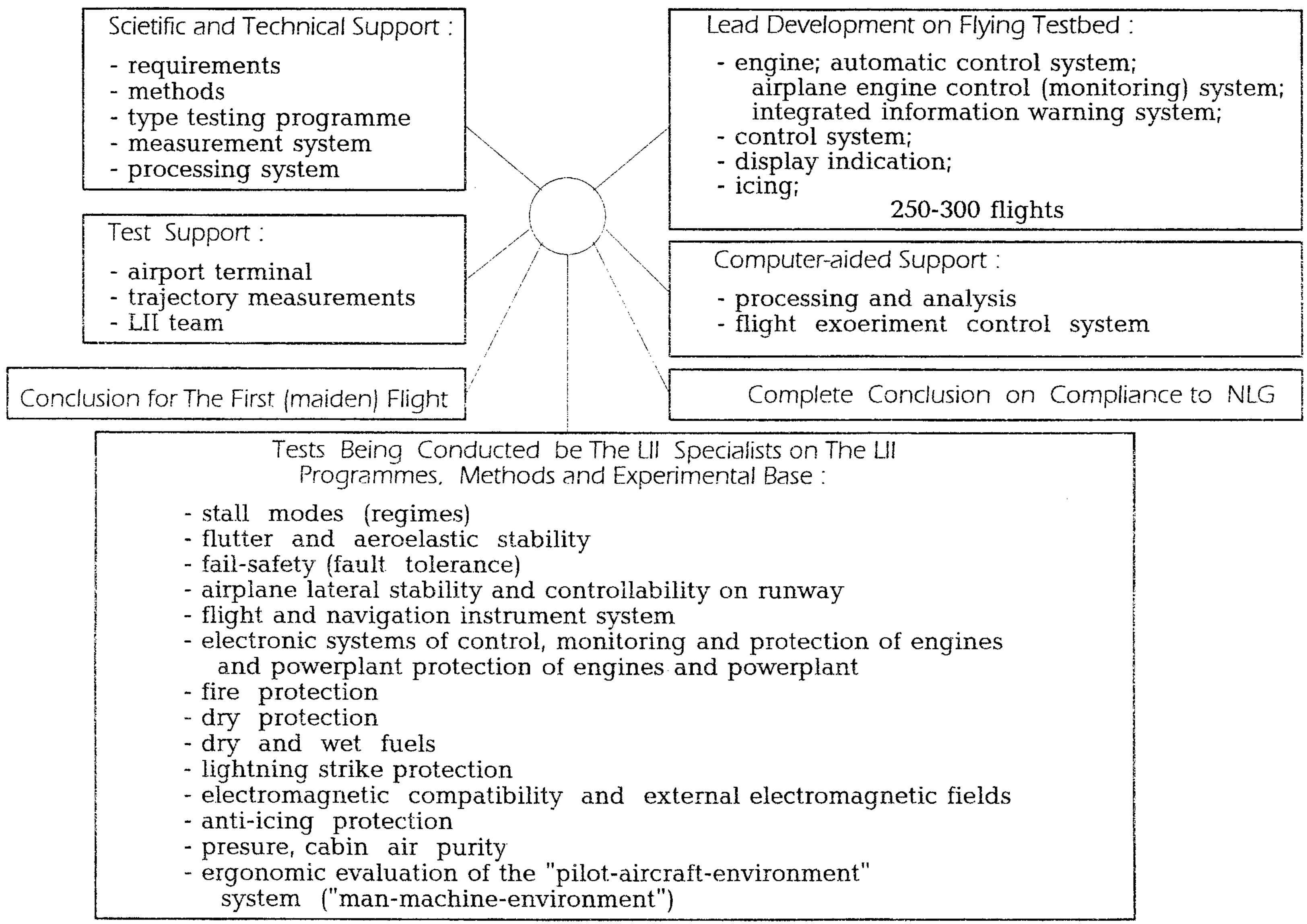


Figure 2. LII Contribution to Development of the IL-96-300 and Tu-204 airplanes

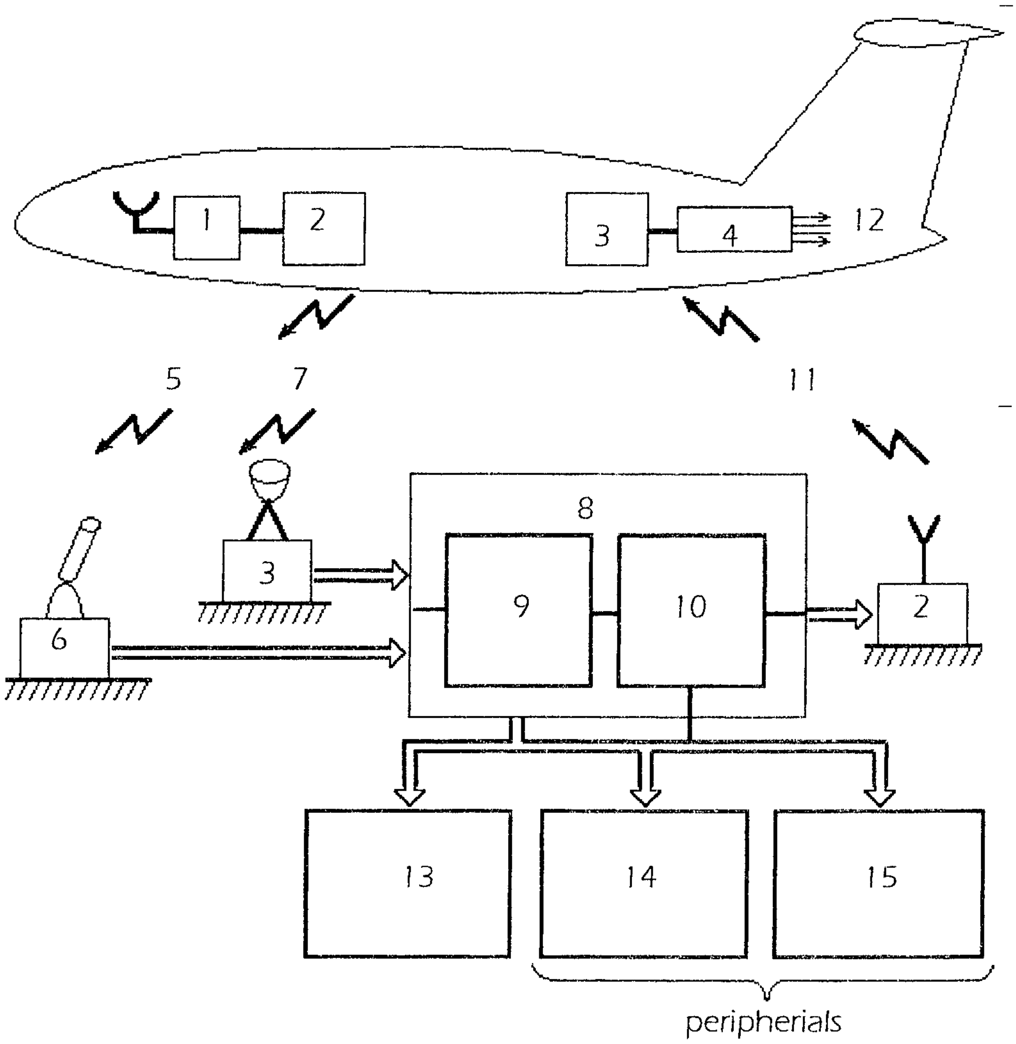
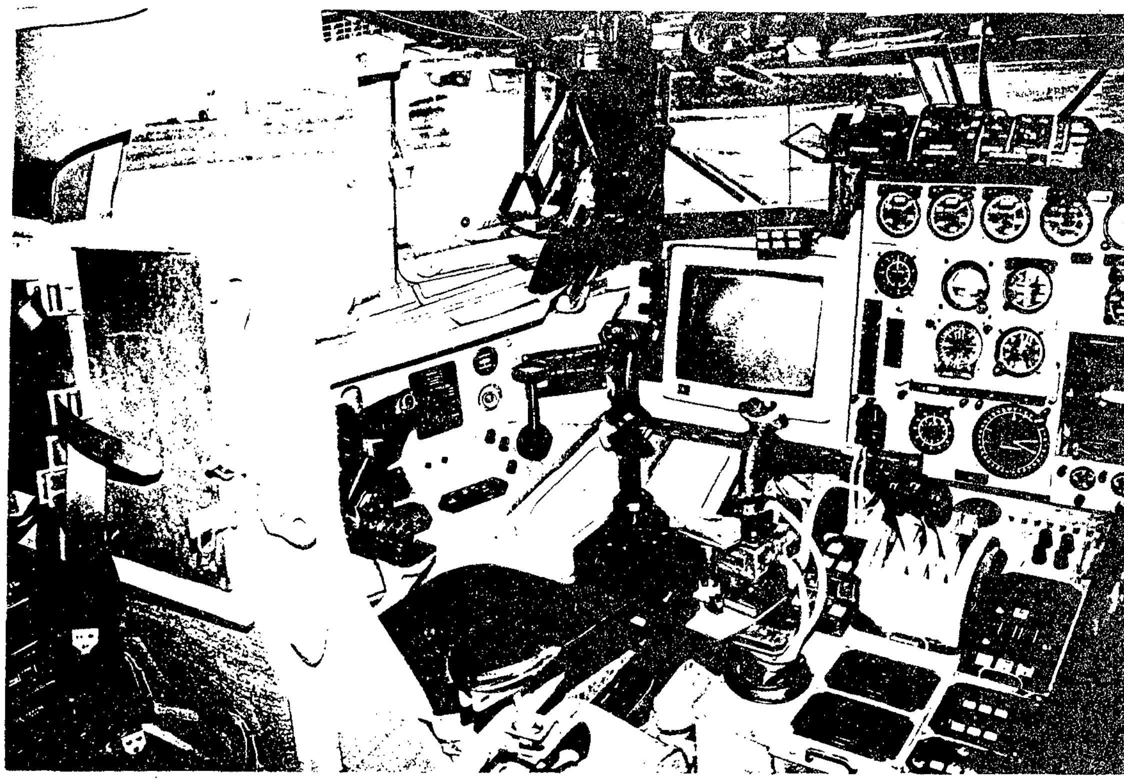
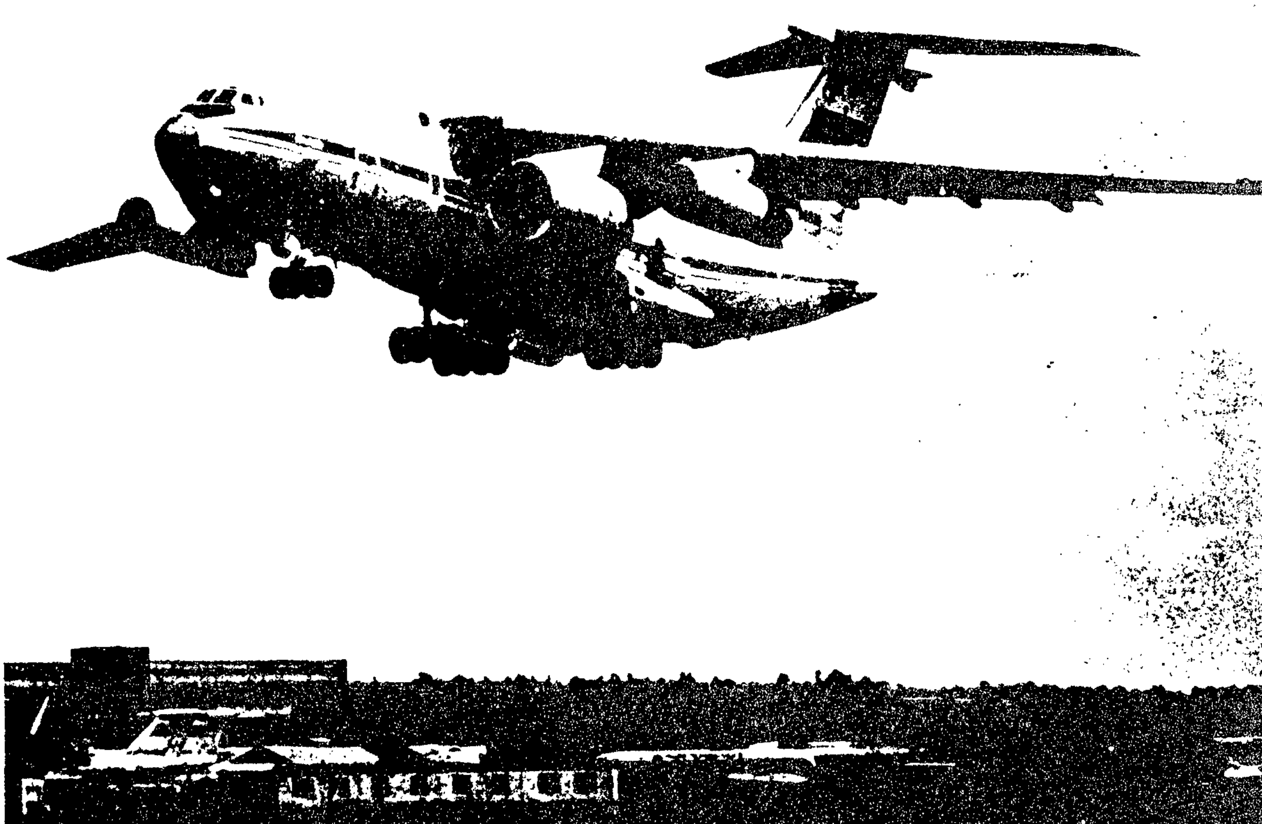


Figure 3. Arrangvent Scheme of Aircraft-in-the-Loop Flight Simulation Complex

1 - dynamometric element;
2 - transmitter;
3 - receiver;
4 - actuators;
5 - trajectory information (data) channel;
6 - telecommunication channal
7 - telemetry downlink
8 - computer;
9 - primary data processing;
10 - control command generation;
11 - radiocommand control channel;
12 - control inputs to rudder, elevator and engine;
13 - computer based real - time data analysis;
14 - graphic presentation (visualisation) of information;
15 - flihgt data recording.





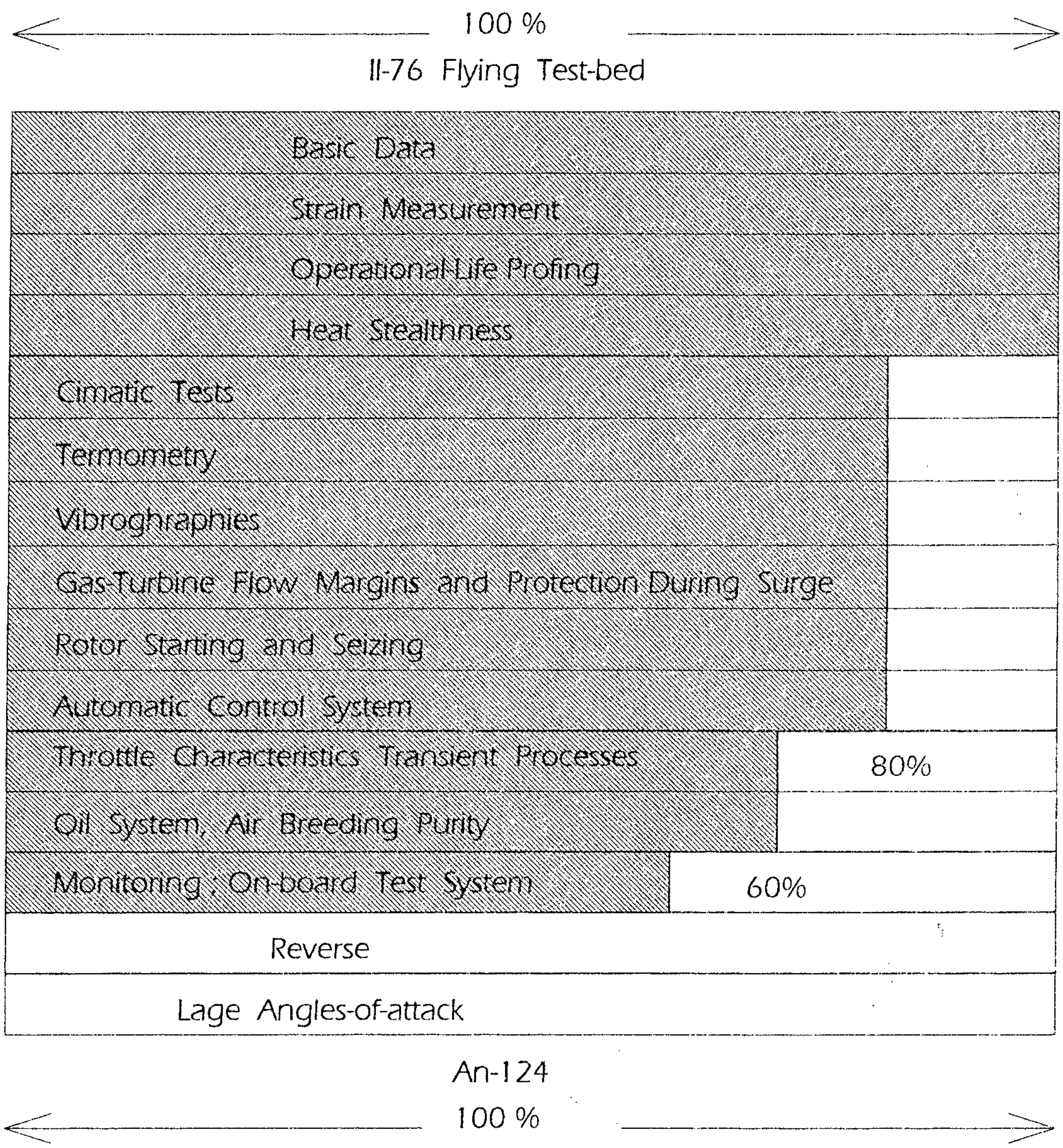


Figure 6. Type and Volume Distribution of the D-18T Flight Trials

Type of a Flying Test-bed	Flight control and navigation equipment system	a Number of flights	Prototype airplane	a number of flights	Notes
An-26	base navigation complex (BNK) -1P	106	YAK-42	40	
IL-62	base navigation complex (BNK) -2P	122	IL-86	54	
IL-62	internal syst-II	50	IL-86	~10	The rest flights were perfomed for the development of long-range navigation systems, fuels ecology
Tu-154	ground priximity warning system-2 (SOSS-2)	~40	IL-86 YAK-42	~8	The rest flights were perfomed on the flying testbed on windshear, ground proximity warning system-SPPZ (GPWS) and flight managment system-SORP (FMS) for the IL-96,Tu-204 airplanes
Tu-144	flying instrument and control system II category	~45	Tu-144	-	The second certification stage was not perfomed
Tu-104	navigation complex-144 (NK-144)	~45	Tu-144		Government joint tests were completely performed in subsonic regime on the Tu-104 flying testbed
An-26 Tu-134 Tu-154 IL-76	flight control and navigation equipment system	hundreds of flights	for aircraft, Department of Civil Aviation and Department of Defense		Efficiency is proved by introducting numerous flight control and navigation equipment systems into series production and reducing landing weathe minima up to the category III and etc.

Figure 7. A Flying Testbed for the Development and Certification of Flight Control and Navigation Equipment

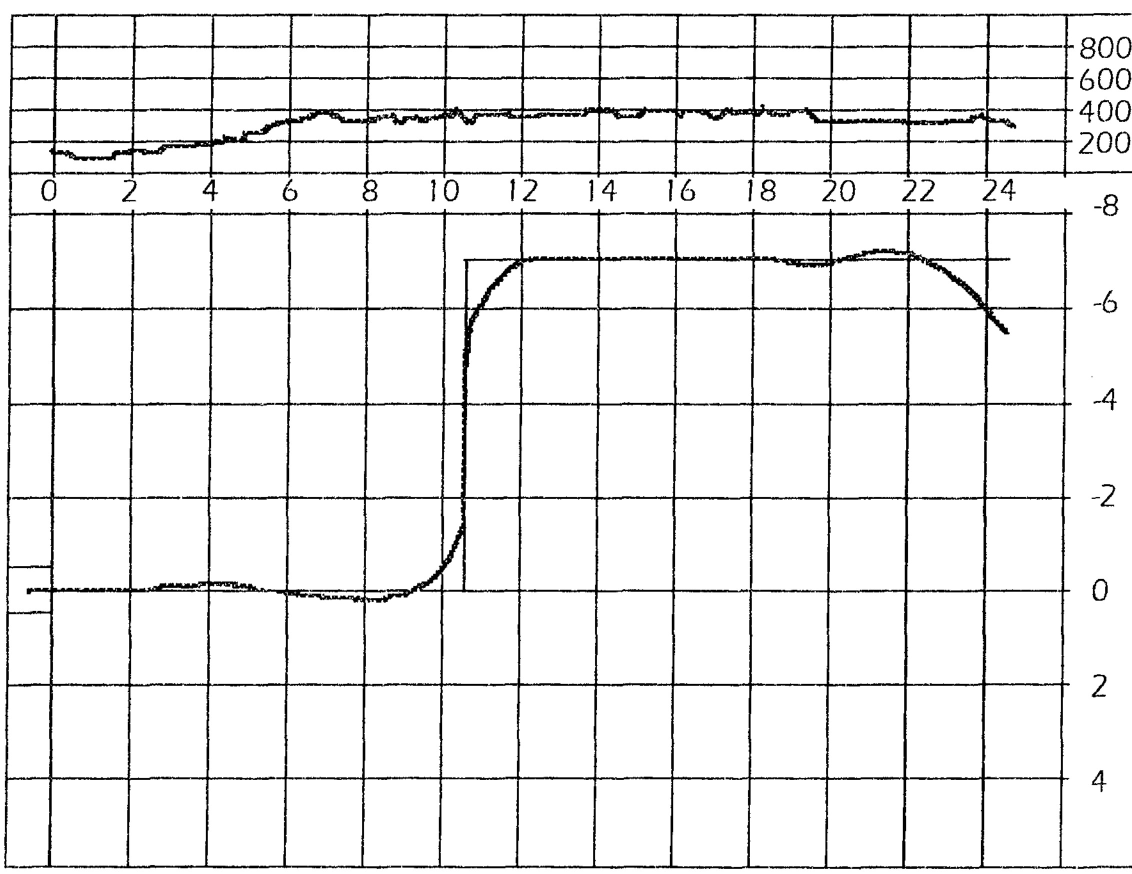
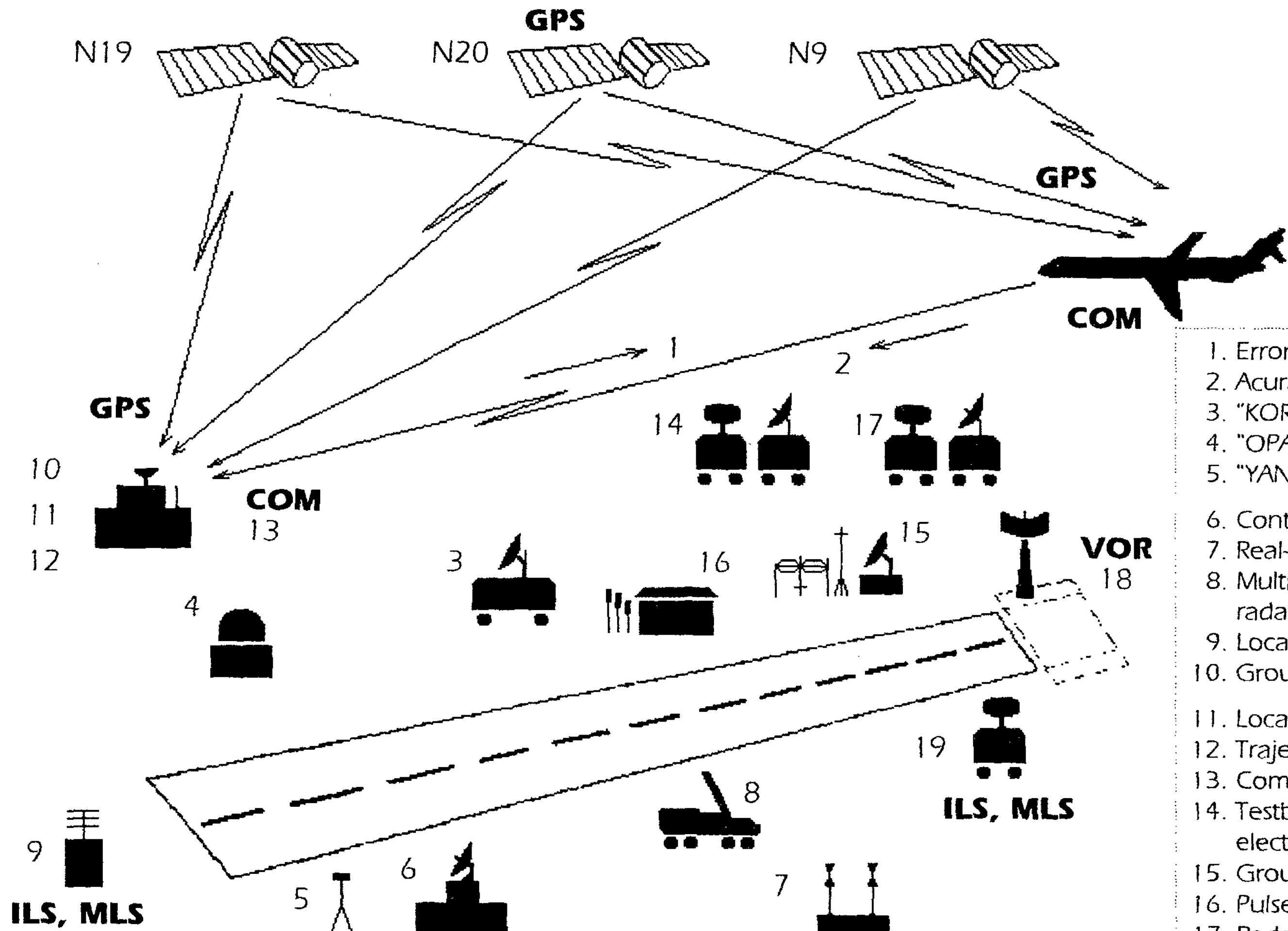


Figure 8. The example of the DGPS automatic curve-line landing approach, the flight performed on August 26, 1993.



1. Error corrections
2. Acurate coordinates
3. "KOREN" ("ROOT") and "IRTYSH" systems
4. "OPAL" system
5. "YANTAR" ("Amber") system

6. Control tower
7. Real-time flight-test control room
8. Multifunctional radar (landing radar complex)
9. Localizer
10. Ground reference station

11. Local computer center
12. Trajectory measurements
13. Comunications
14. Testbed of high-capacity electromagnetic fields
15. Ground-based communications center
16. Pulse voltage generator
17. Radoi measuring system complex
18. Ground-based lighting facilities system
19. Glideslope transmitter

Figure 9. Flight Research Institute Radiotechnical Test Range for Researches and Approval Tests of Prospective Onboard and Ground-based Equipment Complexis.

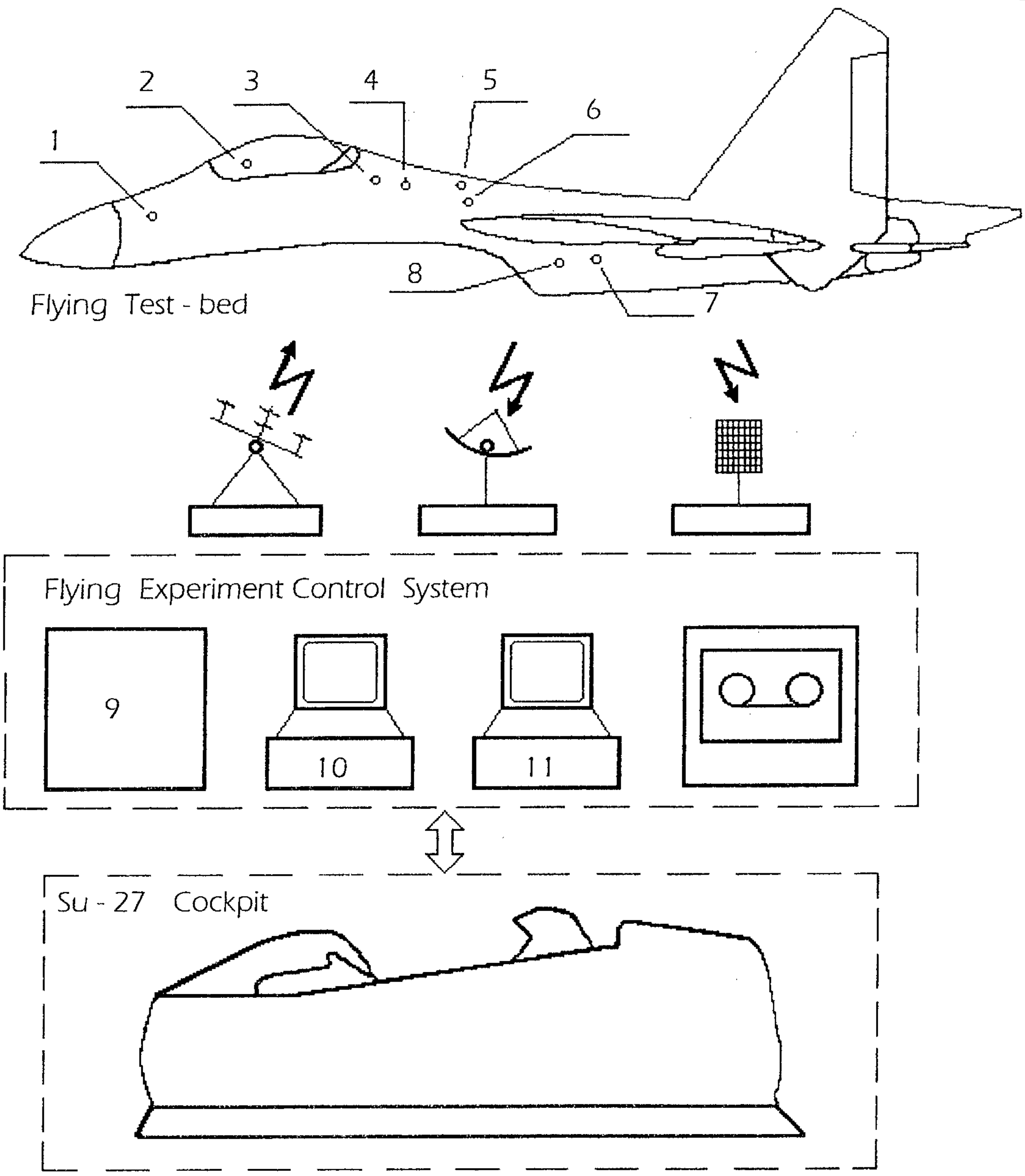
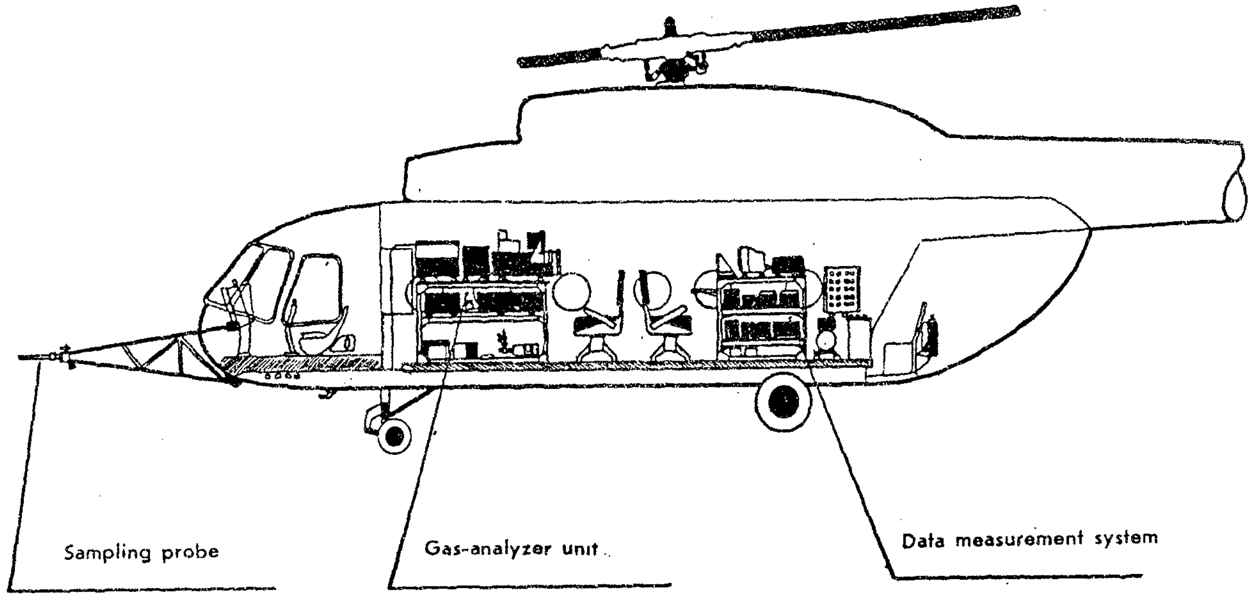


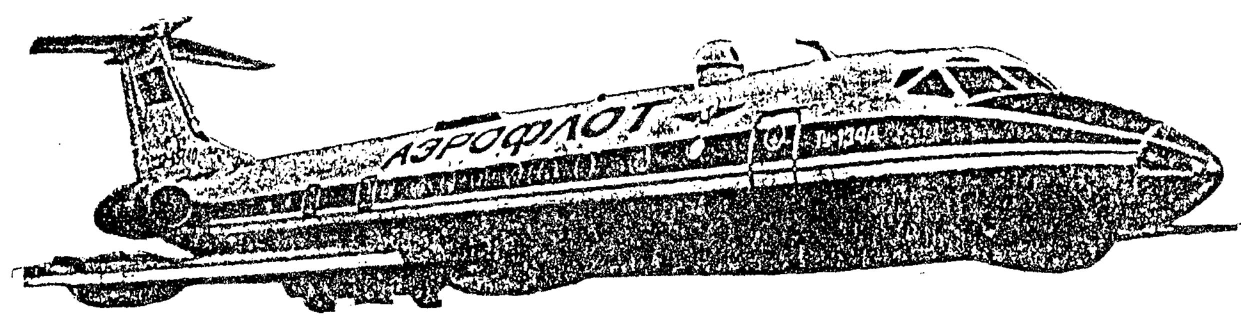
Figure 10. LMK - 2405 Complex Components

1. Television Command Link
2. Data Display System (experimental)
3. Aircraft Limitation System (experimental)
4. Automatic Control System (experimental)
5. On - board Test Instrumentation
6. Radiotelephone Station
7. Radiotelemetric Data
8. External Trajectory Measurements
9. Transmitter of Television Command Link
10. Command and Video Control Formation
11. Processing of Radiotelemetric Data and Trajectory Data, Simulation for the Ground - Test Development Stand

MI-17 LIZA FLYING TEST-BED FOR ATMOSPHERIC CONTAMINATION ECOLOGICAL INVESTIGATIONS



TU-134A FLYING TEST-BED FOR PHYSICAL AND CHEMICAL PROCESSES OF ATMOSPHERIC POLLUTION RESEARCH



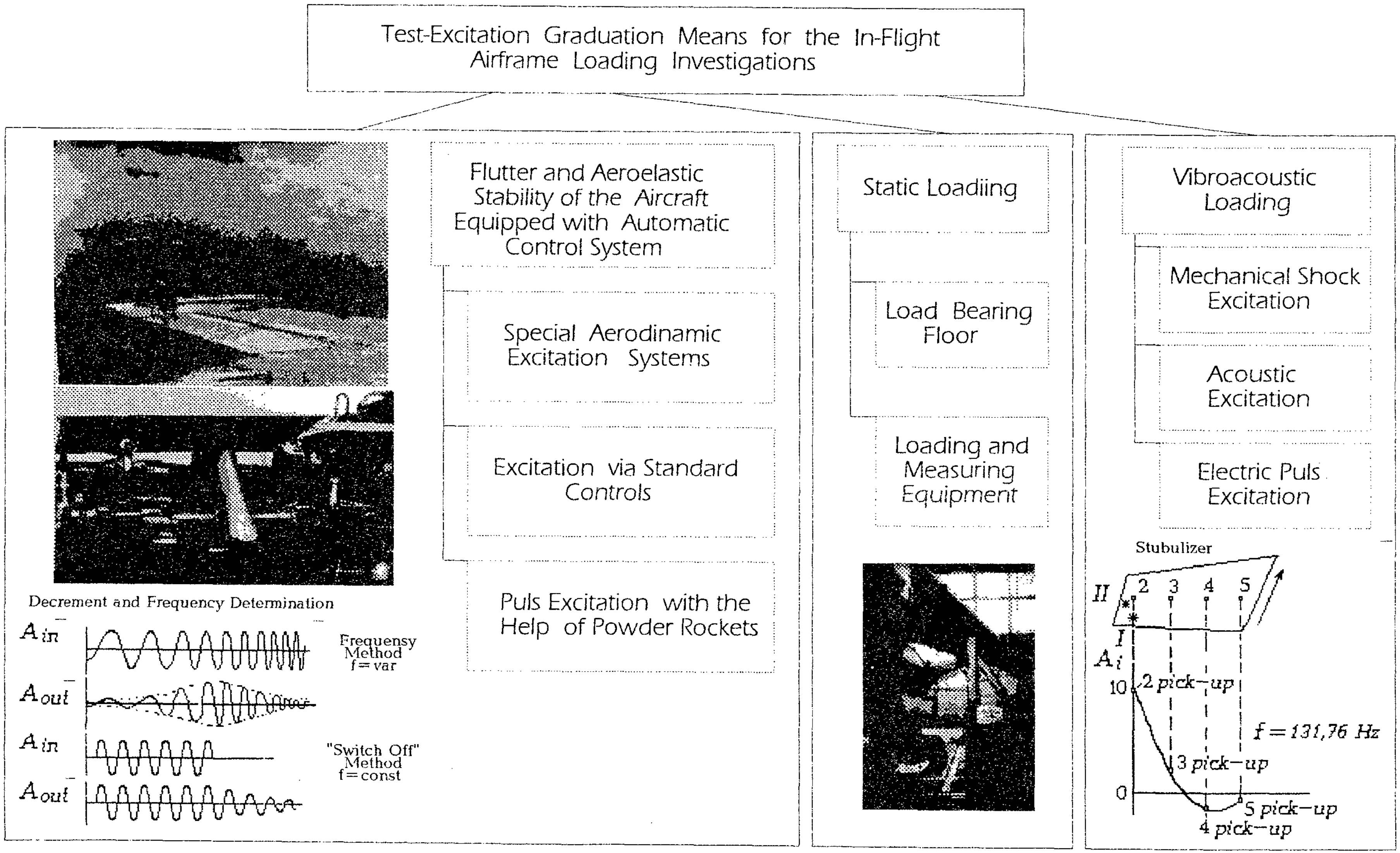
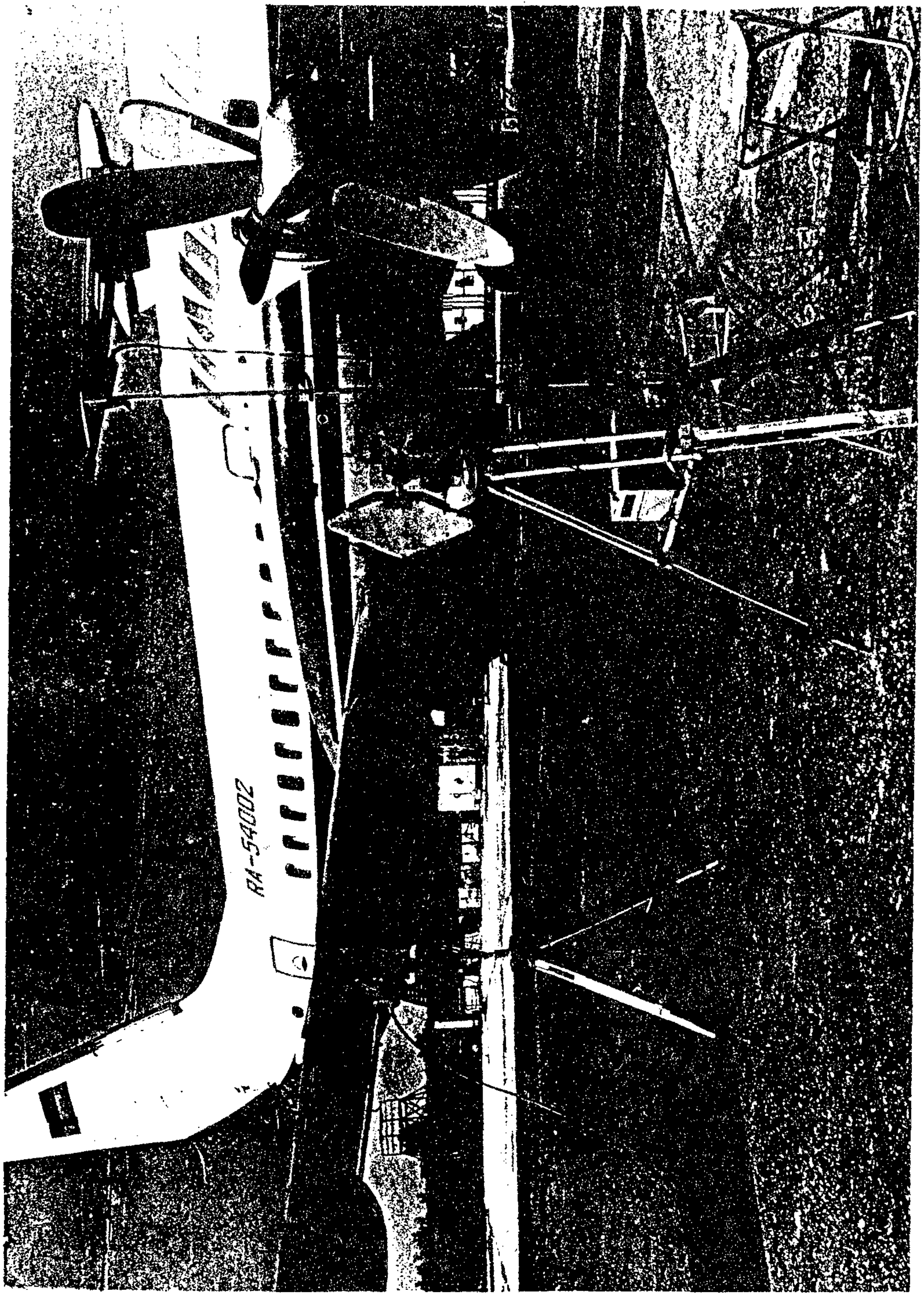
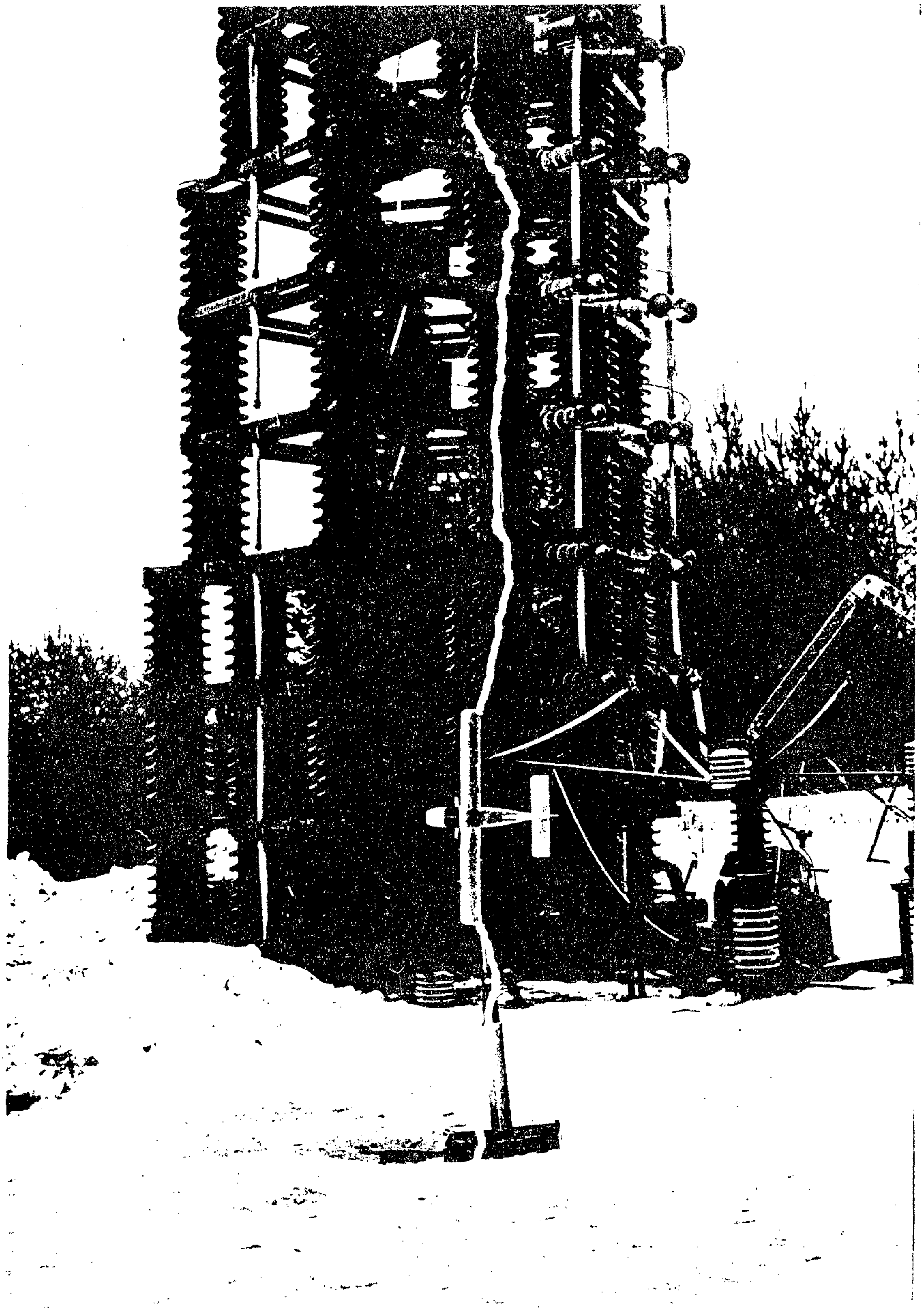

Test-Excitation Graduation Means for the In-Flight Airframe Loading Investigations
Flutter and Aeroelastic Stability of the Aircraft Equipped with Automatic Control System
Special Aerodinamic Excitation Systems
Excitation via Standard Controls
Puls Excitation with the Help of Powder Rockets
Decrement and Frequency Determination
A in
A out
Frequensy Method f=var
"Switch Off" Method f=const
Static Loadiing
Load Bearing Floor
Loading and Measuring Equipment
Vibroacoustic Loading
Mechanical Shock Excitation
Acoustic Excitation
Electric Puls Excitation
Stubulizer
II
I
2 3 4 5
A i
10
0
2 pick-up
3 pick-up
4 pick-up
5 pick-up
f = 131,76 Hz

Figure 12.


RA-54002



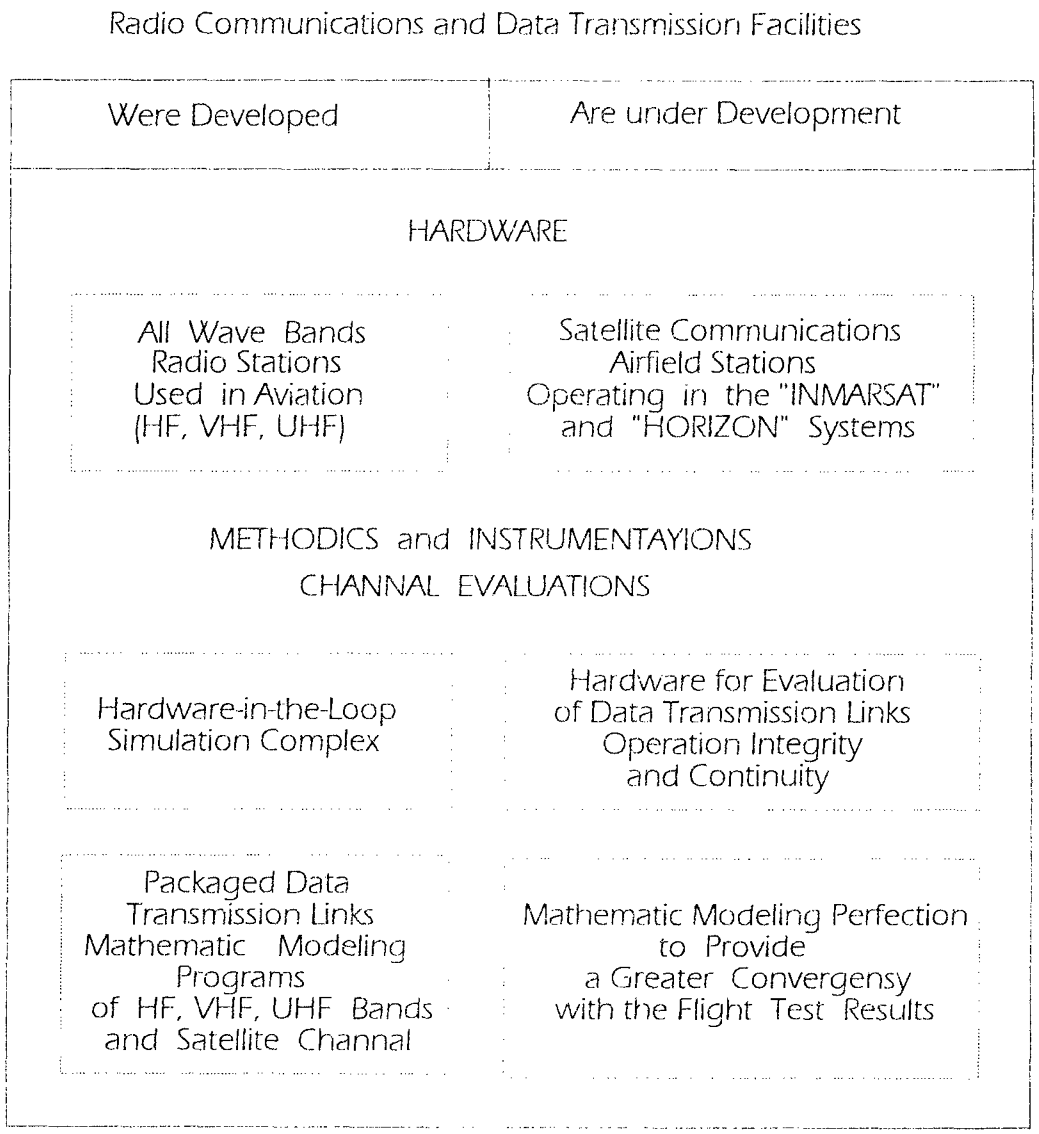


Figure 15. Structure Chart of Tests of Data Exchange Radio Links

DESIGN, ANALYSIS, AND TESTING OF DURABLE AIRCRAFT STRUCTURES

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Douglas Aircraft Company, USA

ABSTRACT

This paper examines the varied aspects of designing, analyzing and testing of modern durable aircraft structure. Subjects are coordinated around the development of structure for a commercial aircraft and include the determination of design parameters, identification of certification requirements, and subsequent analytical and experimental verification of the design and certification requirements.

Structural design issues discussed include the design for ultimate strength, the design for durability/damage tolerance, and the design for the prevention of environmental degradation. Specific examples are included from the MD-11 and MD-90 aircraft certification program including the testing done to verify compliance with structural performance and certification requirements.

The development of the Structural Design Service Goals, in terms of flight hours and landings, is discussed along with the means by which the manufacturer monitors the fleet. Extensions of Design Service Goals are discussed with the lessons learned from the recent world wide aging fleet activities.

Finally, the maintenance program for continued airworthiness of the structure is described along with how it relates to the certification requirements and development testing.

INTRODUCTION

The design of new commercial aircraft is a complex and interdependent task. Balanced on the one end by performance and on the other end by the amount of time required to perfect the design, the manufacturer is often faced with making decisions that are crucial to the program's outcome. One significant issue in the design process is the development and verification of the desired performance level of the structure. The design requirements that determine structural performance are based on regulatory, customer, and Douglas requirements. Whereas the regulatory requirements generally deal with issues of airworthiness, the customer requirements are generally associated with economic issues. This paper examines the varied aspects of designing, analyzing and testing modern durable aircraft structure to meet both regulatory and customer requirements.

THE REGULATORY REQUIREMENT

The regulatory requirement for the design of aircraft structure in the United States is the Federal Aviation Administration's (FAA) Federal Aviation Regulations (FAR). Specifically those regulations exist in Part 25 of the FARs. The FARs were codified in 1965 from the older CAR 4b requirements. Amendments to the FAR were promulgated for a variety of reasons including new technology, previously undefined safety issues and on occasion to correct errors or improve interpretations. Since February 1, 1965, FAR Part 25 has been amended 85 times. Similar regulatory requirements are published in many different countries such as the Joint Aviation Authority's (JAA) Joint Aviation Regulations (JAR) for the European Community and the Aviation Register's NLGS-3/AP-25 for the Commonwealth of Independent States. Differences in the regulations between the various certifying States and the way they are interpreted can cause significant slowing of the certification process with little increase in overall safety.

Establishing The Certification Basis. A manufacturer who desires to certify a new aircraft within the USA begins the process by letter application to the FAA signifying his intent. For new designs, the date on the application generally determines the application date and the particular amendment of the FAR under which the Aircraft will be certified. For derivative designs, earlier versions of the FAR are generally used for 'common structure' with the latest version used for the newer structure.

<u>New Designs</u>. As the name implies, new designs are aircraft that are revolutionary in design concept rather than evolutionary. For structure, this means that advanced metallic or composite materials are used in designs that are unique and innovative. The design configuration is such that extensive development, component, and full scale testing is required to verify the structural performance design parameters before it can be certified. In today's state of design, the consideration of a 'New Design' goes beyond the consideration of just the structure to the design and possible interaction of sophisticated avionics, hydro-mechanical and mechanical systems.

<u>Derivative Designs</u>. Derivative designs are designs that contain modifications to aircraft previously certified under a given regulatory

requirement. The design modifications may include one or more of the following changes:

1. Increased gross weight.
2. Fuselage lengthening / shortening.
3. Wing tip extensions / winglets.
4. Engine changes.
5. Avionics/mechanical system modifications.

In general, the areas affected by the design modifications are subject to the latest Amendment of the FARs, whereas previously certified structure is subject to the FAR Amendment under which the aircraft was originally certified. While this may seem inconsistent, it expresses that in many respects the older structure has been certified to the latest amendment through a combination of successful service experience and required changes in the design that have been evidenced as airworthiness issues during the service experience. In the United States, it is against the law to issue a certificate of airworthiness to a production aircraft in which there is a known airworthiness issue (AD).

<u>Certification Basis</u>. At Douglas, the certification basis of the aircraft is initially embodied in a document called the 'FAA Master Type Certification Plan' (see fig. 1). This document has no regulatory status in and of itself but represents an agreement between Douglas and the certifying agency on what needs to be accomplished to certify the aircraft. At the beginning of the certification process this document contains a Douglas proposed certification basis presented to the FAA for review and comment. Changes are made as appropriate until a final position is determined. At this point, the certification basis is agreed to by both Douglas and the FAA and the FAA officially acknowledges the basis by the publication of an Issue Paper. The certification basis is further amplified in a Certification Basis Document. The Certification Basis Document normally consists of the FAR Amendment(s), a number of issue papers requiring resolution and any special conditions to which the Aircraft will be certified.

While the intent of the FARs are clear, issue papers and special conditions are perhaps not so clear. An Issue Paper provides a means for the identification and resolution of significant technical, regulatory, and administrative issues that occur during the certification process. Issue papers are primarily intended to provide an overview of significant issues, a means of determining the status of issues, and a post-certification summary statement on how issues were resolved. Under the provisions of FAR § 21.16, a Special Condition is issued only if the existing applicable airworthiness standards do not contain adequate or appropriate safety standards for an aircraft because of novel or unusual design features of the product to be type certificated. Issue Papers and Special Conditions may also be issued for a variety of other reasons including special requirements to address chronic in-service problems. The complete FAA Certification process used for the MD-90 program is shown in fig. 2.

CUSTOMER REQUIREMENTS

Ideally, the customer's needs are expressed in terms of economics rather than safety. With safety as the common denominator, the customer desires an aircraft that is less costly to operate and maintain than the aircraft that is being replaced. Consequently, the operators are participants in the design process to insure that their basic concerns are accounted for in the design. Some of the more significant issues that customers are frequently interested are shown in fig. 3.

MODERN AIRCRAFT DESIGN PRACTICES

The Douglas Aircraft Company, a component of McDonnell Douglas Corporation, has established design policies and goals which ensure that Douglas products have long lives and predictable aging characteristics. The design of a new aircraft starts with the definition of all of the design conditions and design goals for the structure. These conditions not only specify the extreme structural design requirements for the aircraft, but also the length of time (years, flight hours, and landings) the aircraft should be in operation and not suffer from degradation due to fatigue or corrosion, The ultimate strength design conditions for a modern day transport represent extreme loading conditions, which occur infrequently in the life of an aircraft. The design conditions for fatigue/damage tolerance and corrosion are the routine loading and environmental conditions that an aircraft sees on a day-to-day basis. In fig. 4, some examples of the extreme conditions include encounters with exceptionally high levels of turbulence in a thunderstorm, abrupt maneuvers to avoid collision, landings at descent rates far above normal, and a dive pullout following an inadvertent upset. On a statistical basis, these extreme events will occur less than once in the life of an aircraft. In the last decade the probability of such encounters has decreased due to refinements in the ability to predict adverse weather conditions and the use of predetermined and controlled route structures. Normal loading conditions, experienced by each aircraft, include all variations in ground and flight loads. These include loads due to routine taxi, takeoff, climb, cruise, descent, and landing. A representative internal load time history that results from normal operation is illustrated in fig. 5.

The Design for Ultimate Strength. The extreme design conditions are dictated by regulatory authorities throughout the world. The authorities publish regulations similar to the Federal Aviation

Regulations (FAR), In certain instances, Douglas not only adheres to the authority's recommendations but also goes beyond the requirements in order to meet established goals.

The regulations identify two different categories of Structures. The first is called damage tolerant structure (also referred to as fail-safe structure) and the second is called safe-life structure. Damage tolerant structure must be tolerant of damage and still able to carry the limit loads after occurrence of damage. Such structure is designed so that major components such as spar caps, stringers, and skin can be partially or completely failed, but the aircraft can continue to fly safely until the condition is discovered and corrected during the next maintenance check. This includes tolerance to corrosion, accidental damage and fatigue. Almost all structure of the airframe can be classified as damage tolerant.

Safe-life structure refers to components that are not damage tolerant. These items are generally single load path structure and involve the use of extremely high strength materials. As such, safe-life structure will fail due to limit load application before a flaw can reasonably be detected by nondestructive means. Therefore, in order to avoid single load path failure and ensure the structural integrity of the aircraft, the structure is periodically replaced. A typical example of safe-life structure is the landing gear. Examples of both types of structures are shown in fig. 6.

The actual design of the primary load carrying aircraft structure is balanced between:

1. The amount of material required to prevent structural failure from one of the extreme loading conditions described above (ultimate strength),
2. The amount of material required to sustain structural integrity with certain levels of hidden or undiscovered damage (damage tolerant or fail-safe), and
3. The amount of material required to delay the onset of wide spread cracking until the aircraft has reached its useful life (fatigue resistant).

This is done through a process of selecting design limit stress levels, materials and design features which accomplish all three requisite conditions.

The Design for Long Life. Douglas' design goals are initially set to an equivalent economic life of 20 years. In the past this has been established on a financial rather than a technical basis because the customer expects an amortization period of at least 20 years. In order to meet this requirement, Douglas designs and builds products for a minimum average useful life of 40 years. This provides the customer with a high statistical probability that the aircraft will be relatively free from fatigue or corrosion problems for 20 years providing the routine maintenance tasks are performed.

Damage tolerance concerns cause one to consider the potential crack growth of a single isolated crack in the design of any feature of the aircraft. A single small crack in itself is not critical to the integrity of damage tolerant structure. It is the subsequent growth of the crack and loss of residual strength that is of concern (fig. 7). The cyclic nature of the loading an aircraft experiences can cause the crack to grow and eventually weaken the local structure to a point at which it cannot withstand the extreme design conditions (fig. 7). It is this condition that must be prevented. Adequate prevention is provided through careful detail design, encompassing both crack growth characteristics and crack detectability (inspection). This is also backed by a comprehensive test program and, ultimately, routine maintenance checks and special inspection programs performed by the operators.

The process of finding and repairing cracks and corrosion in older aircraft could continue indefinitely. However, as shown in fig. 8, economic factors should serve to retire the aircraft. Such factors include:

1. competition with newer technology aircraft whose operating costs are significantly less,
2. increased repair, and maintenance costs associated with aircraft experiencing widespread cracking and
3. regulation changes which retroactively affect the certification basis of the aircraft (e.g., FAR Part 36, Noise rules).

As shown in fig. 9, Douglas' design goals differ markedly depending on the structure being designed. Damage tolerant structure for commercial aircraft is designed to be free from fatigue cracking for the first life and to have no major fatigue crack problems throughout its second life. Safe-life structure is designed to be crack free for a minimum of three lifetimes of the projected usage. These design goals are verified through analysis and a series of development, component, and full-scale fatigue tests. The MD-90 testing programs include an increased requirement for testing to three lifetimes on all structure. This requirement is Douglas imposed.

The Design For Corrosion Prevention. Undetected and unrepaired corrosion damage will ultimately cause the loss of structural integrity. In many areas of an aircraft the loss of integrity is preceded by significant reductions in fatigue life and residual strength (fig. 10). If the corrosion is widespread, the loss in integrity could be manifested in major structural component loss under normal flight loads. While this is an extreme example, there have been recorded instances of in-flight break up due to corrosion (Taiwan Air 1975 accident).

Historically, aircraft were not designed to tolerate specific levels of corrosion. The primary reason for this is based on economics. An aircraft operated in the tropics may require significantly more corrosion protection than one operated in more temperate regions.

Corrosion prevention systems add significantly to the weight of the aircraft and therefore the operator operating in the more temperate region was not willing to incur the extra weight penalty for his aircraft. Therefore operator maintenance programs have been relied upon to find and correct corrosion before it could significantly erode structural safety margins.

In today's world of the mega-airline and mega-leasing companies, this may not be true anymore. While the premise of not designing the aircraft to tolerate specific levels of corrosion remains, airlines are now demanding aircraft to be highly resistant to all forms of corrosion. Indeed, in the design of the DC-10 twenty five years ago and more recently our MD-80, MD-90 and MD-11 aircraft, our operators realized the importance of the manufacturer in providing a corrosion resistant aircraft. In providing designs that meet the operator requirements, the manufacturer must consider the structural environmental threats shown in fig. 11.

Douglas has the design goal to produce an aircraft that will tolerate, at the minimum, one design lifetime of usage without the evidence of significant structural corrosion. With this approach, Douglas has opportunity in three specific areas to achieve this design goal:

- Structural Material Selection,
- Design Features,
- Supplemental Materials and Processes.

Structural Material Selection. The selection of materials that are naturally resistant to corrosion is a significant step towards structure that is corrosion resistant. This includes the judicious selection of materials that have lower anodic potentials and avoidance of materials such as magnesium. This also includes the selection of materials where dissimilar materials may come into contact. Materials that have lower and similar anodic potentials tend to be less susceptible to corrosion. The selection of heat treatments is also an important consideration for stress corrosion and exfoliation. Use of overaged heat treatments tend to reduce susceptibility to stress corrosion and exfoliation. These thoughts are summarized in fig. 12.

Design Features. Design features are employed as one way of preventing moisture retention within the structure. These design features used on Douglas products are summarized in fig. 13 and are generally aimed at providing positive drainage paths to bilge areas where the moisture can be drained.

Supplemental Materials and Processes. These materials and processes form the primary shield in protecting the aircraft structure from corrosion. Their primary purpose is to form sacrificial and impervious layers to prevent moisture from ever reaching the metallic surfaces. Again, where moisture and corrosive materials (e.g. battery acid, lavatory wastes, etc.) can be excluded from the metal surfaces, corrosion can be prevented. A summary of the various surface treatments used in Douglas products is shown in fig. 14.

In-service Experience. The DC-10 was the first commercial aircraft designed with the corrosion prevention system described above. In the design process, our operators demanded that the aircraft have a level of corrosion protection that would set the standard for future aircraft. We listened to our customers and began the design process by reviewing our experiences with the DC-8 and DC-9. What evolved from that review was a corrosion prevention system recognized by International Air Transport Association (IATA) for a superior approach to corrosion prevention. In fact IATA adopted many of the design features into their recommended specification for aircraft corrosion prevention. The design features employed for the DC-10/MD-11 fuselage are shown in fig. 15.

After a total of 25 years of service the DC-10 has yet to develop significant in-service corrosion problems. Even when the aircraft received little or no attention over a period of time, the corrosion prevention systems held up remarkably well. During its production life, Douglas changed the build standard only twenty times to accommodate in-service problems due to corrosion while other manufactures have required literally hundreds of changes to control in-service corrosion problems. As a testimony to the corrosion prevention system used on the DC-10, one operator has successfully extended the first major corrosion check on his DC-10 fleet to 48,000 flight hours. This represents over two and half times the original FAA Maintenance Review Board (FAA MRB) proposals (18,000 hours). The DC-10 was designed for 60,000 flight hours and currently some aircraft have over 90,000 flight hours. The design standards for corrosion prevention developed for the DC-10 are part of the standards for current Douglas products including the MD-80, MD-90, and MD-11.

DESIGN VERIFICATION

Development Testing. In the development of any new aircraft the test program becomes the focal point to evaluate the performance of the design concept, from the small details of the design to the full scale article. Tests are performed both to satisfy regulatory requirements (fig. 16 shows the DC-10 in Proof Test and fig. 17 shows the MD-90) and to ensure the design performance goals of the structure are met. Testing begins early in the design process on structural details of specific portions of the aircraft.

Various design concepts are evaluated for structural strength, fatigue life, and damage detectability. The purpose of the evaluation process is to identify design concepts that meet the design performance goals. As an example, the final designs for fuselage longitudinal skin splices for the DC-8, DC-9 (MD-80, MD-90), and DC-10 (MD-11) are shown in fig. 18. Over 50 different configurations were evaluated for each model. The most promising concepts were integrated into full size panels (fig. 19), which were subject to repetitive loads simulating the full scale aircraft. If premature failures are experienced, the design is modified and re-tested. As shown in fig. 20, full scale portions of the airplane are placed in test fixtures and fatigue tested, using cycle-by-cycle and flight-by-flight loading, for at least two projected lifetimes; three for safe-life structure. If premature failures are encountered during this phase of testing, the design is modified and service bulletins are issued to correct deficiencies in all delivered aircraft. It is the Douglas policy to have completed at least one lifetime of fatigue testing before the aircraft enters commercial service.

Certification Testing. From the smaller component tests to flight testing, the FAA requires a whole suite of tests including conformity inspections to insure that the aircraft structure meets the basic airworthiness requirements. Some development testing is performed as a means of satisfying regulatory requirements. Tests such as fastener allowables, compression allowables etc. are normally configuration sensitive and need to be developed as part of the certification process. In current aircraft certifications, tests are required to demonstrate fatigue endurance and damage tolerance characteristics. Tests are also required to demonstrate structural capability under limit loads. Tests to ultimate load are required for structure that is innovative or use materials that have never been fully characterized.

Derivative designs can and do make use of tests performed on earlier models. Ultimate tests and large scale fatigue tests are normally only performed on the first model of aircraft produced. Ground vibration testing and flight tests are normally required to demonstrate freedom from flutter for the dynamic modes that have modified characteristics. A summary of the major structural tests performed in the MD-90 certification program is shown in fig. 21.

THE MAINTENANCE PROGRAM FOR STRUCTURE

When an aircraft enters service it is the responsibility of the operator to perform routine maintenance checks in order to detect and correct damage or deficiencies in a timely manner. The requirements of most regulators stipulate that an acceptable maintenance program must be in place before the airline initiates service. This process is illustrated in fig. 22. Before certification, Douglas and its operators propose a maintenance program to the FAA Maintenance Review Board (FAA MRB). The FAA reviews, modifies and approves the program based on supporting data submitted by all parties. Once approved, the operators have a basis for establishing a FAA approved maintenance program based on the MRB report. Individual maintenance programs developed at the airline from the MRB document are then approved by the Principal Maintenance Inspector (PMI) who surveys the airline. When sufficient data exists to warrant a review (e.g. expansion of time limits) the airlines can petition the FAA MRB to modify the maintenance program.

Currently jet aircraft maintenance planning documents are developed under an Airline Transport Association (ATA) document titled 'Maintenance Steering Group-3' (MSG-3). The guidelines provided in MSG-3, for program development, deal with both systems and structure. Maintenance procedures for the structure are initially directed towards corrosion and accidental damage. Later, a separate mandatory inspection is introduced to deal with fatigue related damage. MSG-3 was developed, in a joint effort between the ATA, Aerospace Industries Association (AIA), International Airline Transport Association (IATA), Association Europeenne Des Constructeurs De Materiel Aerospatial (AECMA), and the FAA, as an acceptable means of addressing the requirements of FAR 25.571 and FAR 25.1529, Amendment 54.

A revised version (MSG-3R2) has been issued, which eliminates the ambiguities of the first document and incorporates many of the aging aircraft initiatives discussed later in this document. The MD-11 and the MD-90 have complied with the full intent of this document in developing maintenance programs. The MD-80 and DC-10 maintenance programs are currently being revised to comply with MSG-3R2. This is expected to result in an extensive operator cost savings in the maintenance program activities for both aircraft.

Once the new aircraft model is delivered, Douglas continually monitors the performance of the airframe. If unanticipated cracks or corrosion develop, Douglas will evaluate the cause, issue recommended repairs, and appropriate modified maintenance instructions. Douglas also modifies production aircraft in order that the newer aircraft will be protected against the event.

DOUGLAS PRODUCT LINE

Douglas has produced over 3200 commercial jet aircraft since 1958. There have been three distinct model lines and six different model types spanning the nearly 40 years of production. A summary of the DAC Commercial product line is shown in fig. 23.

DC-8. The DC-8 was the first Douglas jet to be introduced in 1959. Of the 556 originally produced, over 300 continue in use today.

All of those in-service have exceeded the original 20 year design life goal. Operation of these aircraft is expected well into the 21st century.

DC-9. The DC-9 was introduced into revenue service in 1965. The initial 70 passenger aircraft was quickly grown to seat up to 130 passengers. A total of 976 of these aircraft were delivered in five different models. Currently over 880 are still in service. Some of these remarkable aircraft are successfully being used into their third lifetime with little or no signs of structural degradation. This aircraft has been type certified by the Aviation Register.

DC-10. The DC-10 was introduced in 1971 and was Douglas' first wide bodied aircraft designed to hold upwards of 280 passengers in three classes. The aircraft quickly became a success in both domestic and international routes. Of the 446 aircraft produced, 419 are still in service around the world. This aircraft is the newest type of aircraft to be operated in the CIS. This Aircraft has also been type certified with the Aviation Register.

MD-80. The MD-80 is the first major derivative model of the DC-9. First delivered to operators in 1980, the MD-80 has been the all time best seller of any Douglas commercial product. Still in active production, over 1,120 have been produced with a total of 1,111 still in service around the world. Demand for this aircraft is expected to keep the production line open well into the next century.

MD-11. The MD-11 is the first major derivative model of the DC-10 aircraft. Introduced in 1990, 131 have been produced and are active. Customer acceptance of this technologically state of the art aircraft has been high. Production of this aircraft is expected well into the 21st century.

MD-90. The latest aircraft added to the Douglas commercial line is the MD-90. This is also derivative of the DC-9 aircraft with engines that have been designed to be quiet and environmentally friendly. The MD-90 entered revenue service in April of 1995.

MD-95. The first aircraft to meet the challenge of a global manufacturing community is yet another DC-9 derivative. Aimed at augmenting the 100 passenger market, this DC-9-30 sized aircraft is being actively offered to airlines around the world. Authority to proceed with design and production of this aircraft will come when launch orders are booked.

AGING AIRCRAFT LESSONS LEARNED

In April 1988, worldwide attention was focused on the subject of Aging Aircraft when an airplane's roof blew off. Evaluations of the accident revealed that the major cause was directly attributed to age-related structural degradation that was not found in a timely manner.

As a result of the Conference on Aging Aircraft held in Washington DC in June 1988, 21 aging aircraft issues were identified for resolution (fig. 24). The Airline Transport Association (ATA) sponsored the development of three task groups — one each for Boeing, Douglas, and other manufacturers — composed of industry representatives, operators, and regulators to address the issues identified at the conference. The ATA appointed a blue ribbon Steering Committee (now known as the Airworthiness Assurance Working Group (AAWG)) to direct the work of the task groups. The primary tasking was to examine on a broad basis the situation that allowed the 1988 accident to occur and then to propose paradigm changes to the way industry viewed a fleet of aging aircraft. The prime objective identified was to identify actions that would restore an aircraft to a baseline configuration where the inherent fail-safe and damage tolerance design concepts were intact. By accomplishing these actions on an aging aircraft, the aircraft would be restored to a point where the normal maintenance program could, once again, be relied upon to insure continued airworthiness. The AAWG called for major tasks to be accomplished in the areas of product evaluation and research and development.

Product evaluation. The AAWG identified the following steps to be accomplished in the Product Evaluation phase:

1. Select Service Actions for termination of special repetitive inspections in areas where a high likelihood of damage exists in combination with specific airworthiness concerns.
2. Implement an industry-wide mandated minimum corrosion control and prevention program by model.
3. Develop an assessment procedure for airline maintenance programs to determine program adequacy against known standards of excellence.
4. Develop the means to assess long-term structural repair quality of repairs on aircraft in the current fleet and schedule the removal of questionable ones.
5. Assess the Structural Supplemental Inspection Document (SSID) program findings in relationship to original program objectives and requirements and make any necessary adjustments.

6. Establish an industry common assessment process to determine a design's susceptibility to widespread fatigue damage and make model specific recommendations to prevent its occurrence in the fleet.

Items 1, 2, and 5 have been formally implemented on the Airbus A-300, Boeing 707, 727, 737, and 747, BAe 1-11, Douglas DC-8, DC-9, and DC-10, Fokker F-28, and Lockheed L-1011 by Airworthiness Directive. These items are reviewed annually to insure completeness and if necessary modified to assure continued airworthiness. Items 4 and 6 are still being developed by the task groups and Item 3 was published as an informational report by the Airline Transport Association.

The actions specified in the Product Evaluation phase represent a comprehensive audit of structural issues and a baselining of the aircraft. Accomplishing the actions identified in the product evaluation process together with normal maintenance provides continued airworthiness of the aircraft past its original design life goals (fig. 25).

The significance of the aging aircraft activities has not been lost for future generation aircraft. Specific activities have been put in place to insure that current and future certification programs will have the full lessons learned from the earlier generation aircraft. For Service Action review, the industry now participates in a lead airline review prior to finalization of the Service Action. During the review, Douglas' operators have a chance to review the Service Action prior to publication where an assessment is made as to how and when the Service Action should be terminated.

The Corrosion Prevention and Control Program (CPCP) has been embodied in the MSG-3R2 document making it a possible to address maintenance programs for corrosion prior to certification. In addition, the FAA has been considering a modification to FAR 121 (129, 135 etc.) requiring a pre-certification maintenance program for corrosion.

The Supplemental Inspection Documents have been incorporated for future aircraft by Amendment 54 to FAR Part 25. This amendment requires the production of an Airworthiness Limitation report that embodies the intent of the SID before certification.

The issue of widespread fatigue damage has been included in a proposed revision to FAR Part 25.571. This revision advocates the use of fatigue testing to determine the likely possibility of when widespread fatigue damage might occur in-service. The change to the FAR and accompanying revision to the Advisory Circular is expected later this year.

By virtue of the damage tolerance certification basis for all new aircraft, the issue of inspection programs for repairs on primary structure is a requirement. The MD-90 and MD-11 have been certified with FAA approved damage tolerance repairs in the Douglas Structural Repair Manual (SRM).

Research and Development. This task is aimed at establishing research and development needed to augment and understand the behavior and management of an aging group of aircraft. An industry team composed of manufacturers, operators, regulators, and NASA has been guiding research in areas such as advanced NDI techniques, analytical methods for widespread fatigue damage, and testing requirements for certification issues.

SUMMARY

In summary, the production of long-life, safe and reliable aircraft is a joint effort between the Operators, the Manufacturers and the Regulators as shown in fig. 26. When one or more of the elements is missing, the system becomes unstable and potentially unsafe. All elements must work diligently and work together to insure product safety and reliability. Douglas is committed to this concept since it began production of commercial aircraft 75 years ago.

Since the Douglas Commercial, First (DC-1) was introduced in 1933, Douglas established a definitive position in commercial aviation. This position was further established with the introduction in 1936 of the Douglas Commercial, Three (DC-3) forerunner of the C-47 and Soviet Li-2 (estimated 16,000 plus produced). Out of the rich heritage of the DC line, over 3200 successful commercial jet aircraft have been produced. As Douglas celebrates it's 75th year as a manufacturer our collective commitment remains focused on producing the same high quality products, known worldwide for reliability and toughness, that has always been associated with the Douglas products such as the DC-3. As Douglas looks to the 21st century and planning continues to introduce the newest member of our product line, the MD-95, Douglas will continue to be at the forefront of commercial aviation.

ACKNOWLEDGMENTS

The authors wish to thank Gary Bartz, Walter Laurence, and Jack Rowan for their invaluable assistance in the preparation of this paper.

Copy number Report Number :

MD-90 FAA MASTER TYPE CERTIFICATION PLAN

Revision date Revision letter

Issue date June 22, 1993 Contract number

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FIGURE 1. FAA MASTER TYPE CERTIFICATION PLAN

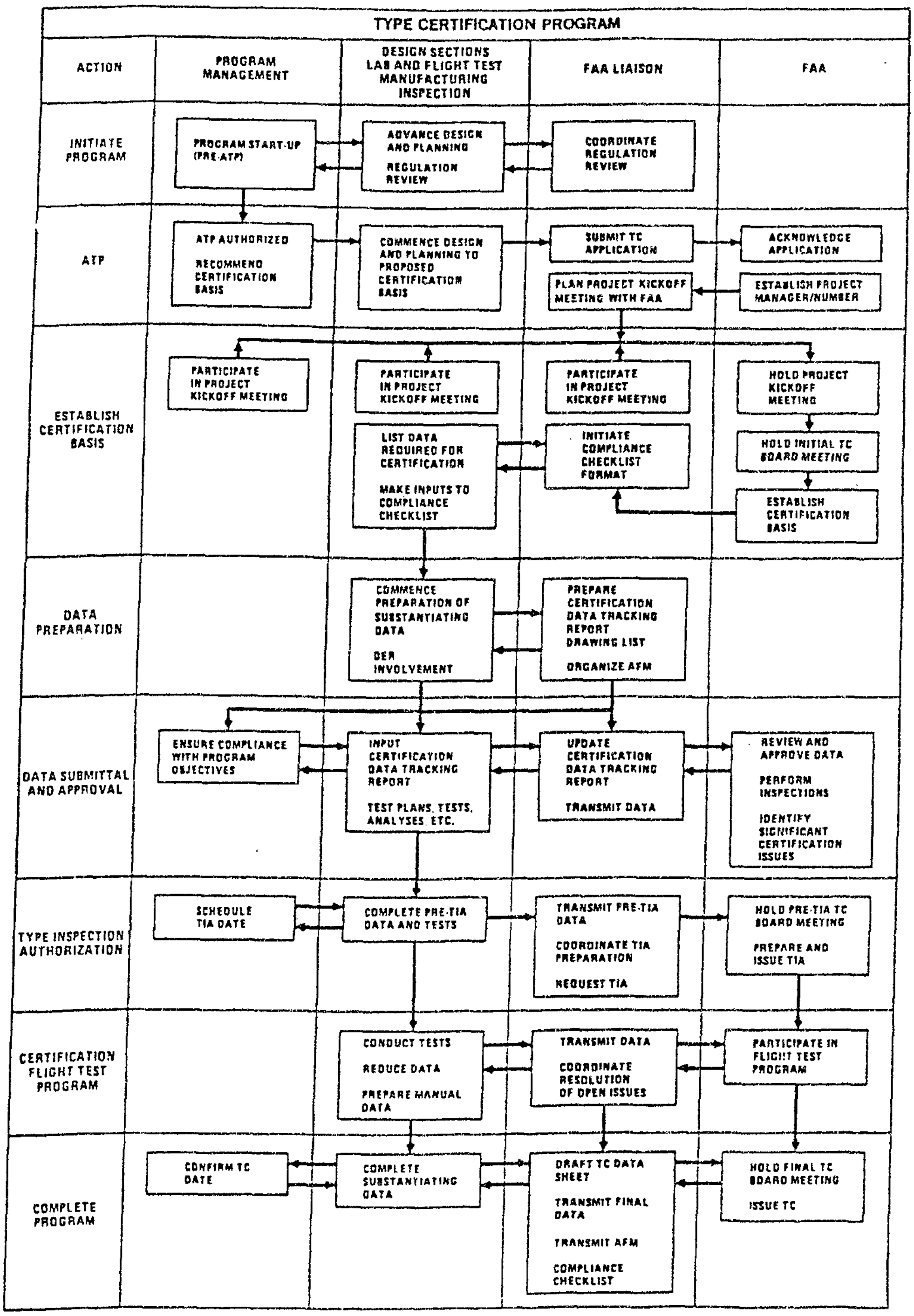

FIGURE 2. FAA MD-90 CERTIFICATION FLOW CHART

CUSTOMER REQUIREMENTS

- LIGHT WEIGHT RELIABLE STRUCTURE
- CORROSION PREVENTION AND CONTROL
- EASE OF STRUCTURAL INSPECTIONS
- INFREQUENT STRUCTURAL INSPECTIONS
- EASE OF STRUCTURAL REPAIRS

FIGURE 3. CUSTOMER REQUIREMENTS

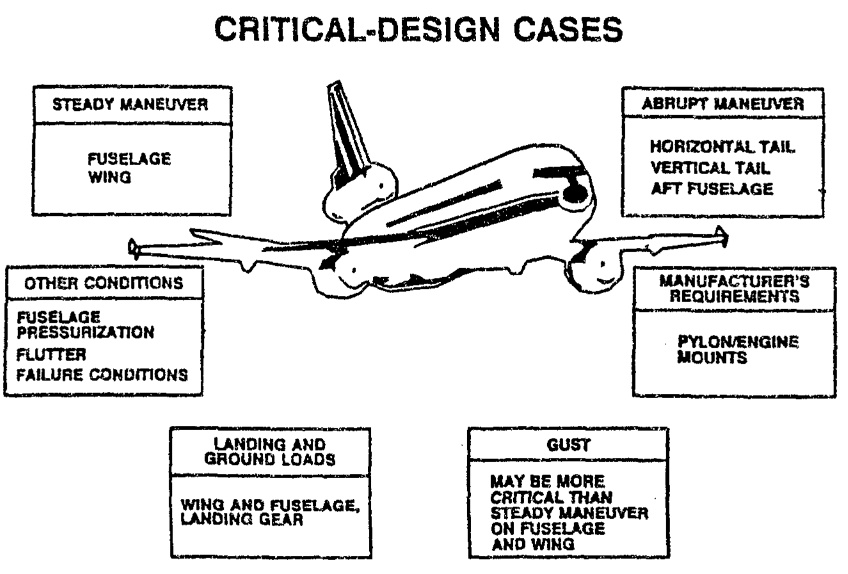

FIGURE 4. MODERN JET TRANSPORT CRITICAL-DESIGN CASES

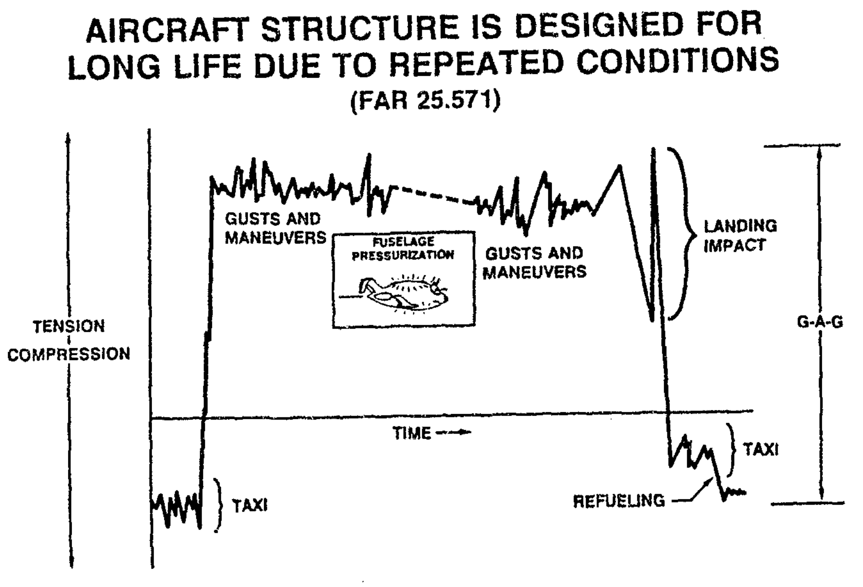

FIGURE 5. TYPICAL LOWER WING SKIN INTERNAL LOAD TIME HISTORY

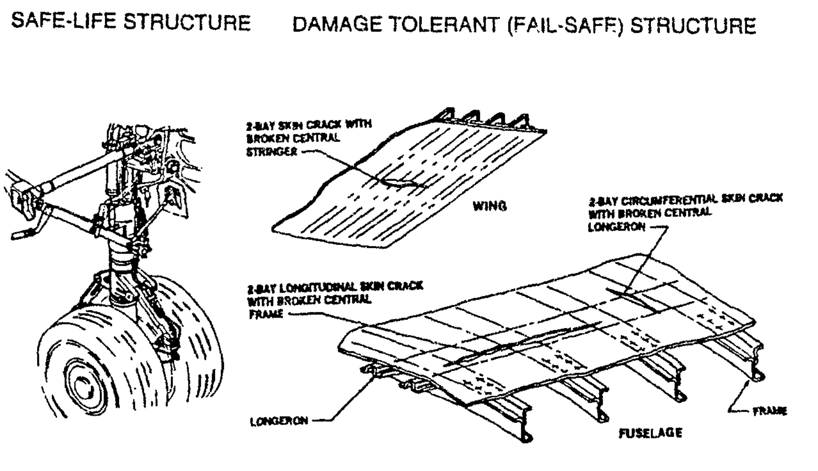

FIGURE 6. STRUCTURAL TYPE CLASSIFICATION

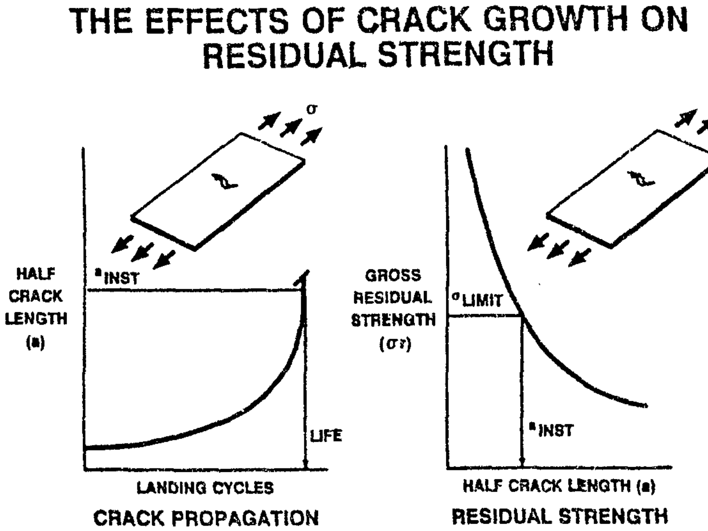

FIGURE 7. THE EFFECT OF CRACK GROWTH ON RESIDUAL STRENGTH

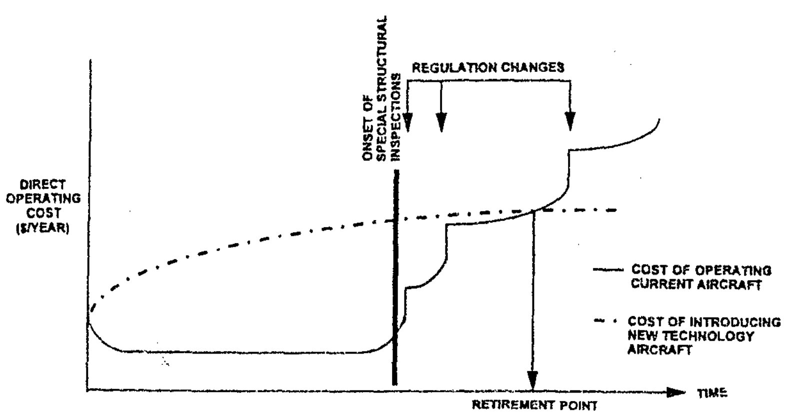

FIGURE 8. WHEN TO RETIRE AIRCRAFT

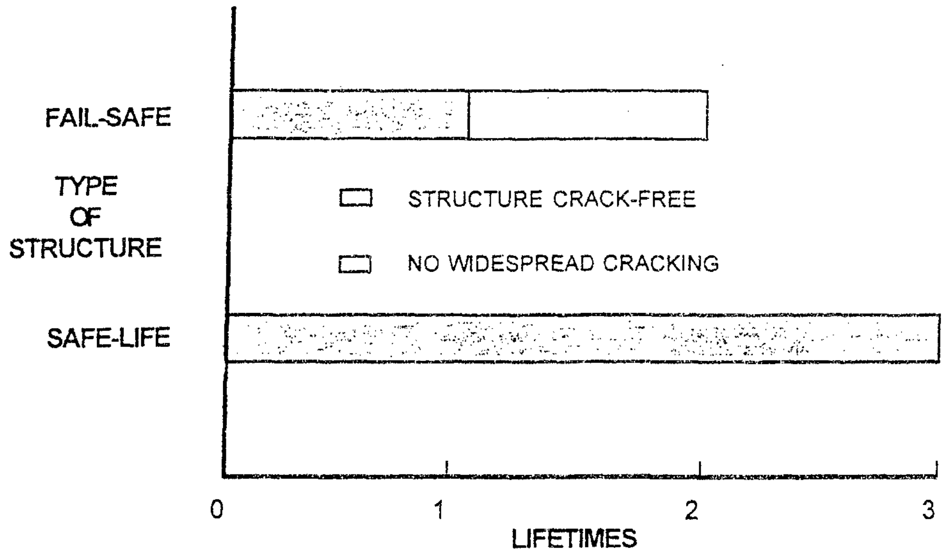

FIGURE 9. DOUGLAS DESIGN LIFETIME PHILOSOPHY FOR STRUCTURE

FIGURE 10. CORROSION DAMAGE DUE TO INADEQUATE MAINTENANCE

SIGNIFICANT FORMS OF AIRCRAFT CORROSION

- CORROSION BETWEEN SIMILAR METALS
- CORRSION BETWEEN DISSIMILAR METALS
- STRESS CORROSION
- EXFOLIATION
- MICROBIAL CORROSION

FIGURE 11. SIGNIFICANT FORMS OF AIRCRAFT CORROSION

STRUCTURAL MATERIAL SELECTION FOR CORROSION PREVENTION

- LIMIT HIGHLY SUSCEPTABLE MATERIALS (e.g. MAGNESIUM)
- USE OVERAGED HIGH STRENGTH ALUMINUM PRODUCTS
- LIMIT DISSIMILAR METAL CONTACT

FIGURE 12. STRUCTURAL MATERIAL SELECTION FOR CORROSION PREVENTION

Design Features for Corrosion Prevention

- Longeron Drainage to the Fuselage Bilge at the Inner Surface of the Skin
- Drain Holes at the Fuselage Bilge Center Line — Longerons off Center Line
- Fuselage Drainage Test to Assure Trap Areas are Gone
- Physical Separation of Insulation Blankets From the Fuselage Bilge — Plastic Egg Crate Sheet Between Frames
- No Cold Bonded Joints or Liquid Shims
- No Adhesive Bonding in the Fuselage Bilge
- Design for Minimum Unusable Fuel
- Jet Pump Automatic Sumping from Fuel Tank
- Landing Gear Static Joint Holes and Lugs Chrome Plated

Figure 13. Design Features for Corrosion Prevention

Supplemental Corrosion Treatments

Fuselage

- Detail Part Anopdize, Prime and Topcoat in Bilge
- Faying Surfaced Sealing of All Joints Leading to the Exterior
- Faying Surfaced Sealing of All Parts Attached to the Exterior Skin Below the Passenger Floor
- Liquid Displacements Sealant to Eliminate Fluid Trap Areas
- Form-in-Place Sealant Gaskets for Floors in the Lavatory, Galley and Entry Areas
- Application of Corrosion Inhibitive Compound to the Lower Fuselage

Wing Fuel Tank

- Detail Part Anopdize and Fuel Tank Coating
- Faying Surfaced Seal of All Joints Leading to the Exterior
- Faying Surfaced Seal of All Parts Attached to the Exterior of the Upper Wing Skin and the Front and Rear Spar Assemblies
- Faying Surfaced Seal of All Parts Attached to the Lower Wing Skin
- Liquid Displacements Sealant to Eliminate Fluid Trap Areas

Aircraft Non-Pressurized Areas

- Structure Under Fairings — Prime, Topcoat and Spray With Corrosion Inhibitive Compound
- Seal Leading Edge and Closing Ribs of Control Surfaces Leave Trailing Edge Open to Drain (Except Honeycomb)
- Coat Front Spar Assemblies of the Wing and Stabilizer (Wing and Horizontal) Under the Leading Edge With Corrosion Inhibitive Compound

Figure 14. Supplemental Materials and Processes for Corrosion Prevention

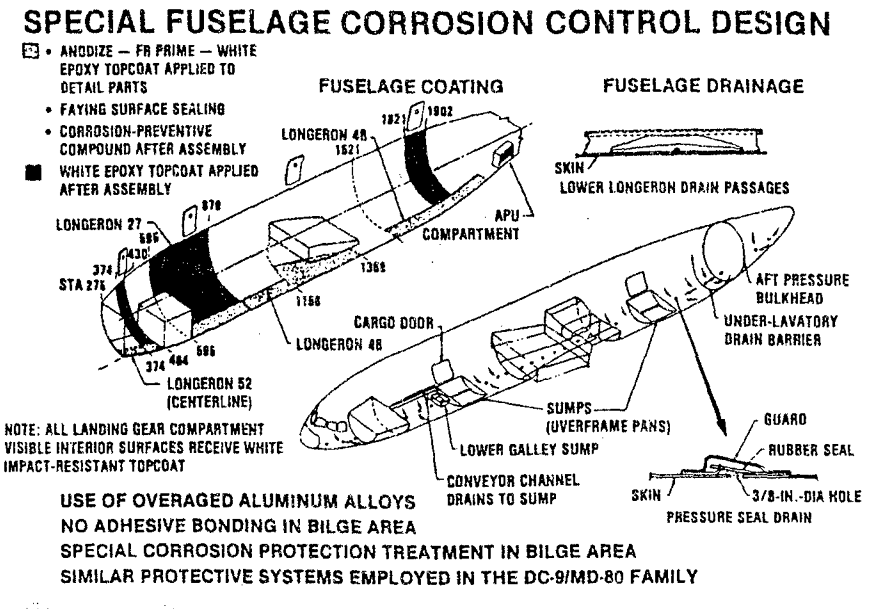

FIGURE 15. DC-10/MD-11 FUSELAGE DESIGN FEATURES FOR CORROSION PREVENTION

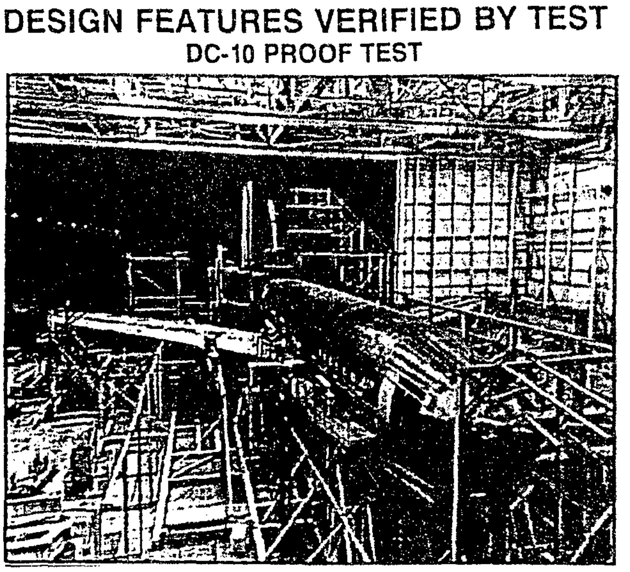

FIGURE 16. VERIFICATION OF DESIGN BY TEST

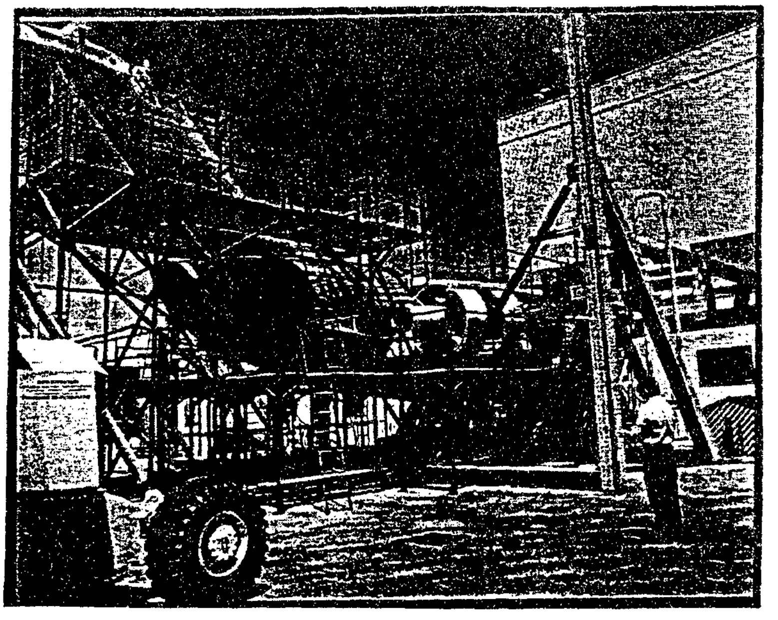

FIGURE 17. MD-90 PROOF TEST

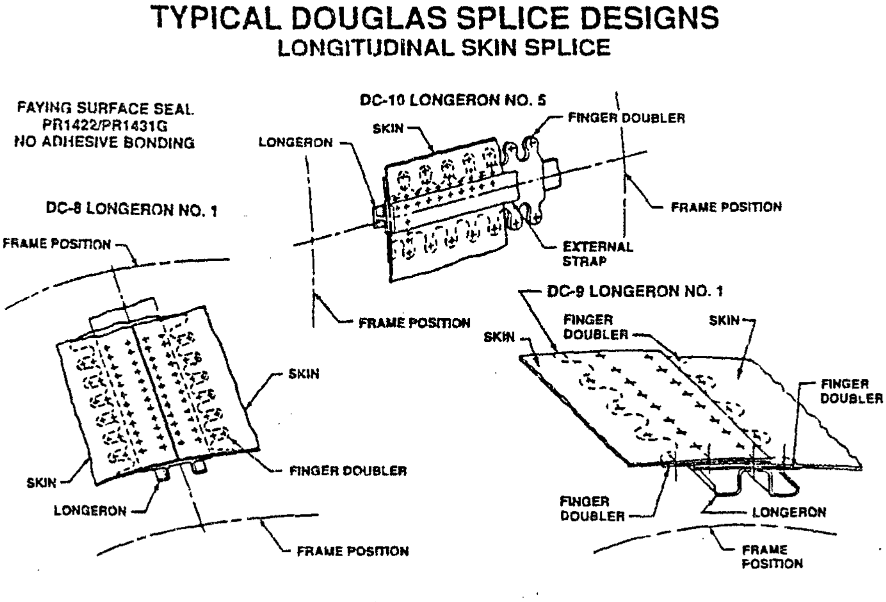

FIGURE 18. DOUGLAS PRESSURIZED FUSELAGE LONGITUDINAL SKIN SPLICES

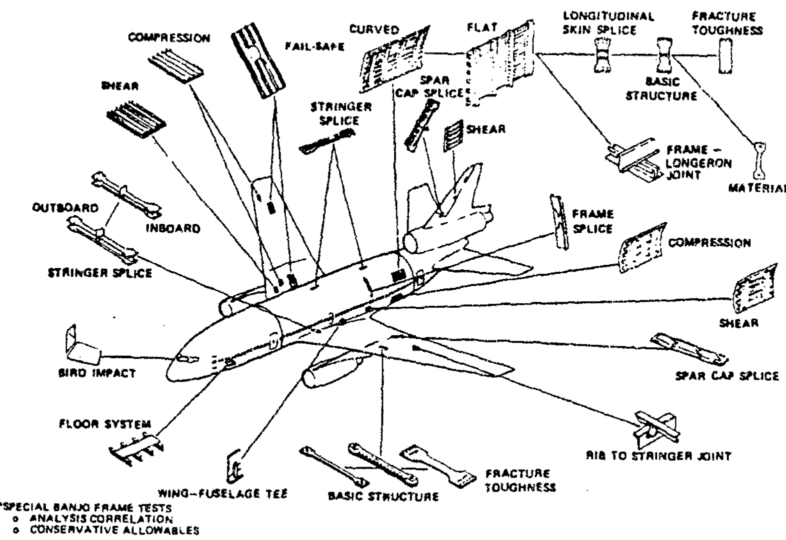

FIGURE 19. DC-10 STRUCTURAL DEVELOPMENT TESTS

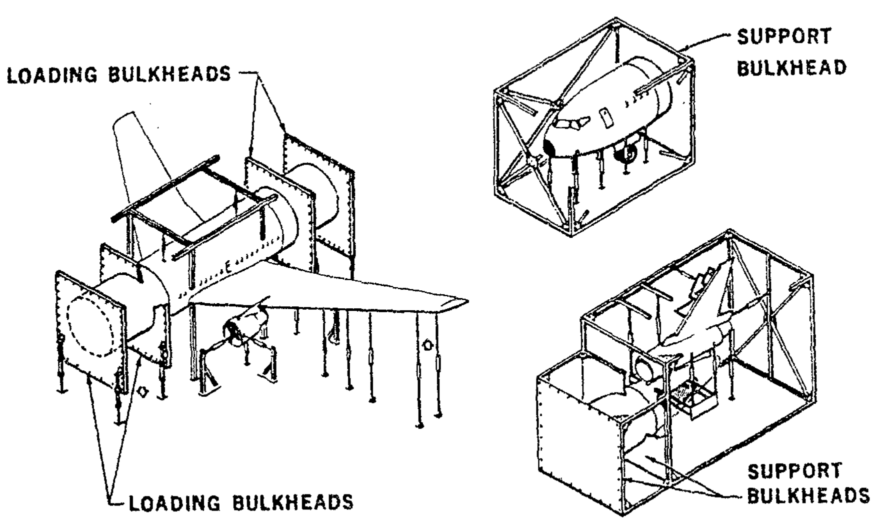

FIGURE 20. DC-10 FULL-SCALE FATIGUE TEST ARTICLE

MD90-30 FAA TEST REQUIREMENTS

STRUCTURE	PROOF/LIMIT	ULTIMATE	SONIC	FATIGUE	DAMAGE TOL.
T1.5 FUSELAGE AND PYLON	MAX UP LOAD	NO			
	MAX TAKEOFF RUN	NO			
	MAX DOWN LOAD	NO			
	INADVERTANT THRUST REVERSE	NO			
FUSELAGE	1.33 * MAX RELIEF	NO		3X LIFE CABIN	
PYLON	YES	NO	NO	3X LIFE	2X LIFE
PYLON FLAP	YES	FAIRED	YES		ANALYSIS
		HINGES			
ENGINE ISOLATORS					
FORWARD	YES	YES TO FAILURE		YES	YES
AFT	YES	YES		YES	YES
THRUST ISOLATORS	YES	YES		YES	YES
THRUST STRUTS	YES	YES		YES	YES
CLEARVIEW WINDOWS	BIRDSTRIKE				YES
SPOILER TEE BEARING	YES	YES			
GROUND VIBRATION TEST	ACCOMP ON T-1				

FIGURE 21. SUMMARY OF MD-90-30 STRUCTURAL DEVELOPMENT TESTS

THE MAINTENANCE PROGRAM

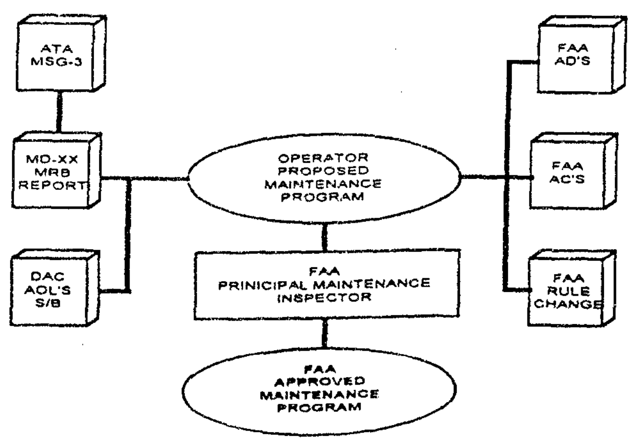

FIGURE 22. PROCESS FOR DEVELOPING A MAINTENANCE PROGRAM

DOUGLAS JET INVENTORY STATUS

MARCH 31, 1995

122,346,000 FLIGHT HOURS
85,629,000 LANDINGS

	PRODUCED	ACTIVE	INITIAL DELIVERY DATE	HIGH TIME ACTIVE AIRCRAFT	DESIGN GOALS	AIRCRAFT EXCEEDING DESIGN GOALS	TEST-SUPPORTED LIFE (CYCLES)	ACTIVE AIRCRAFT MEDIAN AGE (YEARS)
DC-8	556	309	1959	46,621 LDGS 86,223 FH 35.47 YEARS	25,000 LDGS 50,000 FH 20 YEARS	71 252 309	70,200 140,400	27.3
DC-9	976	688	1965	98,866 LDGS 78,638 FH 29.54 YEARS	40,000 LDGS 30,000 FH 20 YEARS	709 843 686	102,400 78,000	23.9
DC-10	446	419	1971	35,733 LDGS 90,332 FH 24.67 YEARS	42,000 LDGS 60,000 FH 20 YEARS	0 173 172	42,000 60,000	17.3
MD-80	1,120	1,111	1980	41,210 LDGS 44,988 FH 15.09 YEARS	50,000 LDGS 50,000 FH 20 YEARS	0 0 0	50,000 50,000	6.9
MD-11	131	131	1990	8,013 LDGS 19,543 FH 5.22 YEARS	20,000 LDGS 60,000 FH 20 YEARS	0 0 0	20,000 60,000	2.5
MD-90	5	5	1995	972 LDGS 1,182 FH 2.13 YEARS	60,000 LDGS 90,000 FH 20 YEARS	0 0 0	60,000 90,000	1.8
TOTAL	3,234	2,863						

FH/LDGS: 0 50,000 100,000
YEARS: 0 20 40

FIGURE 23. DOUGLAS PRODUCT LINE STATISTICS

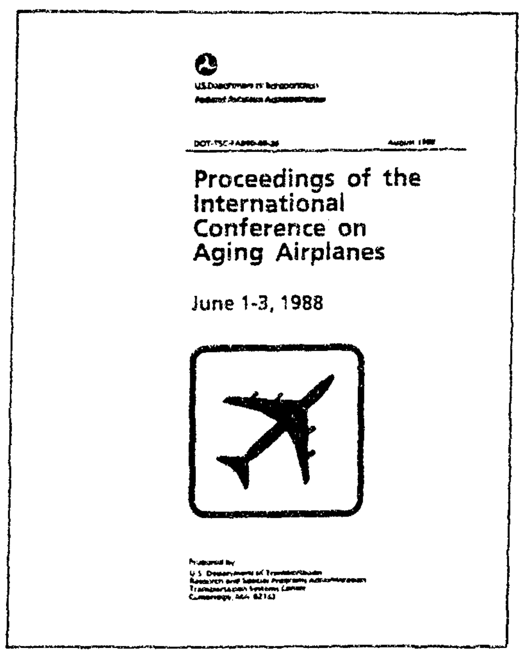

FIGURE 24. JUNE 1988 INTERNATIONAL CONFERENCE ON AGING AIRCRAFT

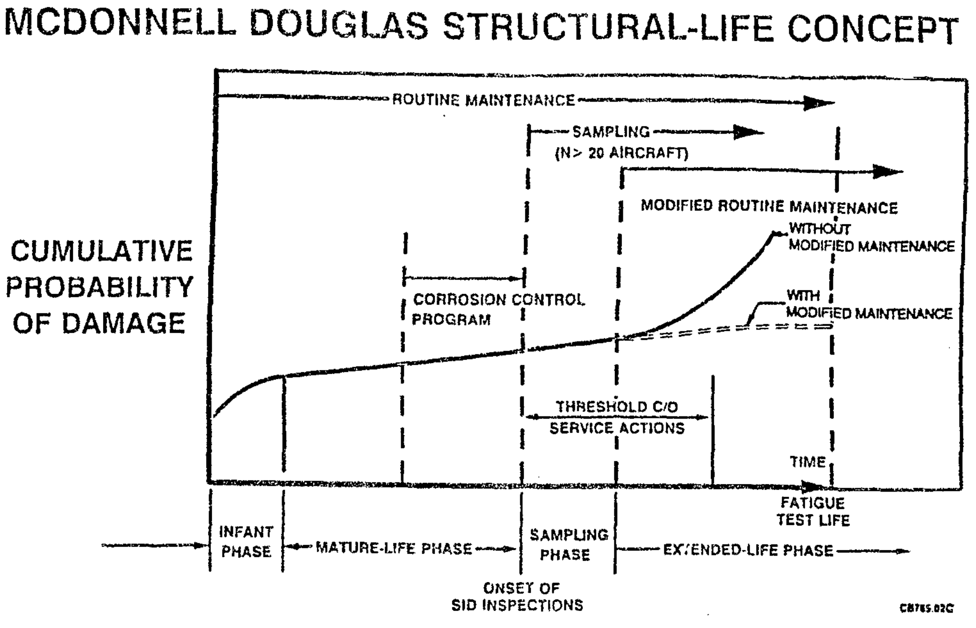

FIGURE 25. DOUGLAS STRUCTURAL LIFE CONCEPT

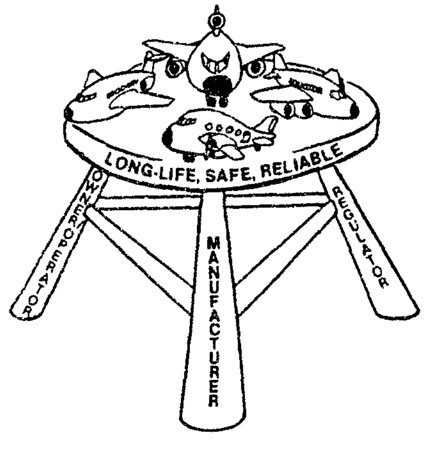

FIGURE 26. THE PARTNERSHIP REQUIRED FOR LONG-LIFE AIRCRAFT

TEST COMPLEX DEVELOPMENT TRENDS FOR AVIATION ENGINE CERTIFICATION

D.A.Ogorodnikov, V.A.Skibin, Yu.A.Nozhnitsky
CIAM, Moscow

S.B.Petrov
CIAM RTC, Lytkarino

Domestic aero-engine production entered the period of exchange relations. Even at the domestic Russian market there is an intense competition with the leading foreign aero-engine production companies. It means that on the one hand safe engine operation and the engine competitiveness have to be provided (including reliability, service life, economy, cost of maintenance servicing repair). On the other hand it is necessary to decline the unprofitable method of engine development and certification.

In spite of the rapid development of calculation methods, the experimental researches remain the most important and expensive aspect of engine creation. Researches are carried out both for direct checking to meet the certification requirements (certification tests) and for supplementary checks of engines, their details and components in conditions difficult for calculations, verification of calculation methods, data base formation on material structural integrity, new technologies implementation.

For a present day period it is typical to reduce the amount of full-scale engine tests, the number of engines used for engine development and certification. It is compensated for the effective usage of calculation methods and the increase of experimental data per a test. The role of assembly, separate component and sample tests increases. Certification tests (for example during the test on large-size bird ingestion to the engine face, the fan blade breaking, etc.) are realized on the following scheme "calculation — successful tests on an experimental facility-official test on the engine". The increasing of ultimate amount of information per a test is provided at the expense of calculations, automatic control, introduction of new methods of measurement and experimental data processing in real time. It is significant that there is a tendency to input data immediately into the computer without intermediate medium usage.

At the improvement of the equipment and methods necessary for experimental investigations and certification, it is essential to take into account the experience of previous generation engine creation and operation. On the other hand the potentiality of the advanced engines development and certification is to be provided.

The domestic industry experience gained in the previous years is analyzed in details. Besides, special attention was paid to the analysis of in flight shut down reasons and to non-localized destructions for the latest 25 years. The experience of foreign engine certification allowed to add the realized analysis with the comparison of home and foreign approaches to engine certification.

Currently a great number of engines from different foreign companies undergone certification tests in CIS. The basic modifications of such engines as: JT8D, JT9D, PW2000, PW4000 (Pratt-Whitney, USA), PW100, PW300, JT15D, PT6 (Pratt Whitney, Canada), CFG-6, CFG-50, CFG-80, CT-7 (General Electric, USA), RB211-535, RB211-524 (Rolls-Royce, Great Britain), CFM56-2,3 (CFMI merging General Electric and French SNECMA), V2500 (IAE consortium including companies from USA, Great Britain, Germany, Italy, Japan), TF-731, auxiliary power plant GTCP331 (Allied Signal, USA) obtained APMAK certificate.

The experience of certification of the above engines first of all showed that in spite of the difference in the approaches to the designing, certification, manufacturing and operation of engines, on the whole we have similar defects and measures to get rid of. At the same time domestic standards are significantly severe than the foreign ones in the respect of the requirements to the full-scale engine tests. Lately the foreign companies had done a great package of work to improve engine reliability. It was possible to develop large-thrust engines and to put a question forward about the issue of ETOPS certificate on 180 minutes flight from the moment of commissioning.

The carried out analysis showed the necessity of tightening approach towards localization of rotor components, assemblies, auxiliary power plants and rotor fan blades destruction (including the destruction of the whole working part of the blade). Certification is also reasonable at the so called "red line" conditions — the ultimate temperature and speed values allowable during the operation. It promotes the increasing of engine reliability and service life. Finally, methodology of service life verification should be developed in the line of endurance test amount reduction for the full-scale engines and in the line of methods improvement to prove the service life of the main details (data base usage on material structural strength in combination with high level calculations, application of destruction mechanics techniques).

In the new revisions of normative documents such as Airworthiness Standards (АП-33), Regulations for Determination and Increasing of Service Life of Civil Aviation Gas-Turbine Engines, Their Units and Complementary Parts (1994) — these approaches found their representation. At the same time the advantages of the domestic approaches were kept in such questions as:

— checking the compressor and fan blade self-oscillation absence;
— verification of absence of dangerous results at the disturbance of kinematic coupling between compressor and a turbine (destruction, disconnection of shafts).

It was mentioned that the development of experimental plant and test methods should provide advanced engines certification. Currently the works on propane engines and by-pass large-thrust engines (~400 kN) have begun. The works on engines using non-conventional fuels, engines for supersonic second generation passenger aircraft and aero-space plane propulsions have begun too. The task to use gas turbine engines developed on the basis of the aviation engines at stationary industrial installations is very urgent. Without a detailed analysis of peculiar problems arising from these engines testing, it can be mentioned that there is a necessity to provide tests of large size engines (large diameter and large air mass flow) and to increase parameters (temperature, pressure) along with the engine passage (especially in the engines for supersonic second generation passenger aircrafts and in the propulsions for aero-space planes). The role of engine/airframe integration increases. The range of problems dealing with verification of service life and reliability enlarges.

For successful application of new design and technological solutions of structures made of single-crystal, small grain and other advanced alloys of different composites, there is a necessity to carry out experimental researches. The program of experimental researches is to include the investigation of these materials deformation and destruction peculiarities. In addition, the following material peculiarities are to be taken into account:
— anisotropy of properties;
— reduced plasticity, etc.

The influence of new design and technological solutions (hollow widechord fan blades, blisks, blings, etc.) on strength and service life is to be taken into account also.

The unique experimental facility to test engines, engine components and details, to investigate material structural strength is available in CIAM. All engines for military and civil aircraft designed and manufactured in the former USSR were tested at the altitude test cells of CIAM and CIAM Research Test Center.

For 40 year period in CIAM RTC 1020 engines and their components have been tested and 40370 tests been performed.

Unfortunately, due to financial problems the amount of tests significantly reduced.

At the same time the program realization to update the aero-engine production experimental facility has begun. The program was developed with the mentioned above priorities taken into account. It is obvious this program fulfillment will depend upon financial support.

Some examples of the equipment and test methods improvement are presented in the report.

The strength branch of our industry is to undergo the definite changes. The efforts must be focused to a considerable degree on:

— strength certification of new engines;
— realization of new methodology to verify the service life;
— implementation of new materials, design and technological solutions.

It is possible to point out that the unique facility has been developed during the past few years to realize super high temperature tests (up to 1600—2000°C) of ceramic, carbonic and other new heat resistant composite materials.

CERTIFICATION OF DIGITAL AVIONICS COMPLEXES FOR THE NEW GENERATION OF CIVIL AIRCRAFT

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The paper concerns the problems of aircraft equipment certification.

Aircraft equipment performs the increasing number of functions. Electronic equipment avionics, plays a particular growing role and performs the tasks of automatic flight control from takeoff to landing, including en-route flight, and flight levels, time intervals and corridors keeping.

Avionics not only controls an aircraft, but also optimizes routes, flight levels, and engine conditions related to fuel consumption.

Flight safety depends much on the reliable performance of these functions by avionics.

Therefore, the requirements for equipment reliability and fail-safety of avionics complexes become more severe and can be satisfied only with careful rig tests and strict certification.

Qualification and certification of aircraft equipment is a time- and labour-consuming task. Successful qualification of accessories and systems of aircraft equipment requires a wide range of test rigs and facilities and special test procedures.

Rigs and sets for testing aircraft equipment have no such dimensions as wind tunnels, equipment of strength test laboratories, engine test chambers, but at the same time they provide complex facilities with precision measurement equipment, automatic mode control systems, parameter recording and processing devices.

Since the range of aircraft equipment is wide and aircraft equipment has different modes of operation, test rigs are also various in purpose and functions. Creation of ground-based test rigs and facilities for testing aircraft equipment is an independent engineering discipline and requires great efforts from design bureaus, factories and test centers.

Developers and manufacturers of aircraft equipment, however, spend a lot of money to reduce the risk and costs of flight tests. Particularly, it is true for systems which affect flight safety. Testing for some emergency modes such as short-circuits in power supply systems is difficult and unsafe, if tests are conducted onboard, and therefore a test rig is the only instrument of testing such conditions.

The Institute of Aircraft Equipment (NIIAO) is engaged mainly in developing and certifying integrated avionics complexes (flight control and navigation systems, radiocommunication complexes, power supply systems) and has the appropriate experimental facilities to support the work in these fields.

The development and certification of flight control and navigation complexes for the new generation of Russian civil aircraft such as IL-96-300, Tu-204, and IL-114, was one of the major work recently performed by NIIAO. The abbreviation "KSPNO" (transliterated from Russian) is used to designate these complexes.

The problems of qualification tests of aircraft equipment will be discussed using the work on KSPNO as an example.

But before considering the certification problems it is necessary to describe the performance of the complexes, i.e. the subject of certification. These are the first complexes developed in Russian Federation in accordance with the recommendations of ARINC Series 700 and Russian Airworthiness Requirements (NLGS-3) (see fig. 1). As compared with existing flight control and navigation equipment of IL-86, Yak-42, etc., the KSPNO provided the considerable expansion of the functions to be performed, improved reliability, decreased crew workload, and reduced operational costs.

The complexes provide the growing number of regular safe flights, including future separation requirements and ICAO Cat. I, II and III automatic approach and landing in IFR conditions. The complex configuration is shown in a block diagram (Fig. 2):

- Flight management system (FMS) ВСС
- Flight control system (FCS) ВСУП
- Thrust control system (TCS) ВСУТ
- CRT display system (full-colour) КИНО, КПИ, КИСС
- Strapdown laser inertial system (not completed, prototype flying)
- Radionavigation and approach systems:
    - Short-range navigation system (SRNS) РСБН
    - Long-range navigation system (LRNS) РСДН
    - VOR ВОР-85
    - ILS ИЛС-85
    - Radio altimeter (RA) РВ-85
- Weather radar (WNR) МНРЛС-85
- Air data system (ADS-85) СВС-85
- Flight envelope protection system (FEPS) СПКР
- Ground proximity warning system (GPNS) СППЗ
- Onboard maintenance system (OMS) ССЛО-85
- Integrated control panels КПРТС
- Standby instruments

The main performance of the complex is given in fig. 3:

Enroute navigation accuracy

— radio beacon zones 1-2.5 km

— reference-free land 1.85 km/h

Vertical separation	300 m to $H = 12$ km
MTBF	5, 000 hours
Readiness time	15-20 min.
Operation	by state
Specific working hours of maintenance	0.3 MH/FH

Fig. 4 presents the performance and number (redundancy) of systems incorporated in the complex.

The following performance are given:
weight;
reliability factors;
test coverage;
accuracy in accordance with ARINC recommendations;
dimensions, power consumption.

At present a microwave landing system (MLS) and a traffic collision avoidance system (TCAS) are not installed on aircraft since equipped aerodromes are not available for the first and the development is not completed yet for the second.

Complex interface is provided in accordance with ARINC 429. The systems included in the KSPNO complex were developed by a number of research institutes and design bureaus of aircraft and radio equipment industries.

To date the systems have undergone qualification tests, have Airworthiness Certificates which insure their compliance with NLGS, equivalent compliance for some cases, and are certified on the IL-96-300 and Tu-204 aircraft.

It should be noted that a part of modes provided by the complex are not certified due to the delay in software development and flight tests. These modes include ICAO Cat II and III landing, automatic vertical maneuver, integrated information processing. This work should be terminated in the course of this year.

A number of systems used in the KSPNO have been developed in the Russian Federation for the first time for application on civil aircraft. These include a strapdown inertial system, satellite navigation system, electronic display system, flight envelope protection system, etc.

Now we shall consider qualification and certification tests and test rigs and procedures used.

Current Russian standard rules provide for 2 test types: preliminary tests which are carried out by the product developer at the R&D stage, and state laboratory tests (or interdepartmental tests with the participation of the customer) which determine if a product complies with the requirements of the design specification, airworthiness requirements (NLGS) and other standards and specifications. Virtually, these are qualification tests. If tests performed by the developer and the customer give positive results, an Airworthiness Certificate is issued which certifies the compliance with airworthiness standards and the possible installation on aircraft.

In addition to the tests of individual units of equipment , the entire complex is tested for fail-safety. Software certification is a separate major problem.

Tests of each system, instrument and unit can be conventionally classified by their character in 2 types:

- Tests performed to show special characteristics specified for a given system: accuracy, parameter drifts, output of valid digital signals, radiation power, receiver sensitivity, applied pressure, cooling power, output voltage, etc. There are hundreds of such parameters to be tested, e.g. 48 parameters are tested for a strapdown inertial system only. They are specific for each type of equipment and therefore rigs and sets for testing such parameters are individual, except measuring devices, and will not be discussed here since it will take too much time. Each test rig is of interest only to a close circle of specialists. These tests are carried out by the developers.
- Tests applied practically to all types of equipment and units. These include environment, EMC, lightning protection, explosion-proofness, reliability, power input tests. The tests also require a wide range of test rigs and test facilities to be used. These tests will be discussed in more detail. Fig. 5 shows environmental factors and corresponding test types.

It can be seen in the table that 51 tests are required to estimate the effects of only 6 environmental factors.

As concerning the types of environmental factors, 9 test types are conducted for <u>mechanical effects</u> (fig. 6). These include the tests of a structure for resonance, vibration, shock, acceleration, acoustic noise. 4 types of test rigs, ignoring their load-carrying capacity, are needed for these tests.

As a rule, vibration and shock test rigs and centrifuges are stock-produced by many companies and can be purchased. A noise chamber in use is in-house-produced. For random vibration tests, vibration rig control equipment is required to generate specified spectra. The test rigs should contain measurement equipment for analyzing the characteristics of random processes. The tests should show the compliance with the requirements of Appendix 8.1 which are in agreement with DO-160C requirements, except some insignificant differences in test procedures for vibration tests and in loads at low frequencies. Our designers of instruments and radio equipment have a rich experience in designing strong and vibration-proof equipment. The cases when equipment does not stand tests happen rarely. The IL-96-300 and Tu-204 aircraft are "quiet" and real mechanical effects on equipment are below the levels specified in the standards. Tests were performed mainly at the test rigs of the developers.

Climatic Effects (fig. 7)

To determine compliance with the airworthiness requirements (NLGS, App. 81 or DO-160) it is necessary to carry out up to 18 test types, including 3 test types for different pressure variations and 7 types related to temperature, humidity, spray, sand and dust, solar radiation, fungus, etc. tests. 10 different test rigs such as thermal pressure chambers, salt spray chambers, sand-and-dust chambers, solar radiation, overpressure chambers, etc., are to be used for conducting the tests. The systems of the KSPNO were tested by both the developers and NIIAO.

Test chambers are generally stock-produced by a number of companies. Test sets for large-sized products are custom-made. The Institute has the test chambers, except sand-and-dust ones. Unique facilities are also available, e.g. the thermal pressure chambers with a volume of 60 m^3, altitude tolerance of up to 40 km and possible temperature variations from $-60°C$ to $+500°C$. A drive shaft is used with rate of 24 000 rpm and the possible air flow rate of 5, 500 kg/hour that provides testing electric generators. Air conditioning systems for different aircraft classes are tested in this chamber. Also, a rocket, fighter cockpit and other large-sized products can be tested. The chamber is equipped with the automatic parameter control system (for altitude, temperature and their rates) and the measurement recording and processing system. Climatic effect requirements specified in App. 8.1 essentially comply with DO-160C.

Currently a test center is created at the Institute. This center is certified by the IAC AR and the State Standard Committee.

The Institute developed the engineering documentation and manufactured the prototypes of an environmental test system built using a modular design. The base is a typical module (chamber) with a volume of 1 m^3 where typical units such as heaters, cooling units, vacuum pumps, vibrators can be attached as required to transform the compartment in a thermal chamber, or a thermal pressure chamber or a thermal pressure vibration test chamber. The required volume of the chamber is provided by means of several typical modules. Such an approach reduces the rig costs and makes the test rigs more flexible.

A separate paper on mechanical and climatic tests of equipment and development of test rigs is presented in the Aircraft Equipment Committee of the Conference.

Electromagnetic Compatibility (EMC) (fig. 8)

To determine the compliance of equipment with EMC requirements it is necessary to carry out 12 test types as a minimum, including 3 tests for interference generation and 9 tests for susceptibility to noise over power lines, and transmission lines and to electromagnetic field effects on cases at different frequencies.

8 test rigs are required for performing these tests. The test rigs should be installed in a special certified shielded room. Almost all signal generators, measuring instruments and systems are foreign-produced (e.g. Rode-Schwarz).

The home industry produces only a limited range of standard-signal generators. Our design bureaus are poorly provided with test rigs and measuring equipment necessary for these tests. The considerable amount of testing for the KSPNO was provided by NIIAO. Since the skills of the designers in providing specified parameters of equipment under development, is inadequate, 11 systems included in the KSPNO do not conform to the requirements. The basic non-conformities relate to susceptibility for power lines and transmission lines within individual frequency ranges. The tests are performed for compliance with App. 8.14.1 which is identical with DO-160B.

The problem of providing avionics EMC becomes more complicated and urgent. On the one hand, electromagnetic onboard environment aircraft deteriorates due to the increasing intensity of external noise since the number of radio stations, radars and other radiating facilities grows and their frequency ranges and radiation power are extended. The same situation is observed on aircraft. At the same time digital equipment and integrated circuits sensitive to interference are finding increasing use in aircraft equipment. The interference voltage of 2 to 3 V can result in equipment failures that led in many cases to serious results of flight operations. Therefore this problem is the focus of great attention.

US standards continuously become more strict. So, DO-160C extends the frequency range for equipment tests from 1.2 to 18 GHz and the signal level required to evaluate equipment susceptibility from 2 to 200 V/m (hundred fold).

Additional requirements for high-intensity radiation field (HIRF) have been developed for systems which perform "critical functions". These requirements are specified in CRI-SE-10.

For this purpose the rig tests are introduced to test susceptibility to HIRF with a level of 100 V/m within the range from 10 kHz to 18 GHz.

Now the Institute is developing special test rigs to provide these new test types. Thermal chambers and corn chambers are manufactured to satisfy these additional requirements. The NIIAO test rigs are currently being certified by the State Standard Committee. This work will be completed in the third quarter this year.

The paper on EMC certification will be presented in the Aircraft Equipment Committee of the Conference.

Power Input

To determine the compliance of equipment with airworthiness requirements it is necessary to carry out 8 test types (fig. 9):

- voltage variation;

- frequency variation;
- surge voltage;
- voltage modulation;
- ripple voltage (DC);
- transient voltage;
- power interrupts;
- voltage spike.

Certification is conducted to show compliance with App. 8.1.5, which practically conforms to DO-160C.

NIIAO developed and operates a special test rig. The purpose of the rig is to change the characteristics of power input to equipment under test within the limits specified in the equipment performance standards, i.e. to change voltage and frequency and provide power interrupts, voltage spike and surge voltage. The test rig is not complicated, but only a few design bureaus have such test rigs. Many systems of the KSPNO were tested at the Institute's test rig. The industry does not provide the serial production of these test rigs.

Although power input requirements have been valid for many years, they are not always met by equipment developers. Aircraft equipment often does not withstand power interrupts and voltage spike. Computer memories are especially sensitive to power interrupts and many microcircuits are unstable to overvoltage.

10 systems of the KSPNO do not meet power input requirements. They are not resistant mainly to power interrupts. Equivalent conformity is achieved by providing power supply from the portside and starboard power-generating systems that reduces the probability of power interrupts to a level required by the equipment performance standards. So, the problems remain and the developers of aircraft equipment should focus their attention on solution.

The test rig intended for testing equipment power input is incorporated in the NIIAO test center. This problem will be also discussed in the Committee.

Lightning Effects

Lightning resistance requirements are specified in NLGS-3. So, "when lightning current flows through the aircraft structure, functional systems and units must have no failures or false operations which can lead to an emergency or catastrophic situation".

Equipment may be tested only on aircraft where discharge current is applied. According to the design cycle this event happens when an aircraft has been already built and equipment is available in developmental prototypes as minimum. If test results are negative, time required for retrofit is not left. Therefore, lightning resistance is determined using calculated estimates and in some cases tests are carried out at special test rigs. The following estimation procedure is used.

The General Designer of an aircraft specifies safety-critical systems. One of the four equipment categories is determined depending on installation locations in the fuselage (fig. 10). Then the effect of rated voltage pulses (3 waveforms) on equipment inputs is estimated.

The Institute developed a special program which enables the resistance of electronic components of input devices to voltage pulses to be determined on the computer using the data on their circuits and electric parameters, and a weak component to be detected which can be replaced or protected from overvoltage.

Equipment retrofitted based on estimation results can be tested, if necessary, at a test rig with applying specified pulses to equipment input. Such a test rig is owned by the Flight Research Institute.

Lightning resistance is finally proved on aircraft during lightning protection tests. If an aircraft passes lightning protection tests before equipment is tested, voltage measured in an equipment installation location rather than rated pulses is used in calculations and rig tests. There are also means which can be used to protect input devices of equipment from voltage pulses. The effects of lightning strikes can be severe, even catastrophic. The paper on lightning resistance of equipment will be discussed in the Aircraft Equipment Committee of the Conference.

Explosion Proofness

Explosion and spark proofness requirements are specified for separate types of equipment that may come into contact with fuel and its vapors in compartments and evolves, while operating, energy enough to cause ignition of a fuel-air mixture or produces sparks.

Test requirements are specified depending on environments where equipment is installed and operates. The requirements are as follows:

Equipment should be designed so that if an explosive mixture penetrates in it, equipment is capable to confine fire inside without case destruction, or should be hermetically sealed. And if equipment produces sparks, spark energy must be so low that the ignition of the mixture at a specified concentration be eliminated.

Special test procedures and complicated test rigs are used, particularly test rigs of research institutes of mining industry.

As a result of testing the systems of the KSPNO for compliance with NLGS, all systems included in the KSPNO were certified before installation on aircraft, but 18 of these systems demonstrated equivalent compliance (fig. 11) with separate requirements, principally EMC and power input requirements.

Fail-Safety

The objective of all efforts of equipment and complex certification is to provide safe operation of aircraft. Therefore. the work aimed to prove fail-safety of the complex is the most important in the certification process. This is a very complicated, labor-and time-consuming work in the process of which it is necessary to show that any single failure or failure combination will not cause a catastrophic or complicated situation more often than specified in the specifications and standards.

Since the KSPNO consists of three tens of systems, each system containing hundreds of electronic components, and interfacing complex systems perform tens of functional tasks, and the complex is connected to a number of interfaced systems, the determination of fail-safety becomes problematic and requires a great amount of efforts.

At present the following procedure of fail-safety determination is used (fig 12):

I stage: fail-safety of systems included in the complex is determined.

II stage: functional and multifunctional failures are analysed.

The number of failures is 100 to 200.

The types of failures are 300 to 1, 000.

The number of failure combinations is several millions. Manual selection of all failures and failure combinations is impracticable. Only individual failures selected using expert estimates were analysed. Validity of estimates was inadequate.

Now, NIIAO has developed software which enables all possible conditions of the complex, including various combinations of component failures, to be searched considering one, two and three failures both top-down from functions and bottom-up from systems and components (fig. 13). If the number of failure types is 400, conditions under estimation are 10 million.

While estimating CPNK-114 fail-safety in enroute flight, the number of conditions with hardware failures was 2.7 million and the number of complex failure conditions for which the probability exceeds 10^{-9} was 119.

The result of functional failure analysis is determining:

- a list of functional failures;
- failure modes;
- failure representation;
- failure probability;
- classification of abnormal situations;
- recommended crew actions.

The functional failure analysis results in detecting the most hazardous failures to be tested by an integrated test and simulation rig (ITSR).

ITSRs were made by the Institute for particular aircraft such as IL-96-300, Tu-204, IL-114.

The configuration and performance of a ITSR is given in fig. 14.

The test rig contains:

- a cockpit mock-up with real hardware, including an information display system;
- an aircraft control system with real load;
- a computer system on which an aircraft model and a KSPNO system model are implemented;
- a visual situation simulator;
- system simulators;
- a rack for installation of KSPNO units;
- test instrumentation.

The test rig is used to develop an avionics complex with software, to train test pilots and determine fail-safety in the certification process. Failure effects are assessed and recommendations for crew reactions to failures are developed. The recommendations are evaluated by test pilots at the test rig. A part of hazardous and complicated failures are tested in flight on aircraft.

The ITSR is a unique, complicated and expensive facility with wide functionality. For example:

Number of simulated channels

1.	Output:	bipolar code	144
		discrete	152
		analog	10
2.	Input:	bipolar code	26
		discrete	52
		analog	10

Recorded information: 14000 kB.

Number of received signals:	bipolar code	150
	discrete	200.

Reliability of Complex Systems and Equipment

Reliability is estimated in the development process at the stage of preliminary design using available λ-characteristics of elements and is refined at the stage of detailed design.

The estimated characteristics of reliability are used for analysing functional failures of complex systems and equipment.

The λ-characteristics of elements are corrected depending on the operation conditions of equipment (environment and load) to improve the validity of estimations.

Laboratory reliability tests are carried out to obtain experimental data on a reliability level before full-scale production and operation. These tests are of long-duration and expensive that results in great problems in test management.

Considering these circumstances, NIIAO developed an original procedure of reliability cycle tests. The procedure is briefly as follows (see fig. 15):

- specifying equivalent effects on equipment that replace 1 year of operation in specified conditions (cycle);
- estimating the number of test cycles as a function of system reliability and the power of forcing environmental factors.

The procedure of generating an accelerated reliability test cycle is as follows:

- determining external environmental factors;
- specifying a test level for each environmental factor;
- selecting a test procedure and specifying the exposure time for each factor;
- determining the duration of a cycle;
- determining the coefficient of test acceleration.

Test levels are specified to be higher than in operation, but such that the limit of equipment stability be not violated (sometimes special tests are performed for this purpose).

Non-informative effects (aircraft parking, cruise flight at a flight level) are replaced by forced effects.

The following is taken into account in generating the element or cycle equivalent to 1 year:

- number of flight hours per annum;
- number of takeoffs/landings;
- climatic area of operation;
- number of equipment on/offs;
- variation of supply voltage (50% V, 25% < V, 25% > V);
- mechanical and climatic effects on equipment.

Experience of estimating test cycles enables the acceleration to be provided that is 20-30 times as high as in real operation (by the calendar) and the before-operation estimate of reliability to be obtained and used for modifying equipment and determining a kit of spares and repair techniques.

The procedures are widely used for tests of military and civil aircraft equipment, and show the good convergence with results of designed operation and cost effectiveness. A great amount of tests were performed using the Institute's test rigs. The CAD system is available for developing a test program and determining a test cycle.

Software Certification

The role of software (SW) in digital avionics complexes is increasing. The size, complexity and labor consumption for SW development is growing and now come to hundreds of man-years.

Naturally the costs of SW development grow and now in a number of cases exceed the costs of hardware development.

SW quality insures the correct operation of systems and complexes and directly affects the safety of aircraft operations.

A SW level is determined in criticality by the system effect on flight safety.

A SW level specifies a list of required checks, test operations, and testing.

The highest requirements to SW testing for the 1st level are:

- determining the compliance between the system requirements and the SW requirements;
- determining the compliance between SW design and code;
- testing SW modules;
- testing the cooperation of SW modules;
- testing the hardware and software operation.

Testing is carried out at special programmers' workstations. The cooperation of SW and HW is tested by the developers' test rigs and NIIAO ITSR.

The KSPNO SW was certified by NIIAO for IL-96-300 and Tu-204 to substantiate the compliance with App. 8.1.6 developed by the Aviation Register WG-23.

Currently SW is being developed for new integrated avionics complexes of the Be-200 and Tu-224 aircraft in the cooperation with AlliedSignal, Inc. (USA) and Sextant Avionique (France).

These activities encourage the improvement of the basis required to harmonize Russian and Western standards and to bring the up-to-date technologies of SW development to a commercial level.

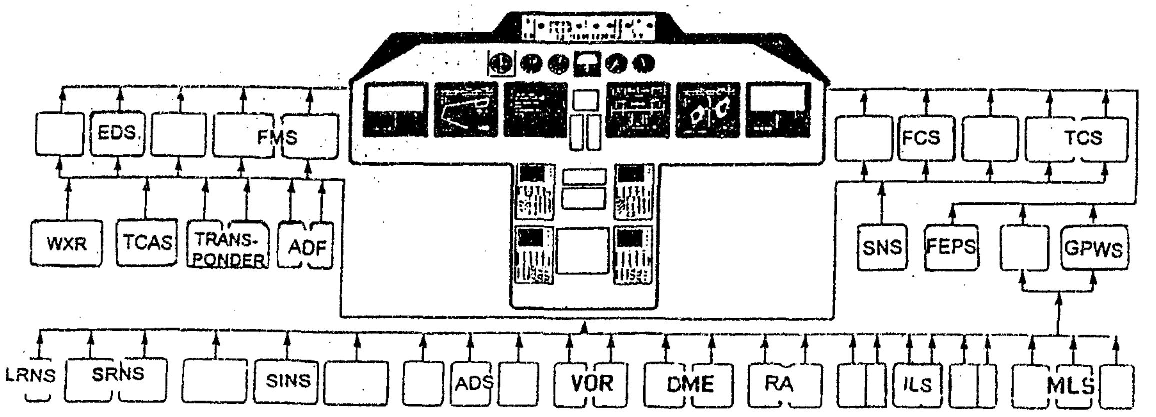
This year a document which replaces App. 8.1.6 and is in complete agreement with DO-178A and a document which agrees with DO-178B based on the experience in the certification of the integrated avionics complex for Be-200, will be issued.

Thus, by considering a specific example of certifying a digital flight control and navigation complex, the amount of testing, necessary test rigs (over 25), standards for which equipment is tested, and test procedures have been shown.

These problems will be discussed in more detail at the sessions of the Committee.

Now we are going from NLGS-3 to Aviation Rules (AR) which are harmonized with FARs. The procedures testing practically are not changed, but test management, responsibilities and document titles are modified.

COMPLEX OF STANDARD DIGITAL FLIGHT CONTROL AND NAVIGATION EQUIPMENT FOR IL-96-300 AND TU-204



PURPOSE:

Complexes of digital flight control and navigation equipment for trunk-route passenger aircraft IL-96-300 and TU-204 are designed to improve flight safety and regularity of operations on domestic and international air routes under IFR conditions, to provide future separation requirements and approach and ICAO Cat III automatic landing.

BENEFITS:

- increased number of functional tasks
- improved reliability
- simple reconfiguration
- decreased crew workload
- reduced operational costs

Fig.1

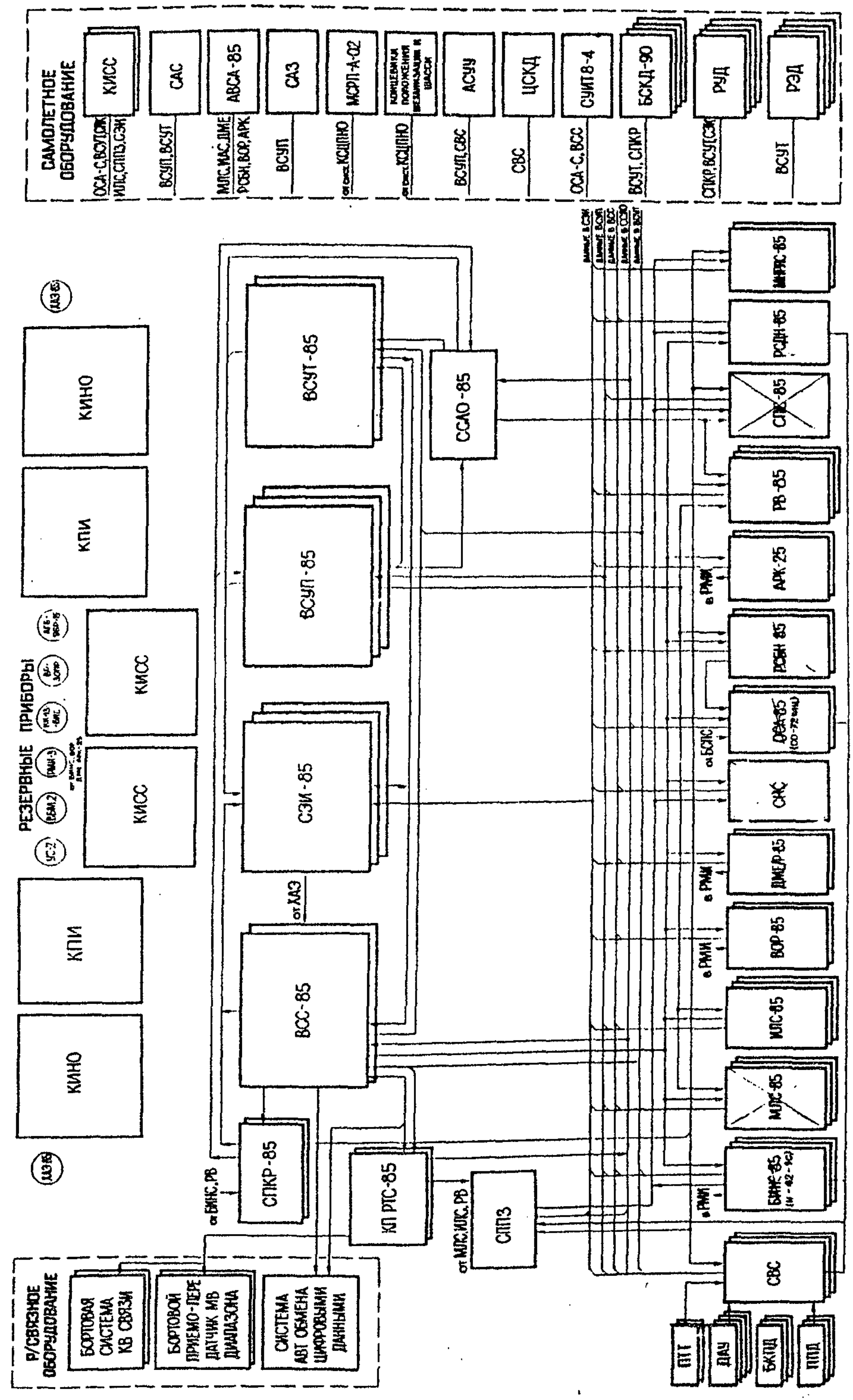


Fig.2 BLOCK DIAGRAM OF FLIGHT CONTROL AND NAVIGATION COMPLEX KSCPNO-96-300, 204

FLIGHT PLAN PROGRAMMING	~70 ROUTES
NAVIGATIONAL ACCURACY	
ENROUTE (IN RB ZONE)	1-2.5 km
REFERENCE - FREE	1.85 km/h
LAND AND NORTH ATLANTIC	
APPROACH ZONE	1-2 km
CIRCLE ZONE	0.7-1.5 km
VERTICAL SEPARATION	300 m, up to H=12 000 m
CRP PASSAGE TIME ACCURACY	0.9 min at a distance of up to 200 km
DESIGNED MTBF FOR SYSTEMS AND UNITS	~5 000 h
READINESS TIME	15-20 min
MAINTENANCE	by condition
TEST COVERAGE	~0.95
MEAN TIME TO RECOVER	15-20 min
SPECIFIC WORKING HOURS OF MAINTENANCE	0.3 MH/FH

Item	SYSTEM	NUMBER OF UNITS	WEIGHT OF UNIT kg	SYSTEM WEIGHT kg	T_C, HOUR	TEST COVERAGE	ACCURA-CY	DIMEN-SIONS	POWER CONSUMPTION 115VAC, 400Hz /27VDC
1	FCS FCC CP	3 1	12 16	42	6000 11000	$P_{HO}=10^{-9}$		8	150×3 75
2	FMS CC CDU	2 2	15 65	43	800 500	0.99		8	170×2
3	TCS TCC TCP WS	2 1 1	15 2 10	42	500 11500	0.9/0.95		6	150×2/25
4	SINS	3	20	60	5000	0.9/0.95	AR704	10	200×3/180×3
5	ADS	2(3)	7	14(21)	10745	0.9/0.95	AR706	4	50×2(3)
6	RA TR ANT	3 3	6 0.8	20.4	500	0.95 0.95	AR707	3	50×3
7	WNR TR ANT	2 1	15 11.5	40	2000	0.95	AR708	8	200×2
8	DME INTER ANT	2 2	8 0.3	16.6	5000	0.99	AR709	4	100×2
9	ILS REC/ILS REC.LS ANT	3 3 1	5 1 2.6	20.6	5000	0.99	AR710	3 1	50×3
10	VOR REC ANT	2 1	5 2.9	12.9	5000	0.99	AR711	3	50×2
11	ADF REC ANT	2 2	3.8 5.9	19.4	3000	0.95	AR712	2	40×2
12	Aircraft Transponder RESP RESP ANT	2 2 2	6 6 2	28	2000	0.9/0.95	AR718	4 4	120×2
13	RMP	2	9		5000	0.9			30×2
14	GPWS	1	5	5	6645	0.9/0.95	AR723	2	30
15	EDS SG DU CP	3 4 2	9 14 15	85	500 T_0=1.4×10^{-7})	0.9/0.95	AR725	6	1290
16	MLS REC ANT	3	6	18	5000	$P_{HO}=10^{-6}$	AR727	4	40×3
17	Aircraft Clock	2	2	4	5000		AR731		-/10×2
18	FEPS	2	6	12	5000	0.9/0.95		4	100×2
19	SRNS	2	30	60	510				200×2
20	LRNS	1	13	13	5000	0.9/0.95		8	130
21	OMS	1	12	12	5000	0.9/0.95		6	25
22	TCAS COMP BEARING UNIT	1 1 1	10 15 5	30	5000	0.99	AR730	4 6 3	310
23	SNS UNIT ANT	1	15 1	16	5000	0.9/0.95		6	150
24	RMI	1	3.5	3.5	5000	0.9			50
25	GH	1	2.5	2.5	5152	0.9			6/40

Test types for aircraft equipment

N	Environmental factors	Number of test types
1	Mechanical effects	9
2	Climatic effects	18
3	Electromagnetic compatibility	12
4	Power input	7
5	Ligtning effects	3
6	Explosion proofness	2
7	Total:	51

Mechanical effects

N	Test types	Test rig
1	Structure resonances	Vibration test rig
2	Vibration resistance	
3	Vibration strength	
4	Shock resistance	Shock table
5	Shock strength	
6	Shock strength of attachments	
7	Linear acceleration resistance and strength	Centrifugs/ linear accelerator
8	Linear acceleration strength of attachments	
9	Acoustic noise resistance	Reverberation chamber

Climatic effects

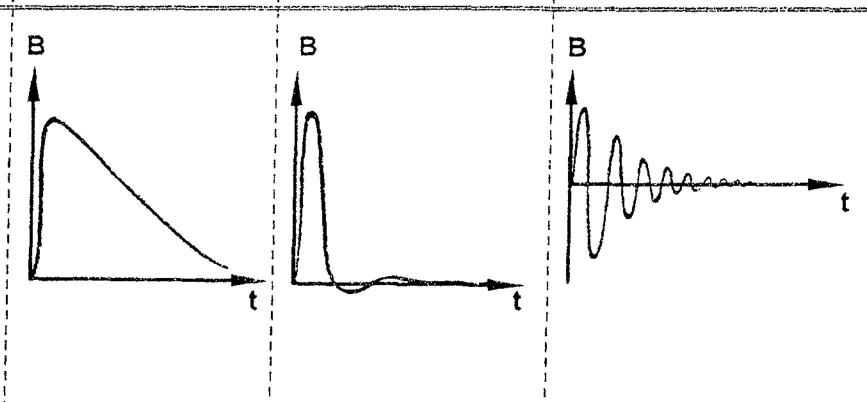
N	Test types	Test rig
1	Low pressure	Pressure chamber
2	Depressurization	
3	High pressure	Overpresure chamber
4	High operating temperature	Temperature chamber
5	Low operating temperature	
6	Short-time operating high temperature	
7	Ground survival high temperature	
8	Ground survival low temperature	
9	Rapid temperature variation	
10	Cyclic temperature variation	
11	Humidity	Humidity chamber
12	Salt spray	Salt spray chamber
13	Dew and icing	Pressure temperature humidity chamber
14	Sand and dust	Sand and dust chamber
15	Solar radiation	Solar radiation chamber
16	Fungus resistance	Fungus chamber
17	Drip proofness	Shower test chamber
18	Spray proofness	

Electromagnetic tests

N	Test	Test rig
1	Generation of interference voltage and current in power lines	1. Generation of interference in aircraft wiring
2	Generation of interference current in transmission lines	1. Generation of interference in aircraft wiring
3	Generation of electric intensity of interference electromagnetic field	2. Generation of electromagnetic field by wiring and cases of aircraft equipment
4	Susceptibility to magnetic effects on enclosures of aircraft equipment	3. Susceptibility to 400Hz fields
5	Susceptibility to audio-frequency magnetic fields induced into interconnecting cables	3. Susceptibility to 400Hz fields
6	Susceptibility to electric fields induced into interconnecting cables	3. Susceptibility to 400Hz fields
7	Susceptibility to fields induced into interconnecting cables	4. Transient fields susceptibility
8	Susceptibility to radio-frequency interference radiated and conducted in power lines	5. Radio-frequency susceptibility
9	Susceptibility to audio-frequency interference conducted in power lines	6. Audio-frequency susceptibility
10	Susceptibility to radio-frequency interference radiated and conducted in transmission lines	5. Radio-frequency susceptibility
11	Susceptibility to the effects of radio-frequency magnetic fields on enclosures of aircraft equipment	7. Radio-frequency magnetic field susceptibility
12	Susceptibility to the effects of radio-frequency electromagnetic fields on enclosures and interconnectic wiring	8. Radio-frequency electromagnetic field susceptibility

№	Test types (effects)	Power input parameters
1	Voltage (frequency) steady state	108,119,102,124,97,134 VAC 380,420,370,430 Hz 24,29,4,18,31,21,33 VDC
2	Surge voltage (AC)	Ktotal = 8% Kharm = 5% Kam = 1,41+0,15
3	Voltage amplitude modulation (AC)	Kam = 1%
4	Ripple voltage (DC)	±2 V (7. 4%)
5	Transient voltage	AC 160 VAC] 0,1s 180 VAC] 7s 60 VAC 0 VAC DC 40(70) VDC] 0,1s 45(80) VDC] 7s 13(8) VDC 0 VDC
6	Power interruptions	80 ms, normal operation 7 s, abnormal operation
7	Voltage spike	±600 V 2 - 10 μs

	AMPLITUDES AND RATED LIGHTNING VOLTAGE PULSES		
CATEGORY	LONG WAVE	SHORT WAVE	SINUSORIAL WAVE
	Um, V	Um, V	Um
1	125	125	250
2	300	300	600
3	750	750	1500
4	1600	1600	3200



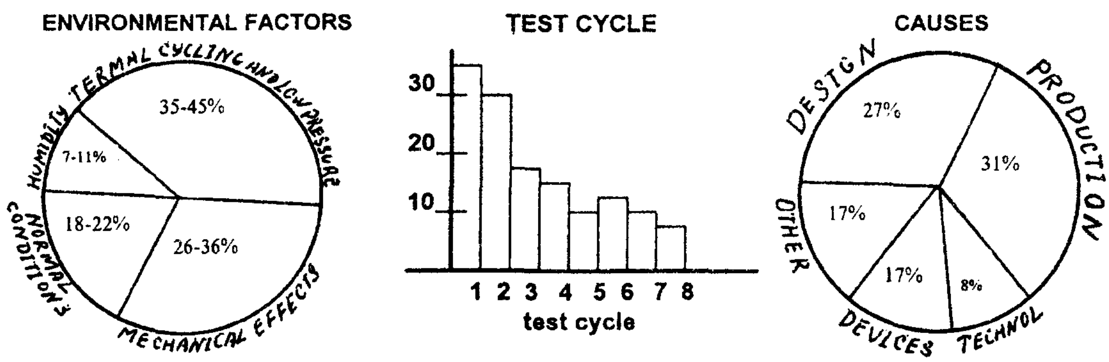
Item	System	Equivalent copliance				
		mechanical	climatic	power input	EMC	Special requirements
1	И-42-1с					
2	FCS					
3	TCS					
4	EDS					
5	FEPS					
6	SI					
7	VSI					
8	PITOTUBE					
9	FMS					
10	ILS					
11	VOR					
12	DME/P					
13	SRNS					
14	TRANS-PONDER					
15	"ASTRA" antenna feeder system					
16	WNR					
17	LRNS					
18	ADF					

PROCEDURES OF ACCELERATED EQUIVALENT
CYCLE TESTS FOR RELIABILITY OF AIRCRAFT EQUIPMENT

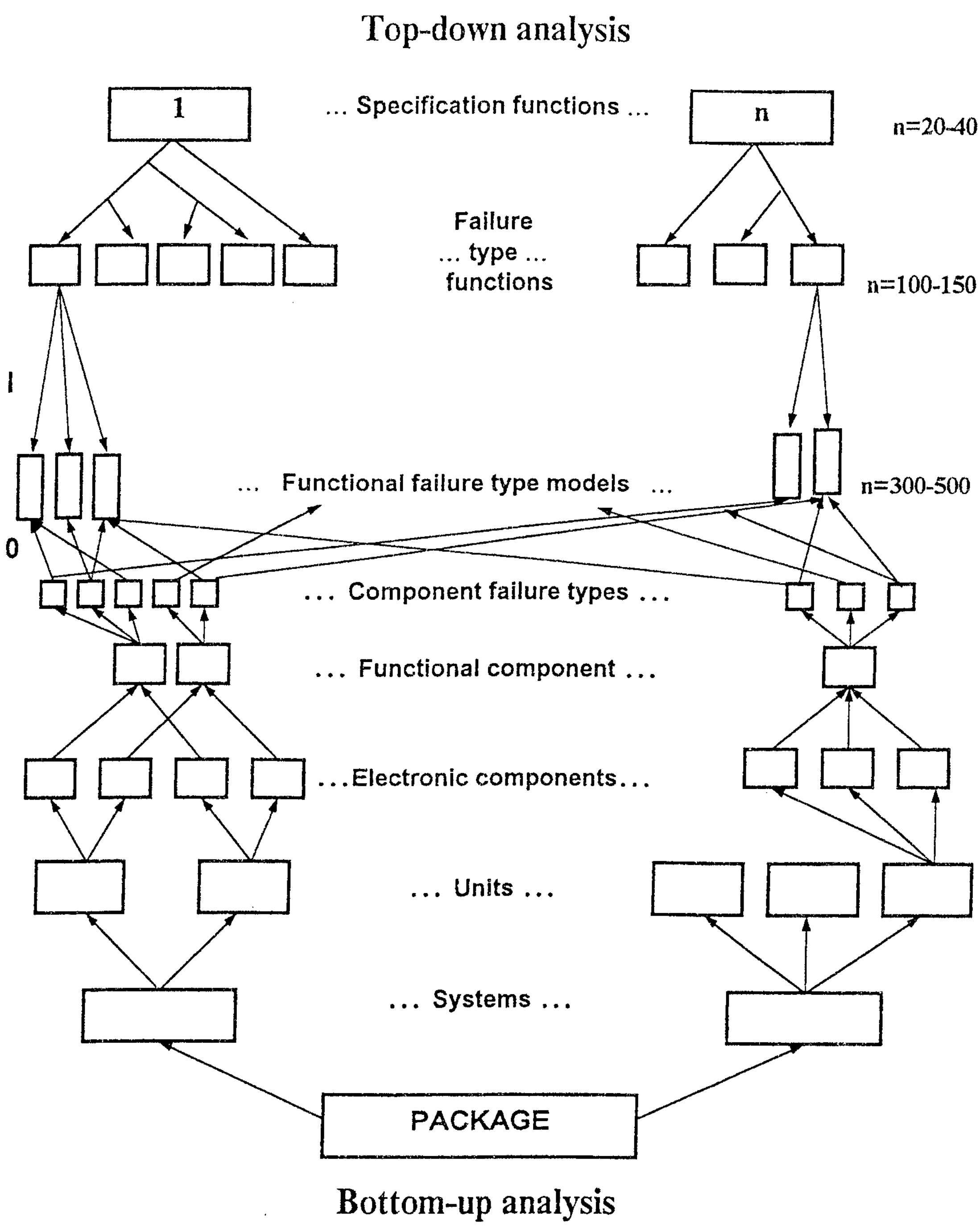
MI 150 - 88

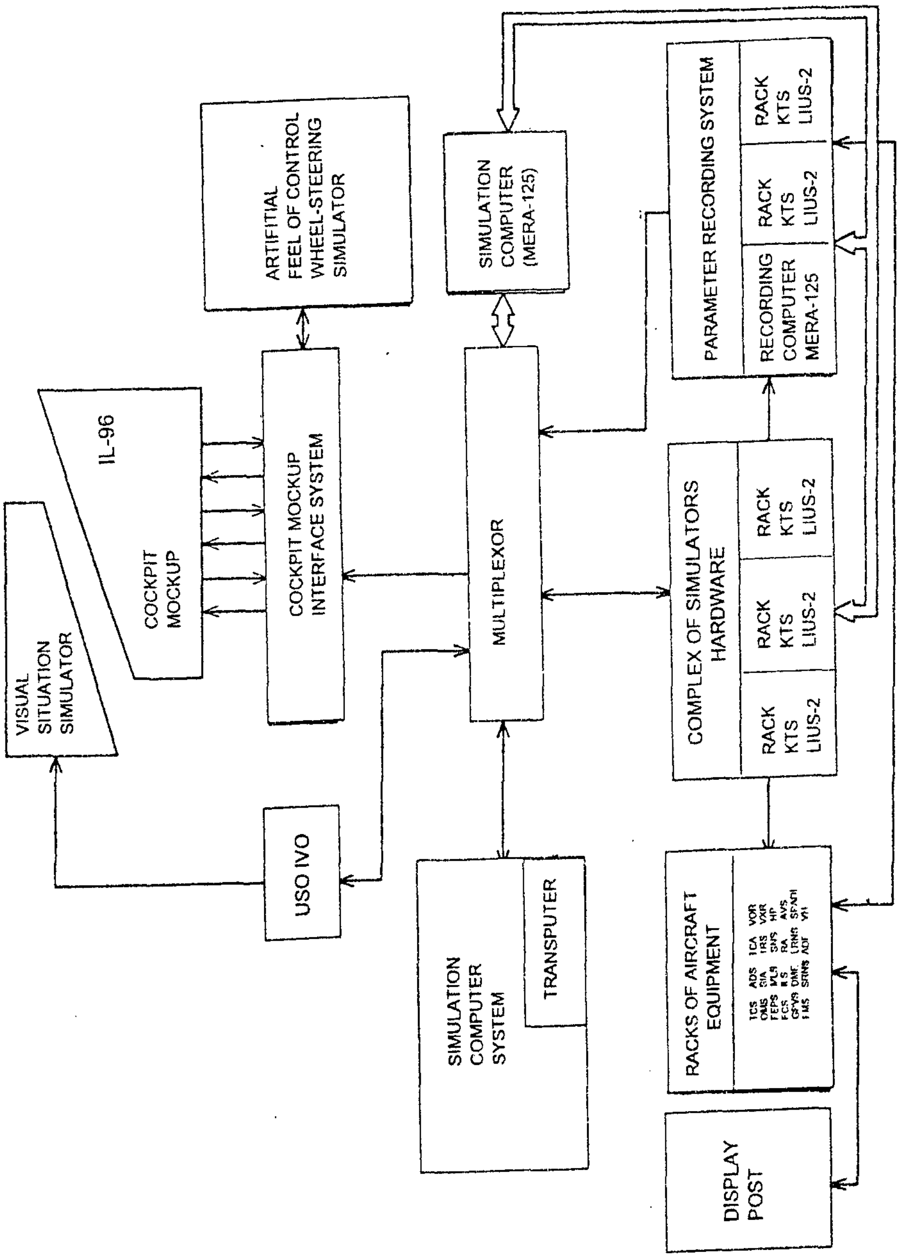
PURPOSE	• detect design, technology and production defects • evaluate the characterishics of reliability and life • determine if it is reasonable to apply technological accomodation
FEATURES	• one test cycle is equivalent to one year of operation • effects are forced within the limits of environmental resistance
IMPLEMENTATION	• simulate the combined environmental exposure and operation • simulate the successive exposure and operation
STAGES	• preliminary tests (prototypes) • qualification tests (lot products) • type tests (lot products)
EFFECTIVENESS	• acceleration by 2-10 times for running time • acceleration by 10-20 times for calendar duration
RESULTS	• over 540 types of aircraft equipments were tested under MI 150-88 • the criterion for convergence of test results and operation data was satisfied with 66%

GENERALIZED DISTRIBUTION OF FAILURES



SCOPE	Methodical Instructions (MI) carry over to electronic and electric equipment
ADDITIONAL INFORMATION	Based on MI 150-88, Methodical instructions for technological accomodation (MI 291-87-111) are devoloped


Top-down analysis
1
... Specification functions ...
n
n=20-40
Failure
... type ...
functions
n=100-150
I
... Functional failure type models ...
n=300-500
0
... Component failure types ...
... Functional component ...
...Electronic components...
... Units ...
... Systems ...
PACKAGE
Bottom-up analysis


VISUAL SITUATION SIMULATOR
COCKPIT MOCKUP
IL-96
COCKPIT MOCKUP INTERFACE SYSTEM
ARTIFITIAL FEEL OF CONTROL WHEEL-STEERING SIMULATOR
SIMULATION COMPUTER (MERA-125)
MULTIPLEXOR
USO IVO
SIMULATION COMPUTER SYSTEM
TRANSPUTER
PARAMETER RECORDING SYSTEM
RECORDING COMPUTER MERA-125
RACK KTS LIUS-2
RACK KTS LIUS-2
COMPLEX OF SIMULATORS HARDWARE
RACK KTS LIUS-2
RACK KTS LIUS-2
RACK KTS LIUS-2
RACKS OF AIRCRAFT EQUIPMENT
DISPLAY POST

QUANTITATIVE FACTORS	SEQUENCE OPERATIONS	OUTPUT
◊ All systems included in the complex	ENGINEERING ANALYSIS OF SYSTEM RELIABILITY	◊ Characteristics of system safety
◊ System failures: 100-200 ◊ Types of system failures: 300-1,000 ◊ System failure conditions: several millions	ANALYSIS OF FUNCTIONAL AND MULTIFUNCTIONAL FAILURES OF COMPLEX	◊ List of functional failures: -failure modes - failure representation ◊ Estimation of failure probabilities ◊ Classification of abnormal situations ◊ Recommendation for crew actions
◊ Types of failures: 30-40	SIMULATION OF FAILURE CONDITIONS	◊ Evaluation of failure effects by testers (pilots) ◊ Revised recommendations for flying to react against failures ◊ List of complex failures to be tested in flight
◊ Number of failures: ~10	IN-FLIGHT CHEEK OF RECOMMENDATIONS FOR FLYING	◊ Revised recommendations for flying with complex failures detected to provide flight safety

FIG. 15 DETERMINATION OF FAIL-SAFETY FOR A FLIGHT CONTROL AND NAVIGATION COMPLEX

SECTION 1

Structure and Strength

DISTINCTIVE POINTS OF RUSSIAN REQUIREMENTS TO AIRPLANE STRUCTURE STRENGTH

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The intention of the Russian aircraft industry to come into the world's markets raises the problem of comprehensive unification of the Russian Civil Airplane Airworthiness Requirements (the NLGS) with similar regulations in effect in other countries: the USA (FAR) and Western Europe (JAR). With this in view, the Russian Aviation Regulations, AP-25, have been developed; these closely correspond to the FAR in respect of composition and contents of most items. However, the AP-25 requirements have a number of notable differences from the FAR that were derived out of the Russian experience in designing the aircraft. These differences are the subject of the report.

Fig. 1 represents schematically the n—V diagram for maneuvers which must be considered in structural analysis. The points I, II, III, and IV here correspond to the complementary provisions included in both the NLGS and AP (and not observed in FAR and JAR):

I point depicts a combination of the maximum negative load factor and the flight speed V_D (FAR specifies the load factor to be zero at the speed V_D);

II point is characterized by a combination of the $1.5g$ acceleration, the speed V_F, and a complete deflection of ailerons for takeoff/landing configurations (FAR does not consider this case);

III point is for a combination of a zero load factor and the speed V_F in takeoff/landing configurations and defines the negative load on a slat (formally, the FAR does not require such situation to be considered);

IV point: a $2g$ acceleration, the speed corresponds to the maximum normal force coefficient of the airplane configured for takeoff or landing (FAR does not consider this case).

Our experience shows that these requirements, as a rule, do not cause notable intensification of loading conditions but can improve flight safety for certain airplanes.

The criteria envisaged in the FAR for determining the loads at the "checked manoeuvre" in the vertical plane seem to be obsolete because they do not allow features of the airplane motion dynamics to be taken into account, including airplanes equipped with automatic control systems. However, instructions in our Civil Aircraft Airworthiness Requirements dealing with pilot efforts during such a maneuver are unnecessarily sophisticated. Therefore the decision has been to adopt the conditions specified in the ICAO Airworthiness Technical Manual; they result in a similar level of loads on the tail unit. As well, such kind of conditions may be seen in JAR.

Design criteria for a maneuver in horizontal plane

Main feature is the fact that the FAR (JAR) propose the rudder to be returned to the neutral position after the angle of sideslip is at its steady-state value, whereas AP considers the maximum angle (attained due to dynamic effects). The latter slightly increases vertical stabilizer loads, the increment being greater for high-altitude flights (see fig. 2). However, mounting a simple damper of yaw motion can reduce the dynamic effects in the angle of sideslip, thus reducing the loads.

It should be noted that, although FAR specifies lesser loads on the vertical tail during a maneuver in the horizontal plane than those in the AP, the FAR comprises an unnecessarily sophisticated procedure:

— if a rigid-body motion of the airplane in yawing is described by more than two degrees of freedom then it is difficult to establish the steady-state sideslip angle. The implicit techniques prescribed for these cases may "spoil" the airplane loading model;

— as for an "actuatorless" control of an airplane in yaw, the FAR requires the pedal force to be kept constant, whereas the AP prescribes maintaining the initial deflection of the pedal. With this, the FAR-obeying analysts should assume the pedal (and the rudder) to deflect to a larger angle as the sideslip angle becomes larger. In this case it is difficult to establish the steady-state angle because the rudder deflection notably varies in time. Such situation prevents application of simplified methods for loads analysis — whereas the yaw motion provided in the AP is easily analyzed with simplified methods. Also, one can see that a ratio of vertical-stabilizer loads from the AP and the FAR for an airplane with "actuatorless" control may be much less than the sideslip angle overshoot.

Unsymmetrical loads after engine failure

FAR and JAR consider failure of any one engine only. AP postulates the four-engined airplane to survive simultaneous or consecutive failures of two engines (depending upon probability of these events) at one side of the airplane; this is in agreement with the requirements written in Section "B" ("Flight") of both the FAR and AP. In this case, FAR (and JAR) assume that the pilot begins to compensate the yawing already when the maximum rate of yaw is achieved, whereas AP assumes these efforts to be applied later, only after the maximum angle of sideslip is attained. Thus, the provisions of AP seem to better cover the situations which are likely to occur in operation. In addition, AP assumes that the loads applied to the airplane after the one-sided engine failure, are considered to be the limit loads irrespective of causes of the failure, while the FAR suggests these loads to be ultimate ones. All Norms incorporate special provisions aimed at minimization of a probability of a catastrophic result after a damage to the airframe from uncontained fragments of an engine; in this light, the assumption of an airframe failed due to setting the airplane at the maximum design angle of sideslip seems to be illogical. Once emergency situation takes place, AP prescribes this factor of safety to be 1.25 (with respect to the maximum loads likely to occur at engine failure) in AP.

As for the ground handling load requirements, the AP-25 FAR-25 (JAR-25) are compatible in many respects. However, two essential differences should be noted. The first of them is to standardizing the values of the vertical speed at landing. As is known, both the FAR and the JAR-25 specify this value independently of parameters of both the airplane and the aerodrome — 3.05 m/sec. According to AP, this speed (which should be used for designing the shock absorbers) is considered a sum of:

- the vertical component which the airplane has at instance of touchdown and
- an additional vertical speed induced by a rolling of airplane landing gear wheels over the uphill of a local bump.

In this case, the first summand (the augend) depends on the airplane only, whereas the second summand (the addend) depends on both a degree of roughness of the airstrip and the airplane landing speed. According to the simplified relation given in the AP, the slope of the local bumps referenced to a 10 — 20 m length is equal to 0.025 for paved airfields, so the design value of the vertical sink speed can range from 2.8 to 3.5 m/sec. Using the results of flight tests on various types of aircraft, a probabilistic analytical model has been developed employing a spectral description of the bumps on airfields. fig. 3 offers the results of the computational analysis by this model for the Il-96 and A-310 airplanes, assuming two spectra of bumps; the parameters cover the range of actual characteristics of civil airfields in the Commonwealth of Independent States. These data (considered as the first approximation at a probability of $2 \cdot 10^{-5}$) confirm the design values of the vertical sink speed and are indicative of great significance of the bump characteristics.

The second feature of the AP-25 as compared to the FAR in concepts for ground handling loads is in specifying the requirements with respect to landing gear loads for a takeoff ground run. As is known, Section 25.491 of the FAR contains general provisions only. The AP comprises relations for determining the loads caused by a takeoff run on a paved airfield; the formulae are close to those in CCIO-1, an annex to JAR-25.

Much attention was paid by the NLGS to dynamic application of load during bumpy-air flights. Unlike FAR, we consider both the discrete air bumps and the continuous atmospheric turbulence. For the latter case, use is made of the envelope. In the two cases, loads should be referred to a standardized load factor for the airplane center of gravity. Namely, gustiness for an elastic airplane should be assumed such that the center of gravity of an elastic airplane were subjected to the prescribed load factor. This procedure relates definitely the requirements for rigid and elastic airframes. However, the procedure is unacceptable for an airplane with an active loads control system because the simplified equation is not capable of accounting for features of this system.

In addition, a statistical analysis of the in-service data on repetition rates of load factors on domestic transport airplanes (fig. 4) performed over the total amount of more than 1.3 million flight hours indicates the following. The ratio of the load factor expected during 50.000 flight hours in total for an airplane (the probability of $2 \cdot 10^{-5}$, as is normally assumed for determining the limit loads) to the design value required by the NLGS is 0.61 in the mean, rather than 0.67 as the safety factor of 1.5 dictates. Thus, there is a basis for:

— decreasing by 10% the limit values of the load factor accounting for the gust effects,
— decreasing the gust intensity values by 12 – 16%, and
— getting nearer to the requirements of JAR.

In this connection, and to match the regulation of many countries, AP-25 includes the JAR requirement concerning with dynamic loading due to a discrete gust — the tuned gust. Although the magnitude of the tuned gust is less than that in the NLGS, a frequency of the tuned gust may in some cases be adjusted so that the dynamic loading can sometimes be pronounced more clearly than due to the NLGS-prescribed gust.

Freedom from aeroelastic phenomena

Requirements of AP with respect to flutter, divergence, reversal of controls are close to those in the FAR. However, the AP incorporates an additional item (not seen in the FAR): the aeroelastic stability of an airplane with an automatic control system (ACS) must be demonstrated. The challenge is known to consist in the danger of aerodynamic and inertia-related interaction of the airframe and the control surfaces (and, in some cases, danger of interaction of the mechanical control linkage). The AP requires the margins to be provided in both the absolute value and the phase of the frequency response function of the "airplane — ACS" system open loop; these margins must be demonstrated by analyses, by ground tests on a full-scale airplane, and, in certain cases, by flight tests.

Also, AP incorporates special requirements with respect to freedom from shimmy which is not present in the FAR. The safety must be checked by analyses, by tests on impact testing machines, and/or by flight tests.

The AP-25 requirements with respect to airframe fatigue resistance are almost identical to FAR requirements — except for the safe life time determination approach that corresponds more closely to Russian practice. Differences and features in showing the airframe service life capability are reflected in Means of Compliance documents: MOS for the AP, AC for the FAR.

In particular, MOS requires the full-size structure to be tested for fatigue life capability and damage tolerance; MOS (unlike AC) contains a set of reliability factors to be involved to extrapolate the test results for specifying:

- the inspection threshold and the inspection interval, as well as;
- the safe life time in compliance with endurance requirements.

One of major differences between MOS and AC is seen in the methodology of systematic monitoring of safety with respect to fatigue damage accumulation in service. MOS stipulates a stage-by-stage specification of currently allowable service intervals. Before regular operations an initial allowable service interval is established. As time goes on, data from the most intensely operated airplanes in the entire fleet are used to establish the next interval; account is taken of the maximum design service duration, but airframes may be used beyond the design-stage estimates. A typical length of a service stage is 5 to 10 thousand flight hours. Each new ("prolonged") stage duration is specified on the basis of:

- refinement of mission types and operating conditions in the fleet;

— refinement, where necessary, of load levels for some airframe parts by means of special flight tests;
— accumulation of statistical data (from on-board recorders) on relative frequencies of center-of-gravity load factors;
— results of additional laboratory tests for endurance and damage tolerance (including testing of structures flown for a long time);
— experience in operating the airplanes of the particular type.

Summarizing the abovesaid, we can conclude that the new Russian airframe strength standards, AP-25, are close to those of FAR in both the principles and the essence. However, certain provisions follow the more clear and partly more stringent items of the former USSR Civil Aircraft Airworthiness Requirements, the NLGS.

With this, the FAR's requirements are met almost completely; simultaneously, the airworthiness/safety levels characteristic of the previous Russian airframe strength standards are kept.

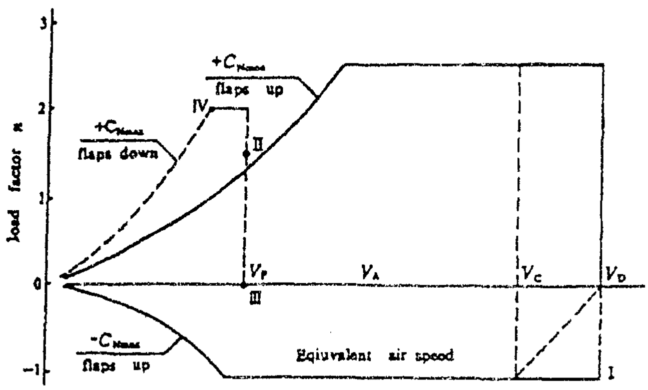

Fig. 1 n-v diagram for maneuvers of structural analysis

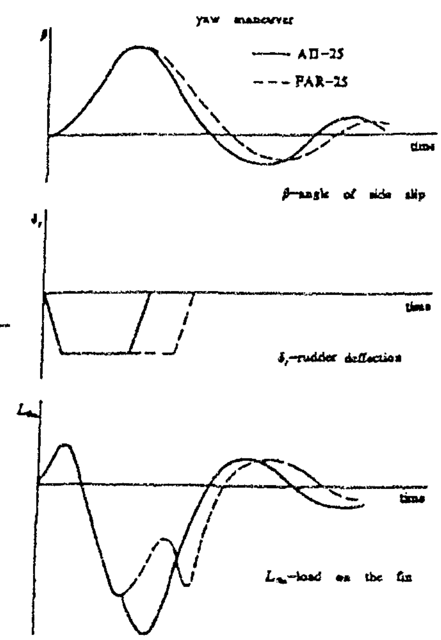

Fig. 2 Influence of dynamic affects on vertical stabilizer loads

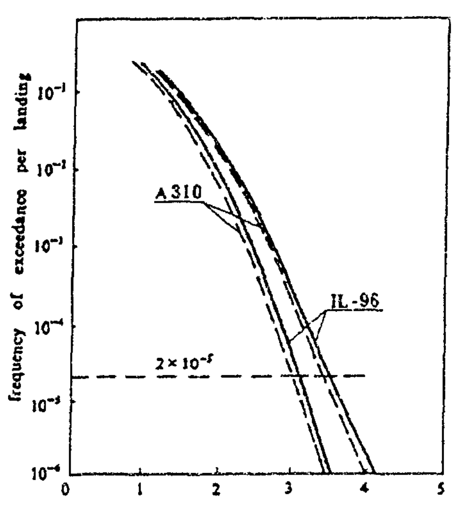

Fig. 3 Results of computational analysis for the IL-96 and A310 Airplanes at two spectra of bumps

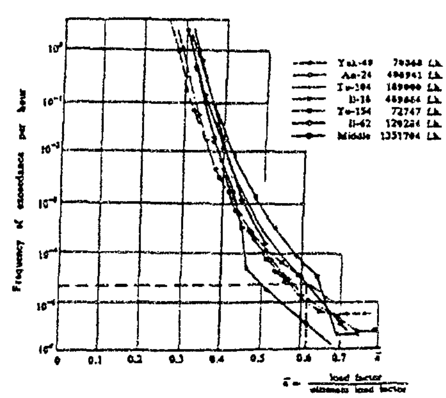

Fig. 4 Statistical analysis of repetition rate of load factors on former Soviet transport airplanes performed over the total amount of over 1.3 million flight hours

THE UP-TO-DATE PROBLEMS OF AEROELASTIC EXPERIMENTAL RESEARCH

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TsAGI, Zhukovsky, Russia

Solving aeroelasticity problems is essential part of certification procedures in civil aviation with respect to structural strength. There are two major objectives of solving these problems. One is to issue recommendations for optimal structure configuration. The other one is to confirm the existence of appropriate aeroelastic stability safety margins according to Airworthiness Requirements.

The following fields of investigations can be considered within the framework of general aeroelasticity problems:
- flutter, buffeting, vibrations at transonic speeds;
- aeroservoelasticity, i.e. coupling between structural aeroelastic vibrations and operating control system;
- static aeroelasticity, including aileron reversal, divergence, control surface efficiency etc.;
- shimmy of landing gears.

Comprehensive experimental research is needed to solve the problem successfully. The basic type of investigations is modeling of aeroelastic phenomena in wind tunnels. Results are used either to understand the physical nature of considered aeroelastic phenomena and to correct analytical models or to obtain data related to particular aircraft type. According to our experience of many years, such investigations are considered as the foundation in issuing formal conclusions regarding aeroelastic stability.

Experimental investigations include:
- design and manufacture of dynamically and elastically scaled models for wind tunnels;
- ground vibration testing (GVT) and stiffness measurements of models and full-size airplanes;
- experimental measurements of frequency response functions (FRFs) of control systems and their elements at test benches and GVT of full size aircrafts with operating control system;
- model tests in wind tunnels;
- ground vibration and dropping tests of landing gears as well as shimmy tests on rotating drum.

The state-of-the-art in solving these problems is briefly described below in the present paper.

FLUTTER

The experience of many years of flutter investigations has resulted to date in creation of complete system of experimental research and computational analysis, which makes it possible to obtain the values of critical flight parameters with appropriate accuracy and to issue recommendations for structural optimization to increase critical flutter speed. Lately, the major TsAGI efforts has been concentrated in the following directions:

1. Progressive development of analytical methods intended not only to get higher accuracy of results but also to reduce the scope of expensive experimental research.

The main items here relates to usage of update computers, complicated analytical computational models, knowledge from the previous experience, integration of tests and analysis. Mathematical models are specified step by step at the basis of experimental results, while tests are conducted for most critical configurations derived from analysis.

2. Upgrading of design and manufacturing procedures of models for wind tunnel tests and broad usage of composite materials with high elastic moduli for models of this kind.

To simulate the full-size structure properties as better as possible remains the main goal in manufacturing the models. This is provided for by means of intermediate verification tests (GVT, stiffness and mass measurements) of models and their parts at the stage of manufacture. The obtained results are used to correct analytical models, and this allows one, from one hand, to find out possible discrepancies between model for tunnel tests and full-size airplane and to introduce changes into the model structure and, from the other hand, to plan wind tunnel tests in advance.

To reduce the number of tested models is also a very important item especially for design bureaus under the lack of material provisions. Two ways may be considered to solve the problem. The first one is to use the same model for different tasks. For example, flutter model may be taken for aeroservoelastic and static aeroelastic investigations as well as for buffeting tests. The second way is to reduce the number of models used for studying the Mach number influence on flutter (it is impossible to simulate explicitly all aeroelastically-scaled flight envelope in transonic wind tunnel using one model only). The solution of the problem may be based only on integrated techniques combining experiments and analysis, the last being based on well verified algorithms that give reliable results for any flight parameters, including transonic ones.

3. Improvement of experimental methods used in wind tunnels.

The main goals are to eliminate the risk of model failure at flutter onset and to increase the amount of information obtained in tunnel tests. Eliminating the risk of model failure results not only in direct economical benefits but also in implicit ones as it enables to employ less money-consuming fan-driven tunnels of continuous action type for flutter tests, which was impossible to do using traditional experimental technique of direct achieving flutter boundary.

"Non-failure" test mode has required to develop special methods for flutter boundary prediction based on tests at speeds below flutter, to use means for on-line model vibration state monitoring and special devices for model clamping on account of unexpected flutter onset.

The modern data acquisition and processing systems that can employ a lot of gauges and control shakers to excite models are used to increase the amount of information gained from experiment. The most difficult matter corresponds to processing the data disturbed by noise, reliability of math techniques for solving the problems of identification and flutter boundary prediction, development of test equipment to provide for model vibration excitation under flow conditions.

Thus, the above-described integrated methods of flutter research, completing together all the works to provide for the aircraft safety and its certification, has been created and still under way in TsAGI Division of Aeroelasticity. The flutter problems are described more in details in the paper by S.Paryshev and others in the present Proceedings.

AEROSERVOELASTICITY

All the phenomena related to influence of flight control system on aircraft aeroelastic performance are the matter of aeroservo-elasticity. The cooperation of experts in various fields, i.e in aero-elasticity and flight dynamics is needed to get success, as aero-servoelastity is obviously the combined complex problem. Regarding the subject, one can say about the scientific school uniting experts engaged in development of math models, software, experimental techniques and methodology of solving pertinent problems for flying vehicles of various types.

The following fields of investigations should be noted:

— providing for the aerosrvoelastic stability;
— evaluation of flight control system influence on aircraft structure loading and vibrations;
— development of special systems of active control to alleviate loading, increase vibration mode damping and flutter speeds and enhance flight comfort for passengers.

Investigations and flight practice evidence that dangerous self-excited oscillations can occur under unfavorable combination of structural inertial, stiffness and aerodynamic characteristics and control system performance. Sometimes an on-ground dynamic instability may be caused by pure interaction of inertia loads and control system performance even at zero flow speeds.

The frequency response method has proved to be the most pertinent for use in aeroservoelastic research. The method is based on consideration of frequency response functions of aircraft elastic structure and control systems. The major advantage here is the possibility of obtaining FRFs of structure and control system independently from each other and either by means of experiments or analysis. Then any of these FRFs may be used to obtain FRFs of open-loop system for any control channel to get the conclusion about dynamic stability.

Therefore the principal goal of any experimental and analytical investigations within the problem is to derive reliable FRFs.

The following types of testing are used:
— Physical modeling;
— GVT of full-size airframes, their elements and control systems;
— Flight tests.

The problem of modeling of aeroservoelastic phenomena in wind tunnels may be solved as follows:

Dynamically scaled (probably for flutter tests) models equipped with fast-driven control surfaces, small-size actuators, part of mechanical control circuit, different types of gauges are used. The control circuit is made in such a way that it does not affect model stiffness. The movable laboratory with special equipment is used to simulate the on-board part of control system intended to process control signals. Gust loading may be simulated by cascade of controllable moving profiles installed at the beginning of tunnel working section.

The FRFs of analog/digital control systems within the frequency range of major vibration modes are obtained during GVT of full-size aircrafts with operating control system. The special test rigs are used to realize the prescribed oscillations of control system sensors for the purpose of determining control system FRFs considering control surfaces deflection as output and sensors oscillations as input.

The following tasks are carried out in flight tests:
— measurement of aircraft structure FRFs under flight conditions regarding control surfaces deflections as input and control system sensors signals as outputs;
— confirmation of aeroelastic safety margin stability.
— evaluation of efficiency of active loads alleviation and vibration modes damping systems.

STATIC AEROELASTICITY

Experimental investigations are the obligatory part of works carried out to ensure the safety with respect to static aeroelastic phenomena such as controls reversal, divergence and static deformations caused by aerodynamic loading.

The tests are conducted at models of various types — full elastically scaled models, semi(half)-models and cantilever models. The following kinds of model internal structures may be used:

- simplified elastically scaled models for which stiffness properties are simulated by means of internal frame consisted of beams or plates made of metals or composites, and which outward aerodynamic shapes are provided for by covers made of materials with low elastic moduli;
- structural copies with scaling of main full-size structural elements such as spars, ribs, skin etc.

Sometimes the models designed for flutter tests may be used also for tests on static aeroelasticity.

At present, simplified elastically scaled models has gained most often usage. The reason is that they meet the principal requirements on models for aerodynamic tests as they have high accuracy of skin surface, sufficient strength safety factor and enable to install various in-model instrumentation. Models parameters are determined by means of specially developed software. Models are mounted at six-component strain gauge balance via special duplicated sting providing for the minimum of model aft-part airframe and outward geometry distortion. Model angle of attack may be changed by means of remote control. Pertinent instrumentation is installed to investigate the influence of structural elasticity on total and distributed aerodynamic loads and hinge moments and to obtain data on in-flow model deformations.

The testy results are used for verification of relevant analytical models at the stage of preparation of formal conclusions on aircraft safety with respect to controls reversal and divergence.

Models-copies are utilized only for aircrafts with complicated unusual configurations and airframe structural layouts that are hard to be subjected to computational analysis and simulating by means of simplified models.

At present, TsAGI Division of Aeroelasticity concerns with development of unified approach for creating the models made of composites for any kind of tests (pure aerodynamic, static aeroelastic and flutter) in wind tunnels at the basis of using the same moulds. This promises to have more operative, profitable and accurate ways to make models that are necessary for creation of modern and advanced aircrafts.

SHIMMY OF LANDING GEARS

The integrated methodology for development of shimmy-free landing gears has been developed by middle eighties. Within this, the computational analysis based on different math models is the most important matter. The FRFs of landing gears determined in GVT of gears at test benches as well in GVT of the whole full-size aircraft are used to tune the math models. The essential nonlinear features of stiffness and damping properties of landing gears should be taken into account in developing the test techniques. The tests are conducted under conditions of landing gear loading as the last essentially affect the mechanical properties of landing gears. Various means are used for the loading. At present, tests of heavy aircrafts are performed with the aid of special pneumatic supports and light aircrafts are tested with the aid of pneumo-static supports.

At the first stage of development prior to manufacture of full-size actual landing gear, the reliable evaluation of safety with respect to shimmy may be obtained, if parameters of math model are determined from the tests on rotating drums or from GVT-derived FRFs of tyres. At this time mechanical modeling by means of dynamically-scaled model of landing gear is proposed to be conducted.

At the second stage, hese are full-size landing gears that are mainly concerned with and tested at laboratory test facilities, during GVT of full-size airplane prototypes and in flight (take-off/landing runs). Nevertheless, computational analysis remains the principal matter, and the system "full-size landing gear + test rig " is subjected first to such an analysis. As the conditions for rolling of wheels at laboratory tests may essentially differ from those at actual aircraft operating, these test results are necessary primarily to adjust the math model of shimmy. The evaluation of various anti-shimmy means may be also done during laboratory tests.

The accuracy of modeling of stiffness of cantilever clamping of gear strut in laboratory tests can be assessed on the base of special GVT of landing gear installed at the full-size aircraft. The 15 % difference between test facility clamping stiffness and fuselage/strut interface stiffness is acceptable.

The flight tests of prototype is of great importance in providing for the safety with respect to shimmy. The flight test results are used to confirm that landing gear is shimmy-free and to evaluate actual safety margins in terms of damping. The special means for vibration excitation of wheels and appropriate data processing systems are needed.

GROUND VIBRATION TESTING

The ground vibration testing of models, actual airframes, control systems and landing gears is especially important constituent of experimental aeroelastic investigations. These are the results of such tests that allow one to tune and specify analytical models and determine the degree of compliance between actual structures and tested models.

TsAGI Division of Aeroelasticity and Structural Dynamics has gained the unique experience, methods and modern facilities for GVT of flying vehicles and control systems.

The available means make it possible to obtain natural vibration mode characteristics (eigenfrequencies, damping, mode shapes, generalized masses) of various complicated airframe structures quickly and with high quality.

The following performance data are determined from GVT of articles with operating control systems:

- FRFs for different control channels;
- safety margins of closed-loop system "structure + control system";
- validation of on-board control system and computers operating under conditions of vibrations.

The special movable test laboratories with data processing and excitation systems, computers and software as well as facilities for "soft" suspension of tested articles are available for GVT of structures of any kind with no restriction of mass at any place designated by a customer.

Finally speaking, these are all the means mentioned above in this article that enable to provide for aeroelasticity-related investigations and certification procedures at high level for any aircraft types.

moveable support and also special flight tests with air structure oscillations.

In **the aircraft loads division** are determined the loads on the all airframe units in accordance with the airworthiness standards requirements, airplane units and landing gears load spectrums (static and dynamic loads) during flight according to the flight envelope and also structure service life tests programs forming. One of the important department tasks is control over real service conditions, life time expending rate and equivalents between fatigue life tests programs and real loading conditions.

In their work the division's collaborators use load analysis different methods beginning with empirical methods to recount from analogy and using experimental data on loads (weight and drained models wind tunnel tests, and loads measured in flight tests), and finishing with oriented on modern computers mathematical models (linear and non-linear steady aerodynamics, static aeroelasticity, spectrum approach and non-steady aerodynamics at the analysis of elastic aircraft movement through turbulent atmosphere and runway roughness).

In the division there are special algorithms and programs of data obtained from objective control tools (MCPП, K3-63) for aircraft being in service and static analysis of data on cracks and damages in service.

Generally design-experimental department is a united collective of highly skilled specialists capable to decide any tasks of aircraft strength ensuring on modern scientific and technical level.

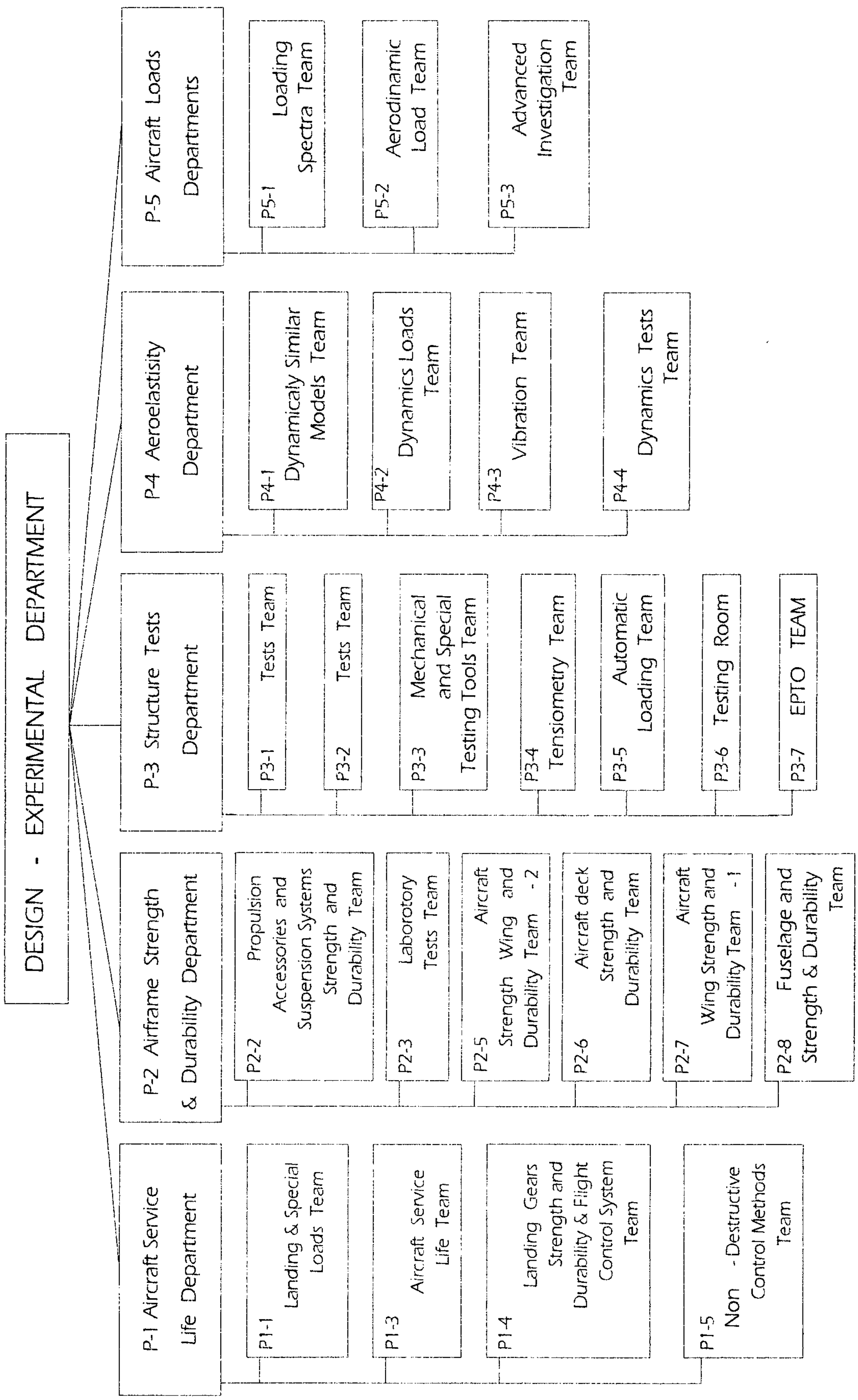
DESIGN - EXPERIMENTAL DEPARTMENT
P-1 Aircraft Service Life Department
P1-1 Landing & Special Loads Team
P1-3 Aircraft Service Life Team
P1-4 Landing Gears Strength and Durability & Flight Control System Team
P1-5 Non -Destructive Control Methods Team
P-2 Airframe Strength & Durability Department
P2-2 Propulsion Accessories and Suspension Systems Strength and Durability Team
P2-3 Laborotory Tests Team
P2-5 Aircraft Strength Wing and Durability Team -2
P2-6 Aircraft deck Strength and Durability Team
P2-7 Aircraft Wing Strength and Durability Team -1
P2-8 Fuselage and Strength & Durability Team
P-3 Structure Tests Department
P3-1 Tests Team
P3-2 Tests Team
P3-3 Mechanical and Special Testing Tools Team
P3-4 Tensiometry Team
P3-5 Automatic Loading Team
P3-6 Testing Room
P3-7 EPTO TEAM
P-4 Aeroelastisity Department
P4-1 Dynamicaly Similar Models Team
P4-2 Dynamics Loads Team
P4-3 Vibration Team
P4-4 Dynamics Tests Team
P-5 Aircraft Loads Departments
P5-1 Loading Spectra Team
P5-2 Aerodinamic Load Team
P5-3 Advanced Investigation Team

Figure 2.

ALTERNATIVE WAYS OF AIRCRAFT STRUCTURAL TESTING

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Fatigue and structural tests of full-scale aircraft provide the most veritable and expensive information on aircraft strength characteristics and potentialities of their improvement. At the same time, this kind of testing is the most labour-consuming and costly stage for obtaining strength and service life performance of a new plane.

Intention to make structural tests more effective compels us to revise the traditional sequence of conducting such tests with due consideration of production rates, flight test schedule and, surely, of designer's financial status. Usually,the normal order of testing is like the following:

First prototype: flying;
Second prototype: for static tests;
Third and fourth aircraft: for flight tests;
Fifth aircraft: for fatigue tests.

Duration of flight test period is about 2 or 3 years.
Duration of static tests: one year upon delivery of an aircraft.

Rate of fatigue tests: N 40,000 of laboratory flights per year for smaller medium aircraft, and N 20,000 laboratory flights per year for heavy aircraft. Thus, two full-scale prototypes are taken out from the pilot batch of aircraft. Later developments that arise out of static and flight test results are not incorporated into the aircraft structure dedicated to fatigue testing, peculiarities of serial fabrication process are not considered, too. In fact, damage tolerance is being left out of investigation scope even when the certification commences. So grave design blunders, if any, are being discovered late enough.

Modifications and developments are introduced to series aircraft behind time and without due experimental checks. Moreover,if the 'fatigue' prototype is delivered last in the pilot batch,the fatigue tests will delay the beginning of certification until the total of operation hours reaches at least one service life period. Along with time spendings there are considerable financial expenses, i.e., construction of two full-size aircraft for laboratory tests, design and fabrication of two test stands in the period preceding certification require big investments in short periods.

To get rid the above-mentioned difficulties, an alternative way of conducting structural tests is suggested. The main of the suggestion lies in combination of fatigue tests (within the scope of one design service life) and static tests in one, i.e. in the first (sic!) structural prototype (ref. fig. 2). Data obtained in these tests should be used for the aircraft certification. Another prototype dedicated to fatigue testing should be later picked out of serial in the early stage of their operation.

The suggested sequence of structural tests has the following advantages:

1. Based on fatigue test results, any further structural modifications become possible in the beginning of aircraft series production, thus eliminating modifications and alternations of already manufactured serial aircraft, deleting release of additional inspection bulletins.
2. Conducting static tests after fatigue tests permits to investigate thoroughly structure's inspectability and to assess the structure's strength with maximum noninspectable defects. The static tests become more representative as compared with the case when a completely new aircraft is tested.
3. Fatigue testing of the second prototype is also more representative since tested is an aircraft produced according to series fabrication methods. Experience of operation preceding tests, and accumulated data on loading permit to sufficiently specify the test program.
4. Material costs of the project development stage are reduced (cost of the second prototype manufacture, cost of fabrication of the second stand, cost of workshop floor space, facilities and equipment, labour cost of the second prototype testing personnel).
5. An opportunity for earlier flight testing of an additional aircraft appears.

The sequence of conducting combined structural tests may correspond to that illustrated in tab. 1.

Let us review in brief the content of jobs for each stage of combined tests taking into consideration the experience gained at Antonov ASTC.

Fatigue testing is accomplished at the first stage. The scope of tests depends on flight test rate. This is so because at certain time the static loading test (stage II) becomes necessary in order to eliminate flight test limitations. It should be marked in this connection that the standard transport aircraft flight test program 75 per cent of flights having loads not exceeding 50 per cent of normalized limit (or 47 per cent ultimate) loads.

Table 1. Stages of Alternative Way of Testing

Stage	Description
I	Fatigue tests
II	Loading up to P^{limit} or to $P^{ultimate}$, in full scope
III	Notches and investigation of fatigue crack growth
IV	Loading up to P^{limit} (residual strength)
V	Repairs

At the same time, the fatigue testing program includes a number of rough flights when loads can attain limit values. This fact may be used for stage canceling of flight test limitations. In our case the period to cancel limitations and acknowledge static tests necessity was a little more than a year and we actually managed to accumulate two design service lives.

At the third stage, in addition to fatigue cracks which had appeared, about 40 notches sized 0.5 mm to 1 mm were made in principal structural elements. Some notches were duplicated and made (up to 10 mm) so that they could generate crack growth in a real structure. During the following operating period we succeeded in growing fatigue cracks close to design critical lengths. The total fatigue failure time exceeded three design service lives.

Residual strength tests were performed at he fourth stage. Loads were applied several times. To avoid interference, major failures were sequentially covered by repairs. Finally, the following important failures were checked:

- completely destroyed lower spar cap of the boundary spar;
- crack in spar web;
- destroyed edge of manhole;
- double-span crack of the lower panel beneath destroyed stringer, etc.

Timeliness of tests depends greatly upon manufacturer's performance. Manufacturer's performance quality was checked at the fifth stage. The structure inspections,using and perfecting NDT methods, were thoroughly performed before repairs. A decision not to repair cracks of the maximum nondetectable size before static tests was taken for the first time in practice. The tests confirmed the calculation accuracy: The maximum nondetectable size cracks had not decreased the wing strength below design values.

Effectiveness of the suggested test method depends upon a number of factors, such as: prototype production rate, design service life, timeliness of repair, overhauls, etc.

In any case, good preparation will give substantial benifits in solving technical problems besides reduction of expenses and time.

For some unique aircraft type, e.g., those with short service live, combined testing of only one prototype can be sufficient.

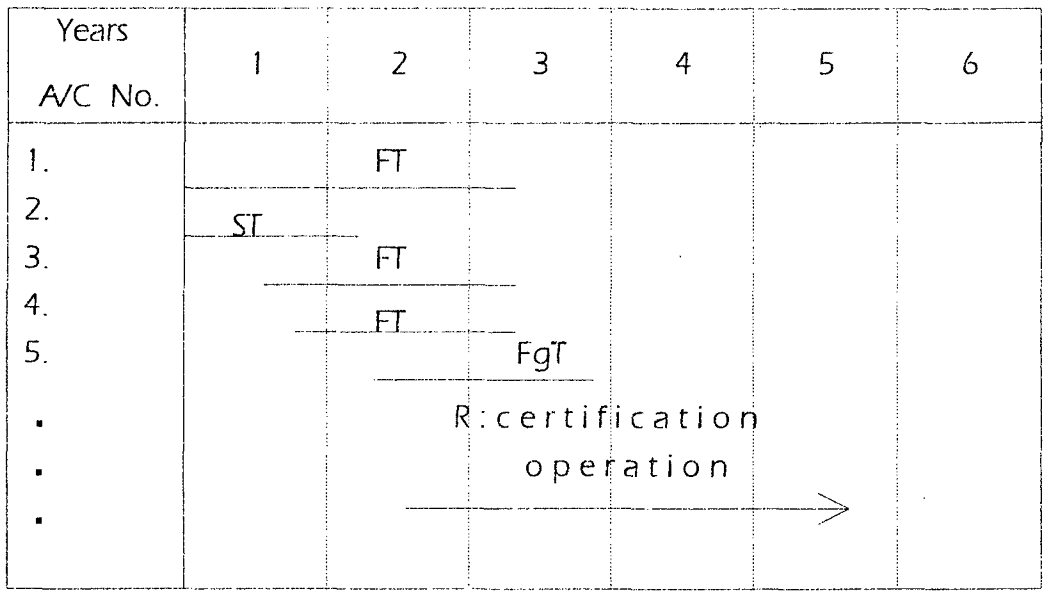

Figure 1. Conventional Order of Tests

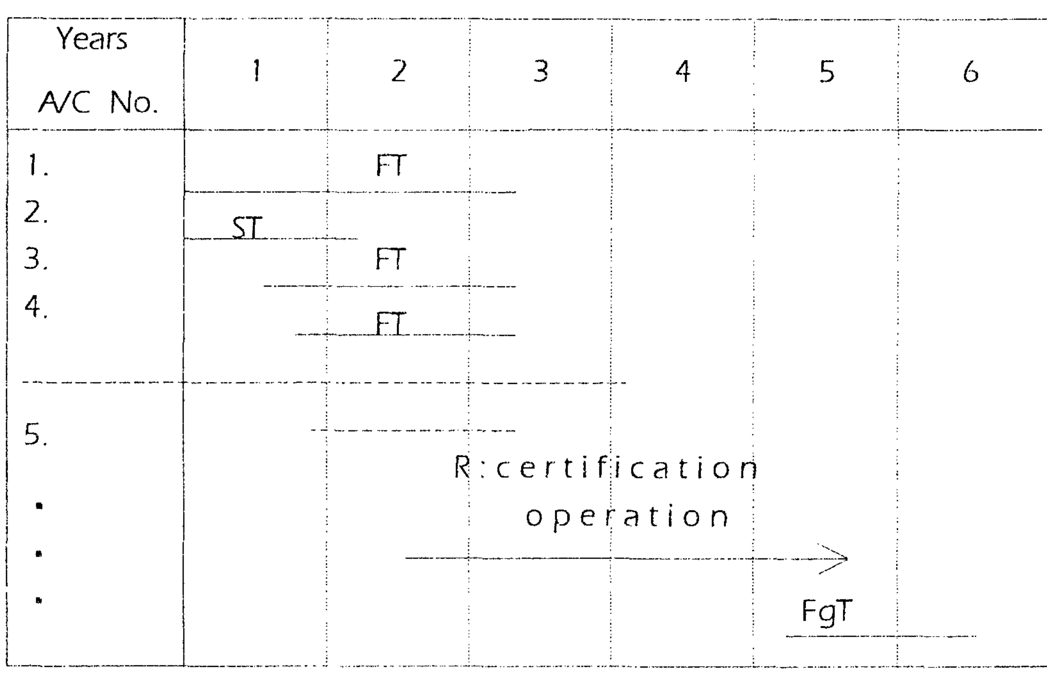

Figure 2. Alternative Way of Testing

METHODS OF NON-DESTRUCTIVE FLUTTER INVESTIGATIONS IN TRANSONIC WIND TUNNELS USING NEW GENERATION OF DYNAMICALLY SCALED MODELS

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Traditionally, flutter tests at transonic speeds in TsAGI have been conducted in T-109 wind tunnel of blow down action type. Within the framework of such a kind of tests, flutter boundary is usually determined directly, i.e. tunnel flow speed is increased untill its critical flutter value is achieved and flutter onset occurs. High risk of expensive model failure due to flutter is inherent to this techniques, though model destruction does not cause sufficient damage of tunnel of such blow down type.

Short tunnel run duration and limited possibility of arbitrary control of dynamic pressure and Mach number in these wind tunnels restrict usage of advanced test techniques including model excitation by means of shakers and actuators, measurements of frequency response functions (FRFs) and determination of in-flow vibration mode frequencies, damping factors and shapes. The outputs of traditional tests are usually turbulence-induced time domain oscillation waveforms which are hard to be compared with analytical data. The second reason of limited implementation of advanced techniques earlier was the lack of appropriate data acquisition and processing systems.

Meanwhile the modern test equipment as well as relevant software and mathematics algorithms have been progressively developed lately to set up the task of actual implementation of advanced approaches. This demands to master wind tunnels of continuos action type for flutter tests.

However, usage of continuos type tunnels set up the very high requirements on flutter tests safety as the broken parts of model failed at flutter can easily damage tunnel compressor blades. Therefore direct achieving flutter boundary is prohibited. Thus, flutter prediction methods, in which flutter boundary is evaluated on the base of subcritical measurements, are needed.

This paper presents some examples of results obtained in flutter tests in T-106 transonic wind tunnel with the aid of advanced methods. The tunnel has been chosen among others for its relatively low run costs and rather simple way of tunnel control.

The second important issue is new materials usage. These are composite materials that are mostly pertinent to be used for transonic flutter models because of their high specific stiffness (elastic modulus / mass density ratio) which is higher than that of any pure metals or other artificial plastics used for models previously. High specific stiffness allow to make model less heavy and meet the criterion of model mass similarity better.

Pertinent new technology has also been implemented to ensure high quality of model design and manufacture. Computer-aided design hardware and software are used to get structural layout, full-scale drawings and templates, composite lamina tailoring, mass/stiffness actual distribution as an input for flutter computer analysis.

Various model structure types within the range from exact copies of full-scale structure to use of high aspect ratio beam approximations with internal beams-spars can be considered. Space between structural elements are usually filled with foam. Models are covered with smooth skin made of composites.

Precision moulds made by means of computer aided design facilities are used for model manufacture. Model main part structure assembly is integrated with moulding of model surface. Thus, required mass/stiffness distribution is achieved together with high accuracy of model surface as for models designed for aerodynamic tests.

Models are verified at all stages of creation. Separated internal elements (spars), assembled model main parts (wing, body, etc.), partial assemblies (e.g. fin+stabiliser), full model assemblies are subjected to stiffness measurements and ground vibration tests providing appropriate possible current corrections in design, manufacture and analytical structural dynamic model. Introduction of measured stiffness of joints appears to be of great importance to create correct analytical model predicting ground vibration modes quite well even for complicated aircraft/model configurations.

An approximate test equipment layout in T-106 wind tunnel is shown in fig. 1. It should be noted that flutter tests in this tunnel has required essential change in usual tunnel run procedures to match specific flutter tests requirements. The tunnel data acquisition and control system software have been updated to provide for continuos recording of main flow parameters with high time resolution and to synchronize it with tunnel test records. Appropriate development of model support and suspension system for tunnel working section has been fulfilled.

The flutter test data acquisition and processing system shown in fig. 1 may be divided into two subsystems. The "fast" one is designed to ensure test continuos safety. The spectroanalyzer provides for visual on-line presentation of spectral densities. Approaching to flutter boundary results in rapid increase of flutter mode resonance peak amplitude. Thus it appears possible for test engineer to approach to flutter boundary too close to estimate flutter speed (Mach number) reliably, but too far to get flutter onset. For emergency, the special "flutter indicator" is used to stop the tunnel as soon as amplitude of some of gauge outputs exceeds prior designated value.

The "slow" system is designed for post-run data processing. Either turbulence -induced model vibrations and corresponding spectral densities may be analyzed or the model may be excited by means of shakers or actuators of different kinds, and accordingly FRFs are considered. The primary analysis may be carried out at spectroanalyzer to get spectra, FRFs or waveforms samples. The secondary processing can be performed at PC computer by means of special software including those created in TsAGI to obtain in-flow vibration mode characteristics, i.e. eigenfrequencies, damping factors and mode shapes. The software allow to deal with musti-degree-of freedom systems and closely located modes.

An example of spectral densities within the range from zero to 100 Hz for a gauge output in one of the tests are shown in fig. 2. One can see two coupled resonance peaks moving closer to each other as Mach number increases. As soon as Mach number exceeds 0.8, the amplitude of the first resonance peak drastically grows up indicating the flutter boundary is getting closer. Some other vibration modes can also be observed, but they are highly damped and their resonance peaks are low and wide.

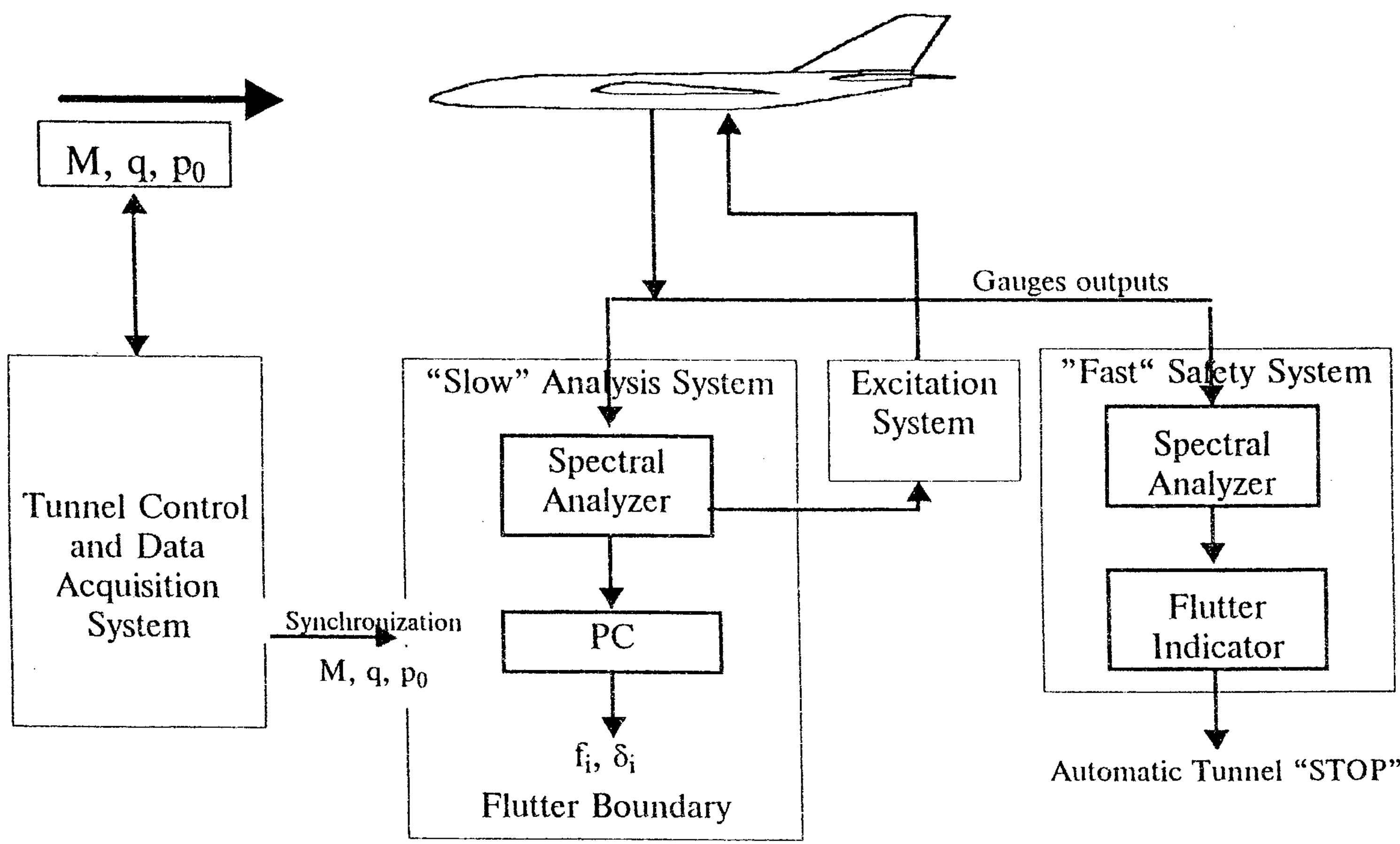
Results of subsequent processing are shown in fig. 3. The two modes frequencies and damping factors in terms of logarithmic decrements versus Mach number are depicted. The frequencies are getting closer. The decrement of the second mode begins growing up since Mach number exceeds 0.8 while that of the first one rapidly falls down. Continuing the decrement curve up to the intersection with zero line, one can get the critical flutter Mach number.

The results above are relevant to simultaneously increasing dynamic pressure and Mach number as tunnel compressor speed grows up in the case of particular pre-run internal air density. This tunnel trace is distinguished by thick line in fig. 4. By changing air density, it is possible to get similar results for other traces. Evaluations of flutter Mach number for each trace produce finally the flutter dynamic pressure curve versus Mach number. Subcritical model dynamic performance data in the space of "dynamic pressure — Mach number" are also determined.

The test techniques is currently being upgraded. The efforts should be applied to better quantitative on-line determination of flutter proximity and flutter mode damping. The post-run analysis should produce not only vibration mode characteristics but also a whole flutter mathematical model enable to predict flutter on the base of the measurements at speeds essentially lower than critical.

The background for this progress has been developed by Russian and foreign scientists.

FLUTTER TESTS IN T-106 TRANSONIC TUNNEL



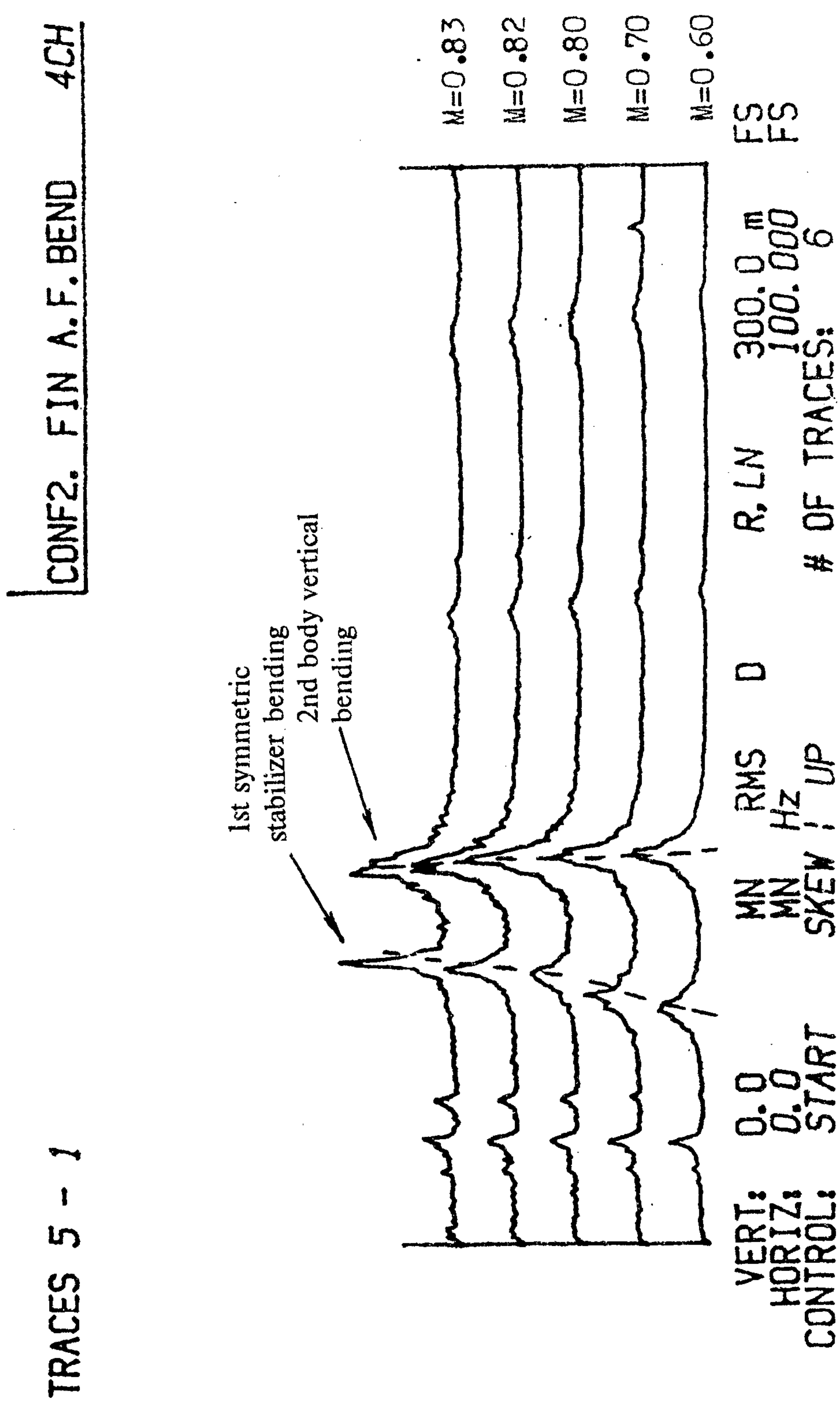


Fig. 2.
Experimental spectral densities.

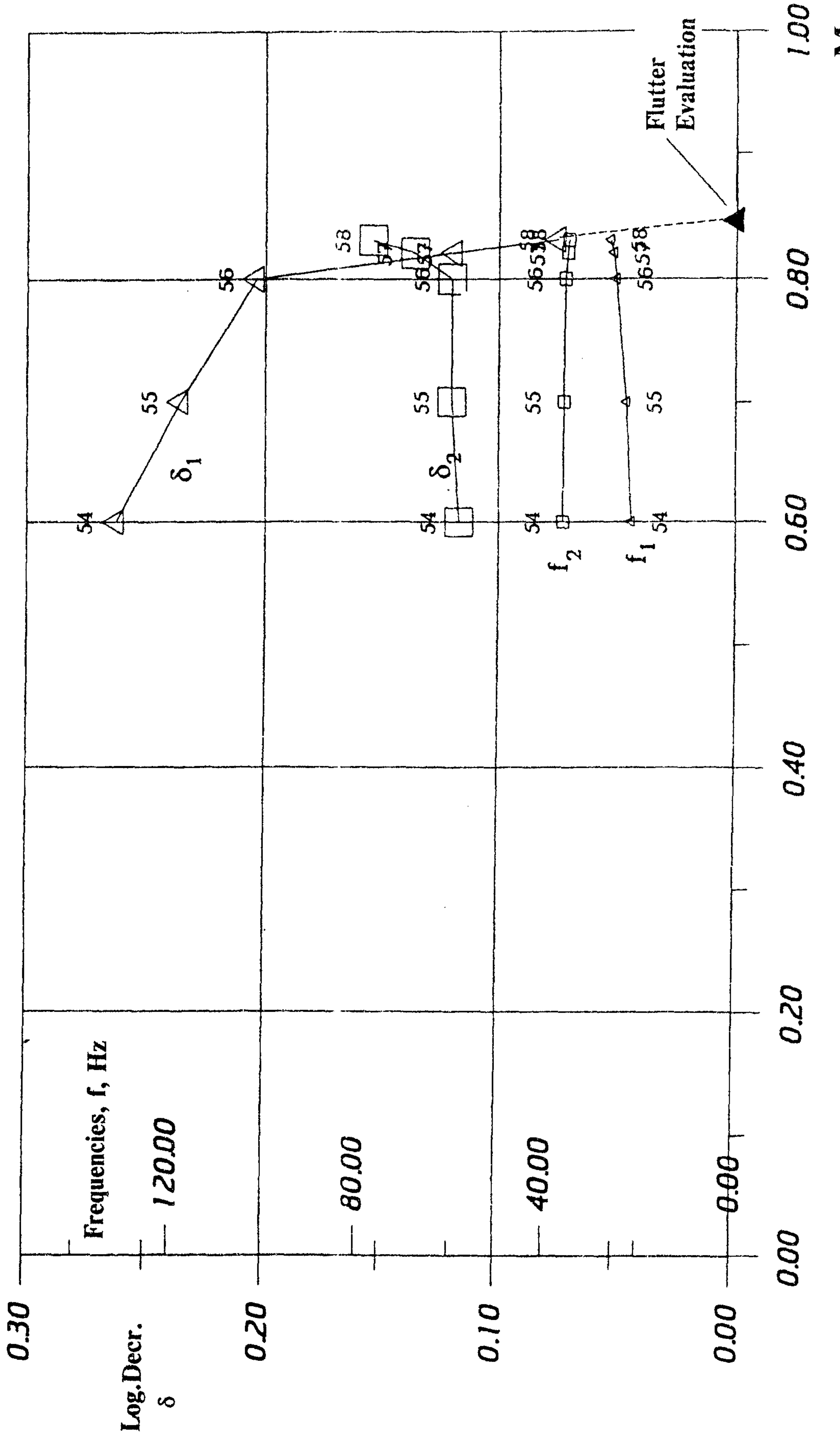

Log.Decr.
δ
0.30
0.20
0.10
0.00
Frequencies, f, Hz
120.00
80.00
40.00
0.00
0.00
0.20
0.40
0.60
0.80
1.00
M
Flutter
Evaluation
δ1
δ2
f2
f1
54
55
56
57
58

Fig. 3.

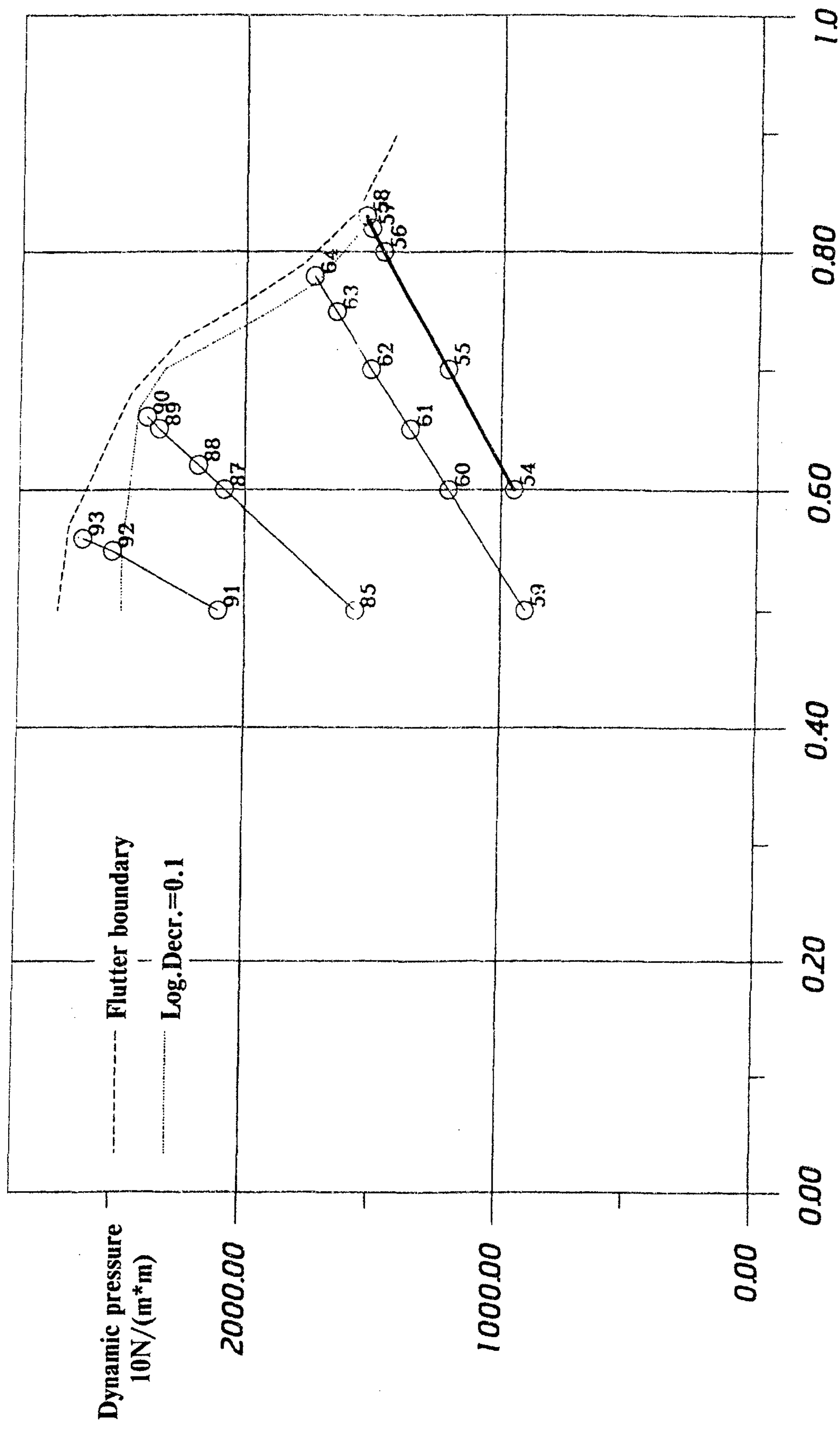

Dynamic pressure
10N/(m*m)
2000.00
1000.00
0.00
0.00
0.20
0.40
0.60
0.80
1.0
Flutter boundary
Log.Decr.=0.1
M

Fig. 4.

BIRDSTRIKE RESISTANCE OF AIRCRAFT COMPONENTS

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M. E. Kelley
J. L. Terry
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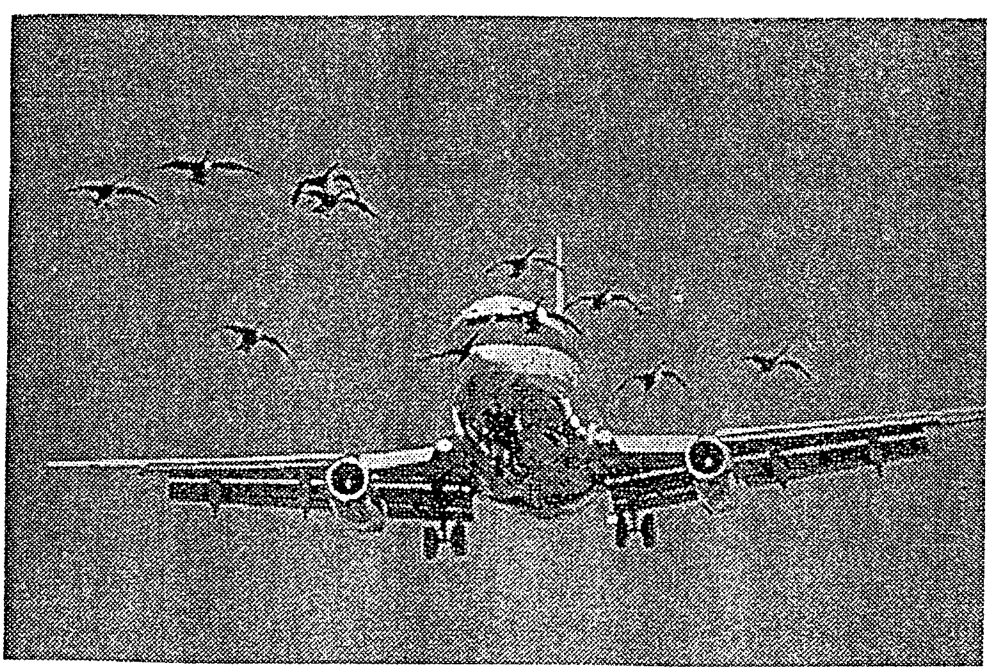
Wright-Patterson Air Force Base, Ohio, USA

ABSTRACT*

Aircraft and birds repeatedly prove that they cannot occupy the same airspace at the same time. Over 3000 birdstrikes per year cause 50-80 million US dollars in damage to USAF aircraft. To the worldwide aviation fleet this problem is estimated to cost more than one billion US dollars per year. The characteristics of the birdstrike problem are described. The predominate impact points causing these high costs are identified. Processes for establishing aircraft birdstrike resistance requirements and capabilities are presented as are processes and facilities for certifying aircraft birdstrike resistance. Also presented are some emerging technologies that show promise for reducing the frequency of birdstrikes.



*To be presented, and used in generating discussions, at the conference on Experimental Facilities and Aircraft Certification, Moscow, Russia, 22-27 Aug 1995

INTRODUCTION

Collisions between birds and aircraft demonstrate a consequence of sharing-the-air. With this acknowledgment comes realization of the need to reduce the frequency and the consequences of such mid-air collisions. A series of words can be used to summarize these collisions: rare, costly, predictable, reducible and tolerable. The intent of this paper is to focus on the facilities and processes used in validating an ability to tolerate a birdstrike. A brief description of these other terms will be covered to place the importance of birdstrike tolerance in perspective.

BIRDSTRIKES

Rare

Birdstrikes are rare occurrences. Most pilots will pursue their career without encountering a significant birdstrike event. Serious birdstrikes are measured in occurrences per million flight hours. Historical records accumulated in the 1970's showed that about 95% of all birdstrikes encountered were at bird weights of less than 4 pounds (1.8Kg). So, the probability of encountering a bird of significant size is rather small. More recent analyses of operational statistics show an increase in this weight to about 4.5 to 5 pounds (2.0 to 2.3 Kg). Two factors are believed to be contributing to this. One is a trend to conduct low altitude flying in corridors where noise will be less objectionable to local civilian populations. This results in increased use of corridors that are likely to be populated with larger birds. The other factor is the increasing populations of these larger birds.

Costly

A significant birdstrike, while rare, can be very costly. The USAF experiences about 3000 birdstrikes per year. These birdstrikes result in a loss of about 1-2 aircraft per year and a loss of about 1-2 aircrew every 3-5 years. USAF costs for birdstrike damage are about $50 Million US dollars per year. Costs for birdstrike damage encountered by the world wide aviation fleet are estimated at about one Billion US dollars per year. Damage costs are a function of three primary variables: bird weight, impact speed, and impact location on the aircraft.

Predictable

While birdstrikes are rare the range of consequences is predictable. The aircraft flight path sweeps through a given volume of airspace. Birds have seasonal as well as day time and night time population distributions within this airspace. Birds also have a probability distribution by weight. The probability of collision with a given weight bird is therefore predictable. The aircraft time and speed in various altitude bands can be predicted for a more precise estimate of the range of probable impact weight and speed conditions.

The probable impact location on the aircraft is directly related to the projected frontal area of the components of concern. While these areas vary with different types of aircraft, a historical distribution of birdstrikes as seen by the USAF is as follows:

Engines	21%
Wings	19%
Windshields	17%
Radome	16%
Fuselage	11%
Multiple	11%
Landing Gear	5%

Analytical tools for predicting both the probability and the structural consequences of a birdstrike are available to those pursuing this task. The tools for predicting structural consequences are also becoming sufficient for use in designing components to tolerate such birdstrike energies.

Reducible

Birdstrikes can not be eliminated but the probability of occurrence can be modified to the benefit (or the detriment) of the aircraft and crew. Information is available on habitat modifications in the vicinity of the airfield to result in either increasing or decreasing bird populations. For example, controlling vegetation height near the runways can reduce attractiveness to birds, and allowing landfills or standing water near the runaway can increase the attractiveness to birds. Tools and reports to assist in this management of the bird population in the airfield vicinity are readily available to those pursuing this task.

Some new technologies are emerging for use in reducing the birdstrike probability. One of the first to find practical application is to modify the birdstrike prediction model with more accurate distribution mapping of bird populations along specific flight corridors. These mappings also include information on seasonal and time-of-day bird activities. These birdstrike probability prediction models then become known as Birdstrike Avoidance Models and are used for comparing alternate flight corridors as to their relative risk for birdstrikes.

As an example, less than 1.0% of USAF birdstrikes are with turkey vultures (cathartes aura) yet these impacts result in about 40% of the damage. It was felt that if the typical daily and seasonal flight pattern of these birds were known, then this could be taken into account in planning a flight corridor or in comparing alternate corridors. Eight birds were captured and fitted with miniaturized radio transmitters. Data being accumulated will be used to modify the Bird Avoidance Models. With this better understanding of the vulture flight patterns, both the birds and aircraft will be better off and that this will result in a USAF damage reduction of at least $10M per year.

Two new approaches to birdstrike reduction are being explored. One involves better information for use in real-time avoidance of a particular birdstrike event that is likely to be significant. The other involves active deterrence of bird activity in the aircraft flight path. Both approaches are still in their infancy.

Radar tracking of birds has long been a tool used by ornithologists in their studies. Radars, both airborne and ground based, that are used for aircraft flight path information, detect birds. This information is not critical to flight management and is filtered out as unwanted clutter. The technology being explored is to take this bird detection information and process it along with aircraft flight path information through an artificial intelligence network to predict birdstrikes that are about to occur. For those that have a high probability of being a serious incident the aircrew could be given a warning to take evasive action. For impact with single birds this is a fairly straight forward calculation. For collision with a flock of birds, it is not so clear cut as to what will constitute a serious situation. To build this understanding, equipment is being developed for use in the field to measure spatial characteristics of representative bird flocks. This information can be used in the probabilistic birdstrike collision prediction models to determine quantities of hits likely to be encountered by various aircraft areas of concern. Analytical tools used in structural analysis of the birdstrke event can then be used to predict when significant consequences will be experienced. This can then be used as criteria in the artificial intelligence process.

The other approach to birdstrike reduction is taking action which will cause birds to avoid the flight path of the aircraft. It is believed that if given the chance birds will avoid confrontation with an aircraft and that most collisions are a result of the bird not seeing the aircraft in time to get out of the flight path. Techniques such as strobe lights and rotating geometric patterns, to increase aircraft visibility have been experimented with for some time. The results are argumentative. These techniques all have a pre-requisite for their effectiveness and that is the bird must first look in the direction of the aircraft. Research has shown that birds detect sound at frequencies below that heard by humans. Under laboratory test conditions they respond to this sound in a manner indicative of visually searching for a source. It is theorized that this system will cause the birds to look around for the sound source and that this will increase their opportunity to see and thus avoid the aircraft. The first probable application of this infrasound warning technique will be in a ground based system for use at an airfield. To reduce habitation, this system will only be activated when aircraft are in landing or take-off operations.

While the process is not understood, it has been observed that birds detect and avoid high energy radar beams. If the ground based infrasound warning system proves viable, then second and third generation spin-offs are being considered. The second generation would modulate a radar beam with an override frequency that would carry the infrasound message. The power and range of this radar beam would only be as necessary to cause birds in the danger zone to search for the sound source and thereby hopefully see the approaching aircraft. The third generation will attempt to benefit from the close anatomical relationship between a bird's hearing system and its balance system. If the modulated radar infrasound message can be received in a format or volume that temporarily disturbs the balance system, the bird may take direct evasive action to avoid, or it may involuntarily fall out of, the danger zone.

It is the goal of this work in active deterrence to, first, reduce the frequency of birdstrike collision and second to do this in a way which does not cause permanent injury to the birds.

Tolerance

When the inevitable proof reoccurs that two objects can not occupy the same airspace at the same time, hopefully the aircraft has sufficient structural integrity to tolerate the birdstrike energy without catastrophic loss of aircraft or aircrew. With only about one out of a thousand birdstrikes resulting in such a loss this is indeed the case in most birdstrikes. Tolerance of the birdstrikes event means the aircraft subsystem(s) being impacted must safely absorb the energy of accelerating the bird mass to some significant fraction of aircraft speed and do this in an aircraft travel distance of about the bird length or width depending on impact direction.

Absorbing the birdstrike energy occurs through deformation of the aircraft structure. Obviously, not all birdstrike energies can be tolerated. It then becomes a tradeoff of cost, weight penalty, and probability of occurrence in setting the level of required tolerance.

For some critical structures and surfaces this means design for, and test to, a bird impact weight of eight pounds (3.6 Kg). For the majority of the aircraft frontal area this means design for, and test to, a bird impact weight of four pounds (1.8Kg). For the engines the requirements vary but are essentially driven by engine inlet size and include bird weights up to eight pounds (3.6Kg) as well as multiple 1.5 pound (0.7Kg) and 2.5 pound (1.1Kg) birds. For certification, each subsystem will have criteria related to the damage that can be allowed. These criteria for various impact weights, and locations, can range from a requirement to have no effect on engine performance, to one of being able to sustain performance at reduced power, to one of being able to safely shut down engine operation entirely.

The speed at which these requirements must be met is generally tied to the anticipated speed in the birdstrike environment. For some aircraft, such as for commercial cargo and passenger use, the weight and cost penalty for achieving tolerance is reduced by imposing a requirement to stay below a certain speed when at an altitude that places the aircraft in the high risk birdstrike environment.

Structural analysis computer codes are becoming available for use in designing subsystems to tolerate the birdstrike energy. These codes have become quite efficient in replicating the non-linear material behavior characteristics that occur during a birdstrike event. An event that may last only 0.001 seconds and result in generation and dissipation of a birdstrike force approaching 100,000 pounds. The use of these codes has greatly reduced the historical and costly design, test, redesign cycle. While the need for test facilities to support this cut and try approach has diminished they are still used for design validation and flight certification testing.

TESTING AND CERTIFICATION

All external components having a forward facing projected area are subject to birdstrikes. It is reasonable to expect those responsible for such subsystems to certify compliance with birdstrike tolerance requirements.

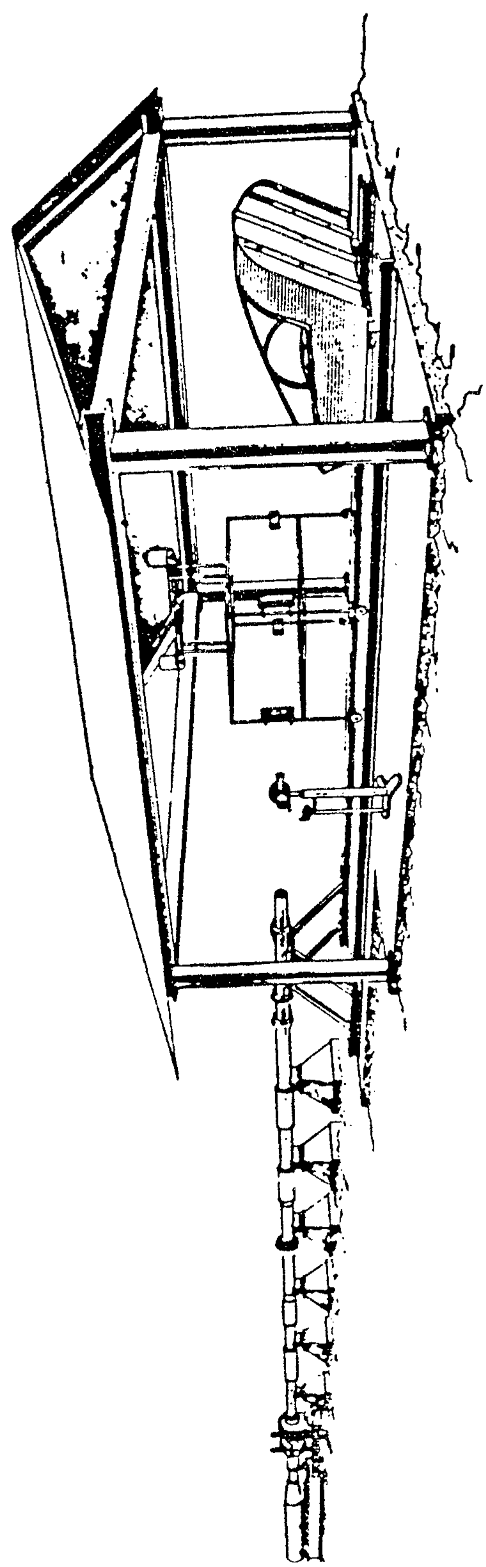
As analytical codes mature for analysis of structural response to the birdstrike event, there is less need to demonstrate compliance via actual testing. Dependence on such codes in lieu of testing requires experience in their use and an understanding of the degree of departure of the design being analyzed, from a design which was verified in full scale testing to be in agreement with predictions.

The item being tested should be representative of operational hardware and should be mounted in support structure representative of the actual aircraft in order to take into account the dynamics of structural response to the actual birdstrike event. The testing should include environmental extremes representative of conditions likely to be encountered in an actual birdstrike.

Testing should include impact locations where maximum stiffness is expected, where maximum deflection is expected, where critical support structure, actuating mechanisms, power lines, fuel lines or hydraulic circuits are hidden and otherwise presumed safe, and where impact shock dynamics can activate or dislodge electro-hydraulic switching or actuating mechanisms that are critical for continued flight.

The use of artificial, real, or commercial birds is a choice that must be based on several factors. The artificial birds do create realistic impact loading and they are economical, both in preparation and in clean-up. They do invariably leave certification authorities with an uncomfortable feeling of "But--are the results real?". Real birds representative of those expected in actual operation certainly answer this question but they are costly to acquire and environmental protection considerations make it difficult to justify their use. Commercially available birds, such as chickens, are bred to have a different structure than wild birds. Both real birds and commercial birds are costly to use in terms of facility clean-up after each test. The series of choices frequently ends up by using artificial birds for development testing and commercial birds for certification testing.

A typical bird impact range is shown in Figure 1. The components consist of a tank for holding pressurized air, a pressure release valve, a chamber for holding a sabot which



holds the impact projectile, a tube for directing the projectile as it is accelerated by the pressurized air, a constrained portion of the tube to strip the sabot from the impact projectile, instrumentation for measuring the velocity and orientation of the projectile, a station for mounting the item to be impacted and a backstop (not shown) for absorbing residual energies. Numerous electrical interconnections are incorporated for safety and data acquisition. Provisions are made to enclose the impact area with insulating blankets or curtains and for use of heating or cooling equipment. These are removed just prior to the actual test so as to not interfere with the test.

High speed photography is accomplished with motion picture or video equipment. A capture rate of 5000 frames per second has been found to be minimal for analysis of results. Multiple cameras and lighting are synchronized and activated as part of the automatic firing sequence. By strategically locating and synchronizing selected cameras, and use of computer aided film analysis, a three dimensional, time based, deflection map of the item being impacted can be created. This map can be compared to the predicted deflection map. Under certain conditions, this comparison lends credence to further use of the predictive tools and can significantly reduce the quantity of tests required.

This facility is obviously for impact of stationary targets. For rotating targets the automatic firing sequence includes additional controls to assure hitting the desired location. On one facility for rotating targets the launching sequence is so precise that a test can be conducted where the bird goes between two blades and the back of the blade hits the bird. Generally the rotating item is connected to the drive mechanism through frangible couplings. For some of these rotating target tests, multiple launch tubes are used and in some cases spring loaded mechanisms are used in lieu of the air cannon to launch the projectile(s).

A second type of facility is also used. Some aircraft certification programs involve testing using a sled - track where rocket motors accelerate the test item to a desired speed. Under these conditions the bird impact tolerance certification can be accomplished at little or no additional cost by suspending the bird carcass in a position where it is hit by the test item.

It is sometimes argued that the lack of airflow in the first type of facility is sufficient reason to justify a requirement to use the second type of facility. Analysis of results from both types of facility shows little basis for this argument. True, the aerodynamic loading does add to the forces on the item being tested but this is well within the scatter of forces from the bird impact. True, the aerodynamic flow field does exert forces that can change the trajectory of the bird but for birds of a size sufficient to damage the structure, this course alteration is insignificant. Unless there is some overriding reason such as to assure that the external trajectory of impact debris does not interfere with an engine, then the cost of the second technique, solely for birdstrike certification, is not warranted.

Facilities for testing of non-rotating articles are located in the US at:

- Arnold Engineering Development Center
  Bird Impact Test Range
  Arnold AFB, TN 37389, USA

- Boeing Commercial Aircraft Co.
  Birdstrike Test Facility
  P.O. Box 3707
  Seattle, WA 98124, USA

- Lockheed-Martin Aircraft Systems
  Structural Test and Fluid Dynamics Lab
  P.O. Box 748
  Ft Worth, TS, 76101-0748, USA

- PPG Industries, Inc
  Aircraft products (Qualification testing)
  P.O. Box 2200
  Hunrsville, AL 35804, USA

- University of Dayton
  Research Institute
  300 College park
  Dayton, OH, 45469-0101, USA

Facilities for testing of rotating articles are located in the US at

- Wright Laboratory
  Bird Impact Test Range (WL/POM)
  Wright-Patterson AFB, OH, 45433-6563, USA

- Each engine manufacturer has a facility for use relative to their engines

CONCLUSION

This paper was assembled as a means to start conversations. Conversations to explore possibilities. Possibilities of sharing in the development, validation and application of technology to improve flight safety by reducing the costly consequences of mid-air collisions between birds and aircraft.

Each of the authors has many years of experience in improving aircraft birdstrike tolerance and would welcome a chance to explore possibilities for extending the underlying technologies.

ABOUT THE AUTHORS

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Malcom Kelley. Telephone 513-255-6524. Malcolm has been involved in this subject area for some 15 years. His primary area of expertise is in finding ways to obtain longer service life of components that have been increased in design complexity to provide increased birdstrike tolerance. For the past four years he has focused his attention on technologies to reduce birdstrike probability.

James Terry. Telephone 513-255-2734. Jim has been involved in this subject area for some 10 years. His primary area of emphasis has been in developing and transitioning abilities to meet the increased energy absorption requirement without an undesirable increase or decrease in other characteristics. For the past four years he has focused his attention on engine birdstrike tolerance.

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COMPOSITE MATERIALS IN AIRCRAFT STRUCTURES

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1. SURVEYING THE USE OF COMPOSITES IN AIRFRAMES

The composite materials are regarded to be much promising from the viewpoints of both extending the aircraft capabilities and improving the economy figures. Therefore aircraft companies conduct comprehensive projects for utilizing the composites.

The international and Russian data available, see Fig. 1, clearly demonstrate permanent growth in aviation composite materials utilization. according to forecast in the technical literature, the amount of composites in passenger-carrying airplane airframes will by year 2000 be as large as 15 - 20% (referred to the weight of all materials used), and in military airplanes, 35%.

As to aircraft industry of the Commonwealth of independent States, all major design bureaus have mastered these materials. Let us address practical examples of the use of composite materials in domestic airplanes.

In TU-204 passenger-carrying airplane (Fig. 2) the amount of composites is 9% of the weight of all materials used; the plan is to increase this parameter to 15% by introducing composites in structures of vertical and horizontal stabilizers.

In AN-70 transport (Fig. 3) the weight of composites if 25% amongst all materials used. Large-size structures of vertical and horizontal stabilizers are completely manufactured out of composite; sandwich panels are with tubular cores.

A mass-produced MIG-29 fighter (Fig. 4) has 7% fraction of composites (by weight); in future fighters the weight fraction of composites in lifting surfaces, wing high lift devices,and body components will be as high as 30%.

Composite materials were also used in Buran orbiter: payload bay doors have been manufactured out of a carbon/carbon material (Fig. 5)

2. LAY-UP AND CHARACTERISTICS

Widespread aviation composite materials are the polymer matrix composites; used as a filler in them are glass, organic and/or graphite fibers. Figure 6 demonstrates usual combinations of components for the fibrous composites utilized in aviation.

Levels of specific strengths and stiffnesses of composites are mainly defined by properties of the fibers. The diagram compares specific strengths and stiffnesses of unidirectional composites with epoxy matrices and various reinforcing fibers.

Glass-reinforced plastics have high allowable stresses and low stiffness. Organoplastics can carry high tensile stresses but have a relatively low Young's modulus and low compressive strength. Carbon fiber reinforced plastics are now the main structural materials for load- bearing components of structures. Various fillers can be combined in a material system, forming a hybrid material.

Mechanical properties of a matrix are determinants for a shear load capability, buckling behavior, and maximum use temperatures; fatigue and impact resistance depends on a matrix.

Currently, efforts are undertaken to develop composites based on thermoplastics. The latter yield at elevated temperatures and harden at room temperature. As compared with thermoset composites, the thermoplastic-based systems have a number of advantages:

- ease of use,
- simpler repair, the potential for reuse,
- almost no limitations on prepreg storage duration,
- insignificant sensitivity to moisture,
- high fracture toughness, damage tolerance, see Fig. 6.

Most widely the aircraft companies manufacture the composite structures from prefabricated prepregs, i.e. unidirectional layers impregnated with a matrix material. The prepregs are built-up in compliance with a prescribed pattern, thereafter united in a final material by polymerizing the stack (Fig. 7). The resulting material has certain average stiffnesses and strengths, which can be controlled by varying the composition parameters such as the total number, lay-up angles/sequence, and thicknesses of the layers.

The average properties of composites are predicted by analysis based on anisotropic multilayered plate theories. Used as initial data are characteristics of a unidirectional orthotropic ply that are listed in a table in Fig. 7. The ply characteristics are evaluated experimentally.

3. DESIGN OF COMPOSITE STRUCTURES

A traditional task during design of a primary structure to a prescribed load is to specify structural parameters such as to attain a minimum mass under constraints. In the case of composite structures the set of design parameters is extended by adding parameters of the stack-- the directions and amounts of unidirectional layers. The increase in the total number of design parameters; more details in an analytical model; extra limitations associated with manufacture and use of composite structures; these factors result in a very cumbersome design process. The problem can be simplified by dividing the design procedure into stages as presented in Fig. 8.

First stage: a multidisciplinary aero/strength/dynamic design of beam/plate models in order to preliminarily specify parameters of a minimum-weight structure under requirements for strength, aeroelastic behavior, controllability, fatigue life, etc.

Second stage: a coarse finite-element model is used to formulate a reasonable structural concept and to improve parameters including the "microstructure" of materials.

Third stage: individual structures, assemblies and joints are optimized.

Fourth stage; a structural analysis is carried out using the final FE model of high dimension; all requirements are checked out.

At TsAGI, methods for the first and second stages are implemented in ARGON complex; its schematic may be seen in Fig. 9

Proceeding from the initial data represented in this Figure, the program conducts the multidisciplinary aero/strength/dynamic design of a continuum-mechanics model. Airloads are evaluated with due account for aeroelasticity; the extreme load conditions are revealed; the fiber orientation angles and thicknesses are chosen; aeroelastic behavior is evaluated.

At the second stage, a finite-element model is formed; a reasonable structural concept is outlined; fiber orientation angles and thicknesses are refined. Figures 10 and 11 illustrate the use of ARGON in application to design of a vertical stabilizer of a high-technology aircraft.

At the stage 1 (Fig. 10) the analytical model is as simple as a sandwich panel with a rigid core. Weight and stiffness requirements are complied with by establishing a reasonable composition of primary layers. At the stage 2(Fig. 11)the finite-element model is used to form a reasonable structural concept, a stack composition, and thicknesses for each structural unit.

The stage 3 (i.e., an individual treatment and optimization of structural elements) s implemented in the TsAGI-developed COMPOSITE program package. Figure 12 demonstrates major parts of the package.The next Figures represent some applications of the COMPOSITE complex:

- computation of strengths and stiffness characteristics, evaluation of load-carrying capability and stability of panels(Fig. 13) and
- optimization of ccc panels and shells to withstand the general loads shown (Fig. 14).

4. FEATURES OF MEETING THE STATIC AND FATIGUE STRENGTH REQUIREMENTS FOR COMPOSITE STRUCTURES

When designing the composite structures, account should be taken of additional factors that weaken the structure. The main factors are as follows (Fig. 15):

- scatter in mechanical properties, more notable than that of metals,
- brittleness and the related sensitivity to stress concentration and impacts,
- sensitivity of structural characteristics to environmental attack and length of service.

The influence of these factors should be allowed for at a design stage; with this, the conventional system of measures for meeting the static and fatigue strengtil requirements that was developed for metallic structures.

4.1 Notable scatter of mechanical properties

The more significant scatter in strengths of structures made out of composites is taken metal structures, a damage(such scratch, indentation) detectable by eye does not almost degrade the load-bearing capability, while initiation of fatigue cracks. After a crack reaches a critical length, the structure fails.

Damage in a composite structure appears mainly due to accidental mechanical impact that can occur at any time during operational use. In this case the strength characteristics fall suddenly. As a rule, usual variable in-service loads develop a damage extremely slowly.

A composite-structure failure mode depend on an impact energy level (Fig. 17). High and medium energy impacts damage a surface; such areas con be detected visually and repaired. The most hard danger is associated with low-level impacts; indications are difficult to reveal visually, and nondestructive methods should be employed.

Investigations show (see Fig. 17) that a low-energy impact can drastically degrade a component strength, and a compression load carrying capability reduces more notable than a tensile load limit.

The sensitivity to impacts poses a number of extra requirements on designers:

- the composite structure should be designed as a fail-safe concept, i.e., it must maintain a desired strength after a standardized damage occurs;
- the influence of the standardized damages must be allowed for when specifying the allowable stresses -- by introducing a coefficient K_{des};
- operators should establish a systematic pre-flight visual inspection and periodic nondestuctive inspections;
- new typical processes for repairing the damaged structures in field conditions should be developed.

5. CERTIFICATION OF AIRCRAFT COMPOSITE STRUCTURES

The existing approach to airframe certification has been made perfect; it assumes structural analyses and theoretical evaluations of the strength properties, to be validated by certification tests.

The diagram in Fig. 18 represents the amounts of certification works for the composite structures; the shaded cells indicate additions caused by the replacement of metals by composites. In comparison with the metallic structure, the composite structure must be shown to be capable of meeting the static and fatigue strength requirements after introduction of standardized damage in the most severe environments; complementary efforts are undertaken to substantiate

- fail-safe operation capability
- damage tolerance,
- inspectability, and
- repairability.

The analytical estimation of the strength includes; - a theoretical computation of stresses and strains by means of state-of-the-art methods; currently finite element methods are widely employed; - the use of data available in handbooks and from the experience of designing and operating similar structures.

The estimates and validated by certification tests that envisage large amounts of testing for static and fatigue strength of various specimens. The range of specimens may be categorized in types (Fig. 19) which differ in sizes, total numbers of specimens, and values of interest.

Elementary specimens are usually small-size and can easily be tested in prescribed environmental conditions. These tests supply mean values of characteristics and the scatters. Experimenters determine degrees of the influence of various service factors in various combinations; the worst conditions by strength degradation are outlined.

Some "fragments" being medium-size portions of a structure are used to evaluate technological concepts and/or to obtain the strength limits. Such specimens are too large to be used for evaluating the scatter, but the strength degradation under real environmental effects can well be estimated.

Units being important parts of an airframe can be very large. The experimentation usually involves one or two copies. The principal problem in this case is to evaluate the concepts and manufacturing processes by strength. For these tests two approaches are possible: - the direct tests with simulation of service conditions and - an "analysis and experiment" approach that envisage tests in conventional environment and a subsequent extrapolation of the data to the most severe conditions.

The entire airframe with composite parts are tested in compliance with the test amount requirements adopted for "all-metal" airframes. The full-size airframe is tested in normal environment and the results are corrected for in-service factors; here, reliance is made on analytical procedures and the data obtained from testing of specimens of other types.

The aim of the full-size tests is to comprehensively validate the airplanes put into service and to ensure their strength and long operational life.

6. DESIGN CONDITIONS FOR PASSENGER-CARRYING AIRPLANES

6.1. The influence of environmental conditions

As to the passenger-carrying airplanes, the environment parameters are as defined in Fig.20. Extreme conditions are simulated during static tests, and/or residual strength tests, whereas the long-term strength parameters are determined by exposing the airframe during the fatigue strength tests.

6.2. Fail-safe design requirements

The impact may be prescribed in terms of an impact energy or a damage area size-- these values are interrelated. An in-service damage to a full-scale structure is modeled by a notch or a spherical indentation. The size of a design limit in-service is defined on the basis of a value at which the damage will be reliably (at a probability of not less that 90%) detected by the usual inspection technologies and inspector's skill.

An airplane operated within the fail-safe design principle must remain strong with in service damage. The design limit conditions establish a relation between a standardized damage and the factor of safety. Figure 20 demonstrates fail-safe conditions; the abscissa axis represents ranges of the damages, and the other axis corresponds to the necessary strength margins which must be substantiated. Specific parameters of the standardized damages depend on a number of aspects(a type of a structure, a material, skills, etc.) and should be established via statistical data and/or special tests for damage inspectability.

7. MEANS OF COMPLIANCE

To demonstrate the correspondence of real strength characteristics of a structure to the certification requirements, the means of compliance are developed. Let us cite some of these methods that were used for certifying the composite assemblies of TU-204 passenger-carrying transport.

7.1. Nondestructive inspection method validation

Using a certain nondestructive inspection technique(specifically, the ultrasound impedance method), the damage sizes are determined and compared with actual dimensions measured on a microesection of the damage area. The comparison makes it possible to estimate acceptability of the method and introduce the proper corrections(Fig. 21).

7.2 Damage detectability

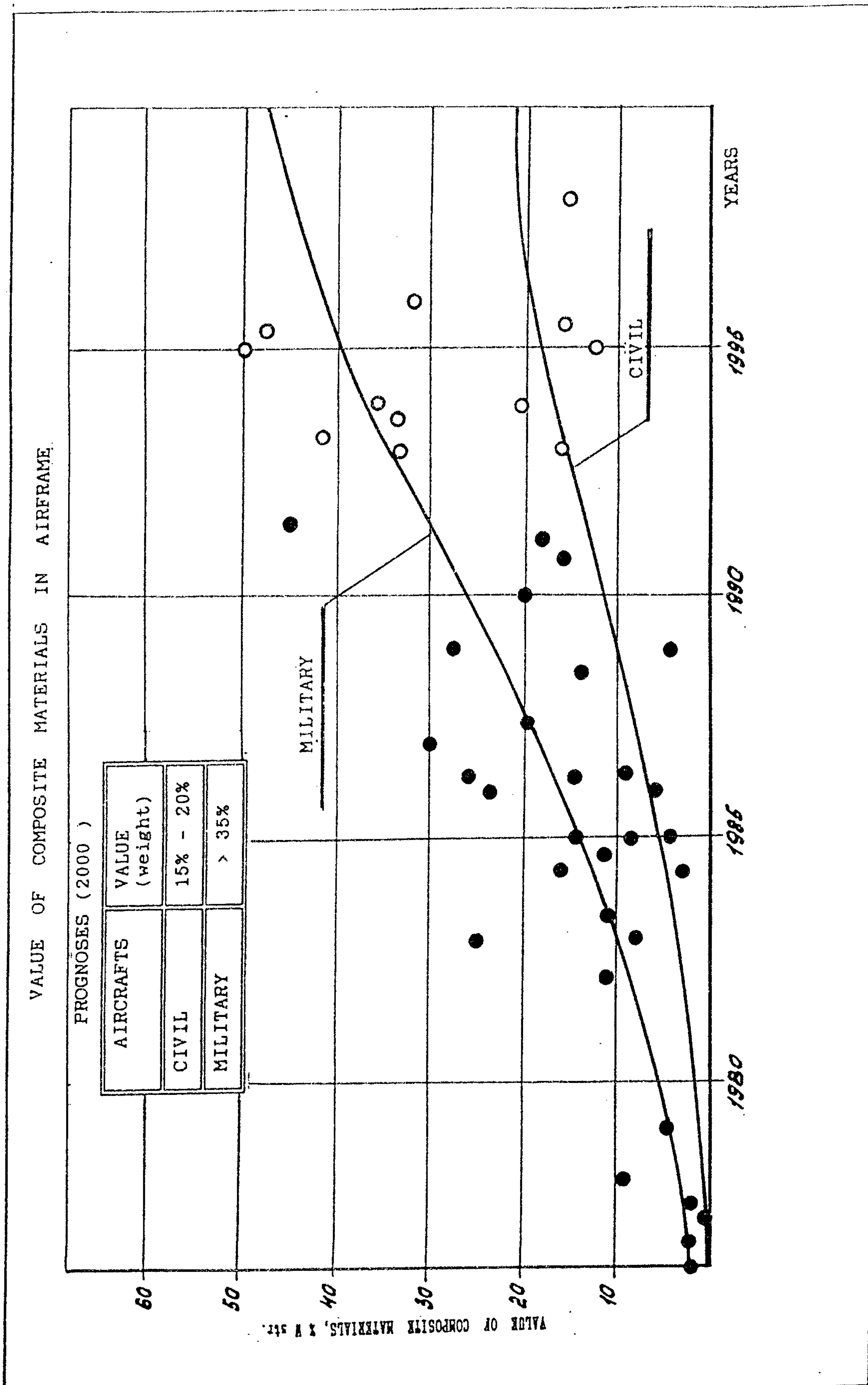
Boundaries of reliably detecting a damage by means of instrumented and visual inspection were outlined via detectability tests. The upper and lover surfaces of an aileron were damaged by impacts with various energy levels, and the damage area sizes were measured. In so doing, the plot of a characteristic length of a damage area versus the impact energy level have been generated, Fig. 21; thereafter, boundaries of reliable detection by instruments and visual inspection were outlined.

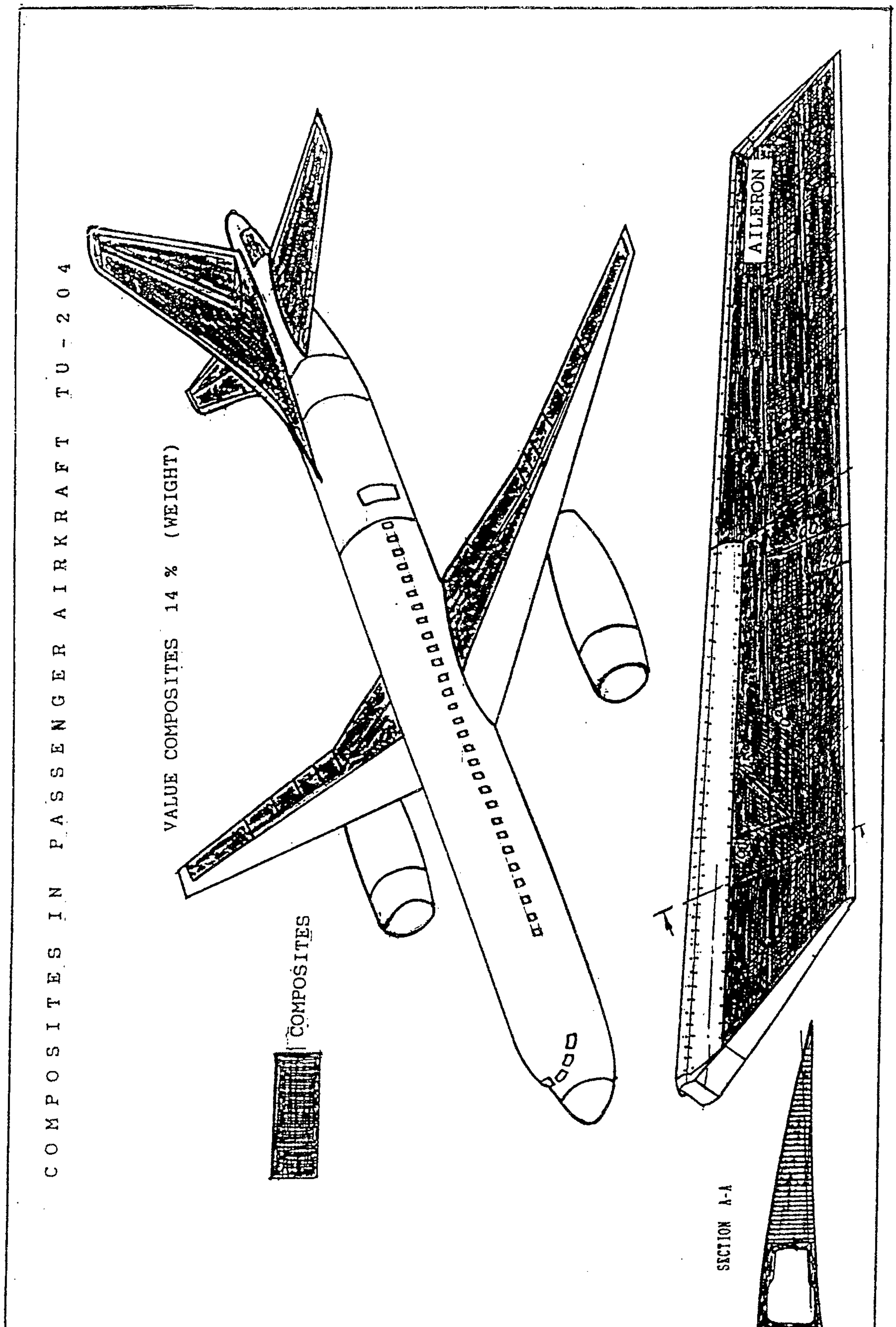
7.3 Environmental effects

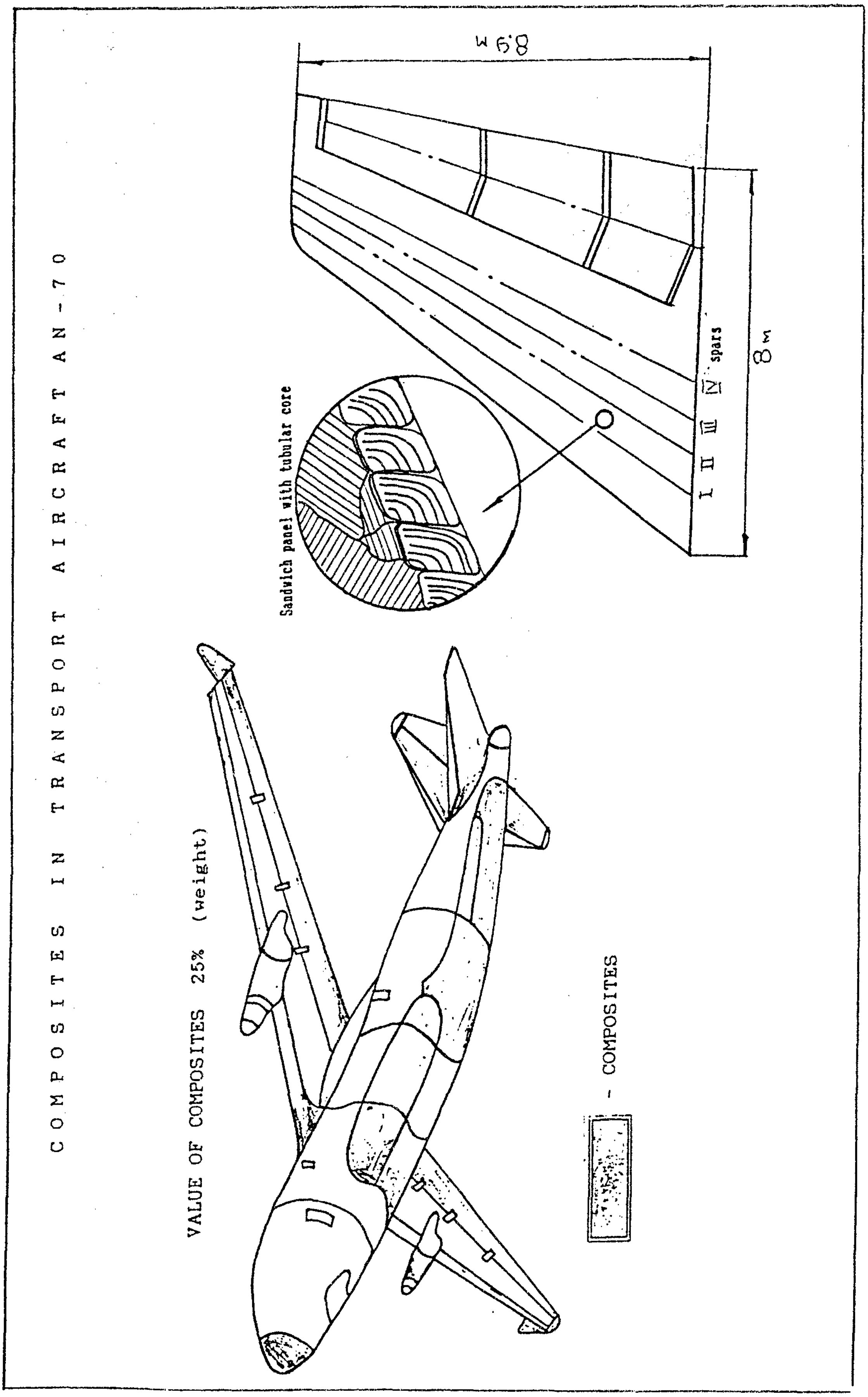
There arises the question of how to demonstrate fulfilling the static/fatigue strength requirements for the composite structure as of the finish of the design service life when properties degrade under environmental effects and cyclic loading? One of approaches is to conduct the full-size structure test to a program that ensures close simulation of critical environmental conditions and loads. Such tests for an airframe as a whole are difficult to perform, but some, medium-size structural units may be studied in this way.

7.4 Damage growth rate

This value is necessary to assess the inspection schedule and the structure life capability. The damage growth rate determination method is as follows: during the life evaluation the structure is systematically inspected with instruments to measure the visible damage sizes. These measurements are a basis for plotting the size variation versus the number of cycles; damage growth rate is computed. This method is illustrated in Fig. 24.


VALUE OF COMPOSITE MATERIALS IN AIRFRAME
PROGNOSES (2000)
AIRCRAFTS
VALUE (weight)
CIVIL
15% - 20%
MILITARY
> 35%
MILITARY
CIVIL
YEARS
1980
1985
1990
1995
60
50
40
30
20
10
VALUE OF COMPOSITE MATERIALS, % W str.

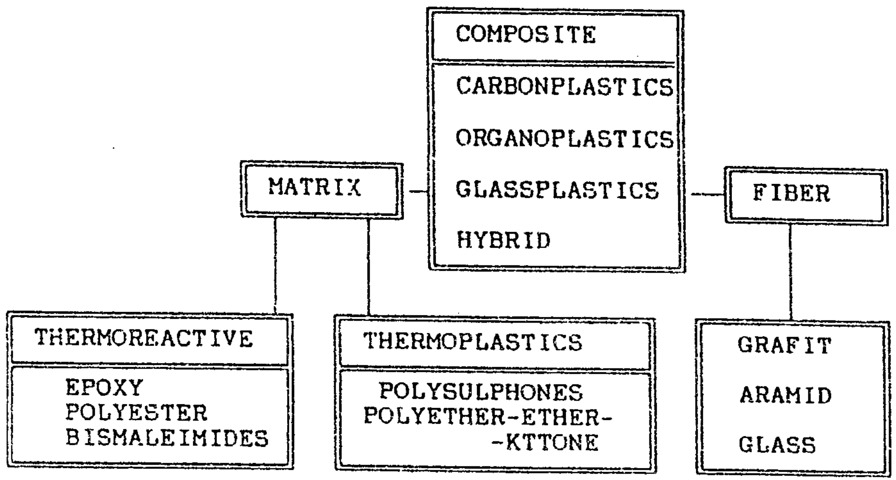

COMPOSITES IN PASSENGER AIRKRAFT TU - 204
VALUE COMPOSITES 14 % (WEIGHT)
COMPOSITES
AILERON
SECTION A-A


COMPOSITES IN TRANSPORT AIRCRAFT AN-70
VALUE OF COMPOSITES 25% (weight)
- COMPOSITES
Sandwich panel with tubular core
8.9 m
8 m
I II III IV spars

DTIC COULD NOT GET MISSING

PAGES 179 & 180 FROM CONTRIBUTOR

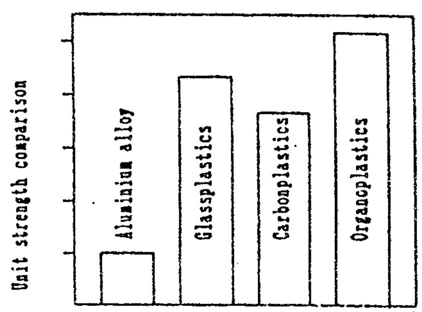
FIBR COMPOSITES WITH RESIN MATRIX

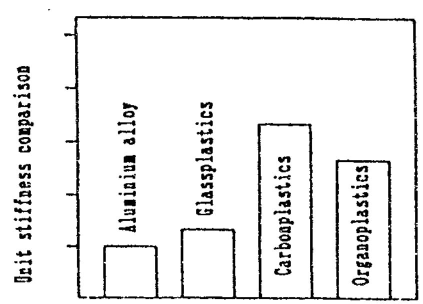


ADVANTAGES OF THERMOPLASTICS MATRIX COMPOSITES

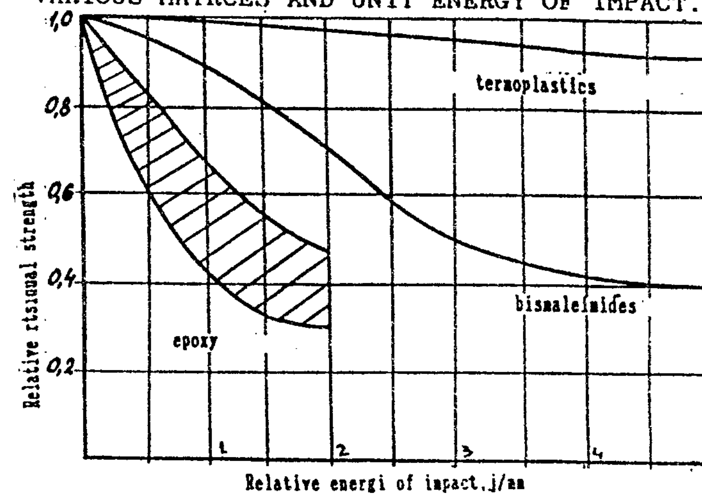
1. MORE TECHNOLOGICAL.
2. MAY BE REPAIRED, MAY BE USED FOR REPATED MANUFACTURING.
3. THERE IS PRACTICALLY UNLIMITED PERIOD OF CEEPING PREPREGS.
4. MORE RESISTANT TO HUMIDITY ACNION.
5. CHARACTERISTICS OF VISCOCITY OF DESTROYING AND DAMAGE TOLERANCE ARE BETTER

UNIT CHARACTERISTICS COMPARISON

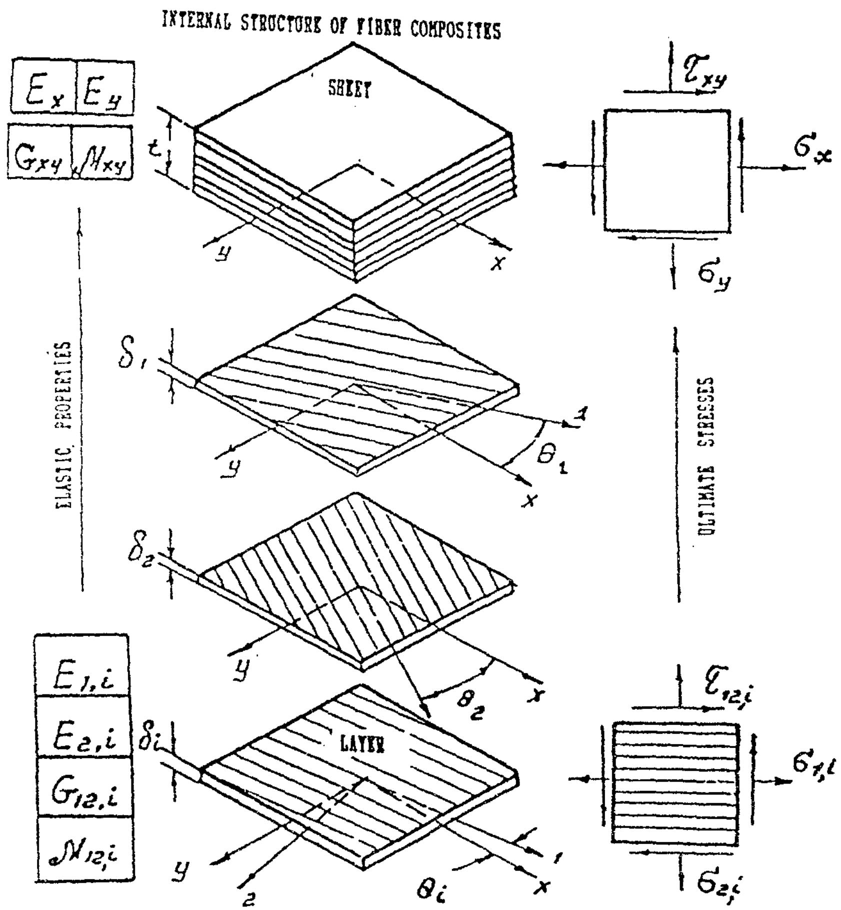




RELATIONSHIP BETWEEN RESIDUAL STRENGS OF COMPRESSED CARBONPLASTICS WITH VARIOUS MATRCES AND UNIT ENERGY OF IMPACT.



INTERNAL STRUCTURE AND PROPERTIES OF FIBER POLYMER COMPOSITES



TYPICAL PROPERTIES OF UNIDIRECTIONAL COMPOSITES.

NN	PROPERTIES		MATERIAL DESIGNATION	CARBONPLASTICS EPOXY .V=60%	ORGANOPLASTICS EPOXY.V=60%	GLASSPLASTICS EPOXY.V=45%
1	JOUNG MODULI	ALONG	E1 kg/mm2	13000	8000	4000
2		ACROSS	E2 kg/mm2	700	600	300
3	SHEAR MODULE		G12 kg/mm2	600	200	400
4	POISSON RATIO			0.28	0.34	0.26
5	ULTIMATE STRESS (TENSION)	ALONG	kg/mm2	110	140	110
6		ACROSS	kg/mm2	2	1	3
7	ULTIMATE STRESS (COMPRESSION)	ALONG	kg/mm2	70	20	60
8		ACROSS	kg/mm2	13	5	12
9	ULTIMATE STRESS (SHEAR)		kg/mm2	6	3	7

- UNIDIRECTIONAL LAYER PROPERTIES ARE DETERMINED EXPERIMENTALLY,
- AVERAGED PROPERTIES OF A LAMINATEY ARE DETERMINED NUMERICALLY BY TECHNIQUES BASED ON THEORY OF LAYERED ANISOTROPIC PLATES.

MAIN STAGES IN DESIGNING THE COMPOSITES AIRFRAMES

ARGON

I STAGE	
Aim	Determination of thicknesses and lamination parametres
Model	Continuum mechanics : roads, beams, plates
Design variables	Thicknesses, ply angles
Constraints	Strength, displacements, aeroelastic behaviour, min/max thickness

II STAGE	
Aim	Upgrading the thickness distribution and structural concept
Model	Coarse, discrete, finite-element
Design variables	Thiknesses, ply angles, structural concepts
Constraints	Strength, stiffness, etc.

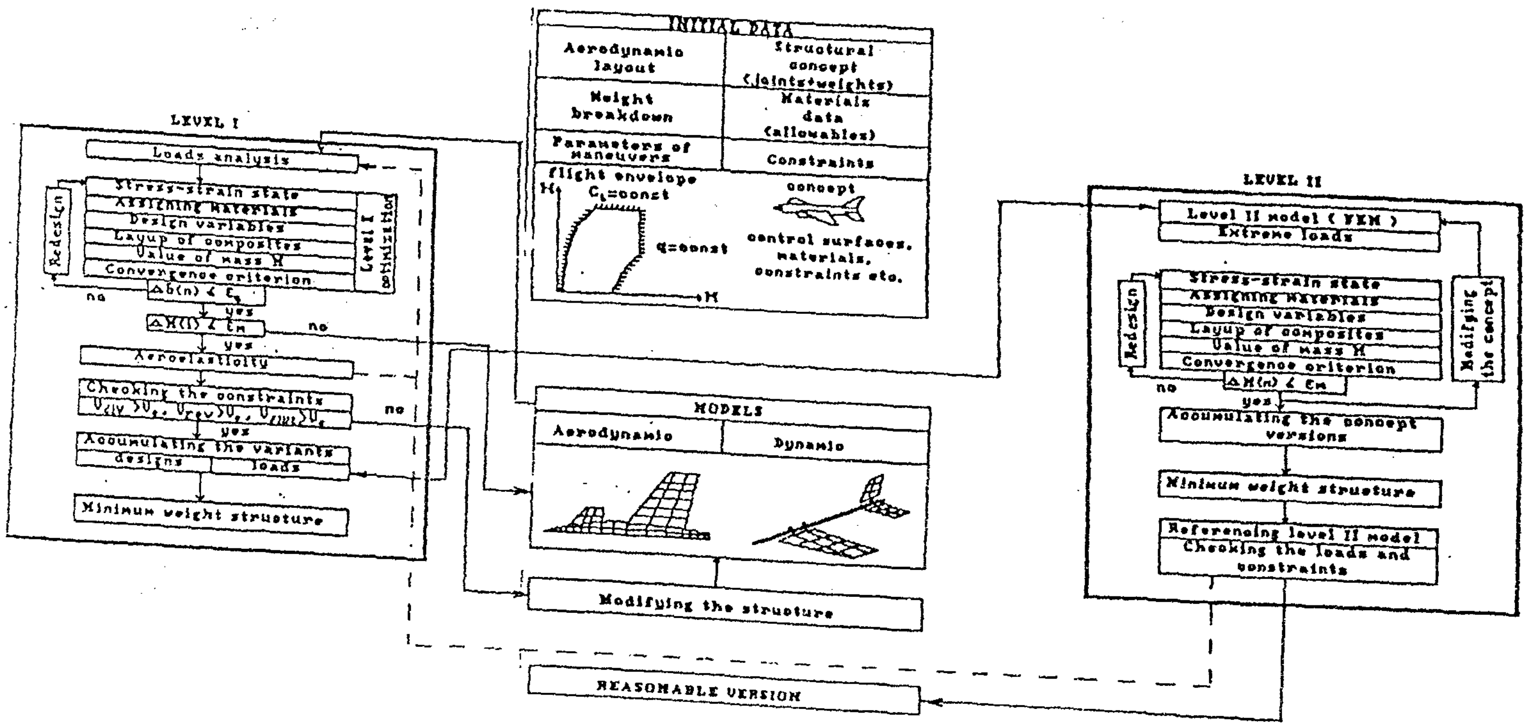
COMPOSITE

III STAGE	
Aim	Optimization of joints and individual components
Model	Roads, beams, plates. Finite Element Model
Design variables	Microstructure of material Structural concept
Constraints	Strength, stiffness. Mass

MARS

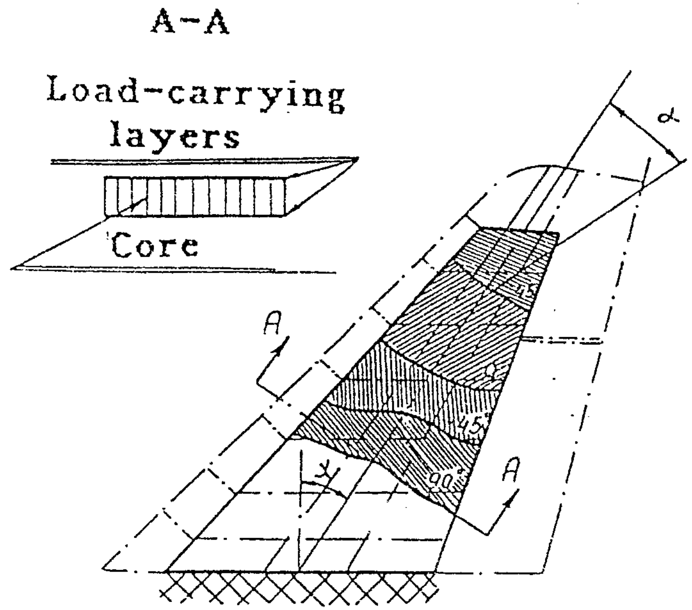
IV STAGE	
Aim	Structural analysis
Model	Detailed discrete model
Design variables	-
Require-ments	Check for all constraints and restrictions

Data flow in ARGON during aero/strength/dynamic designing of the composite lifting surfaces

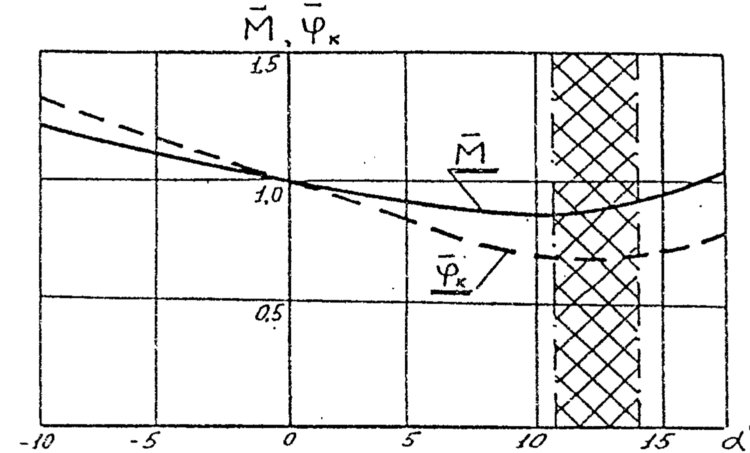


VERTICAL FIN OF A HIGH-PERFORMANCE AIRCRAFT

Level I model



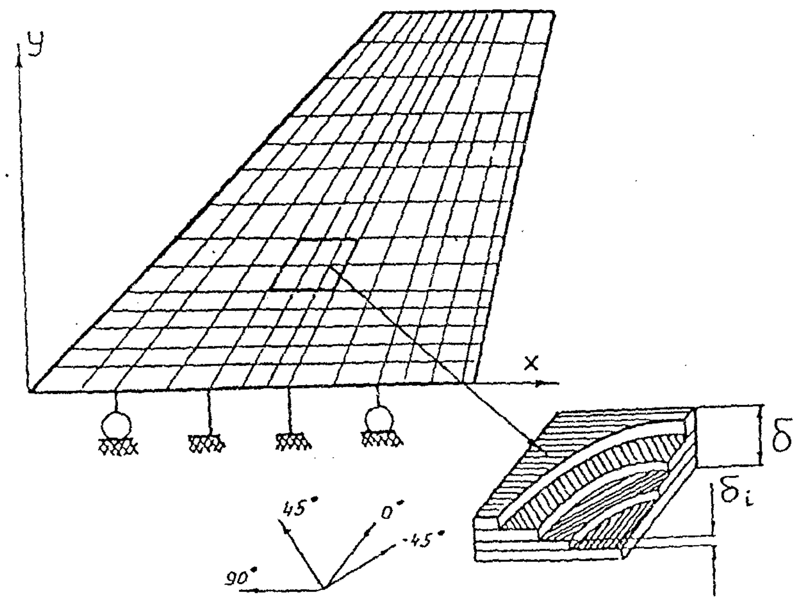
Mass M and fin tip rotation angle φ_K vs. laminate principal axis angle α



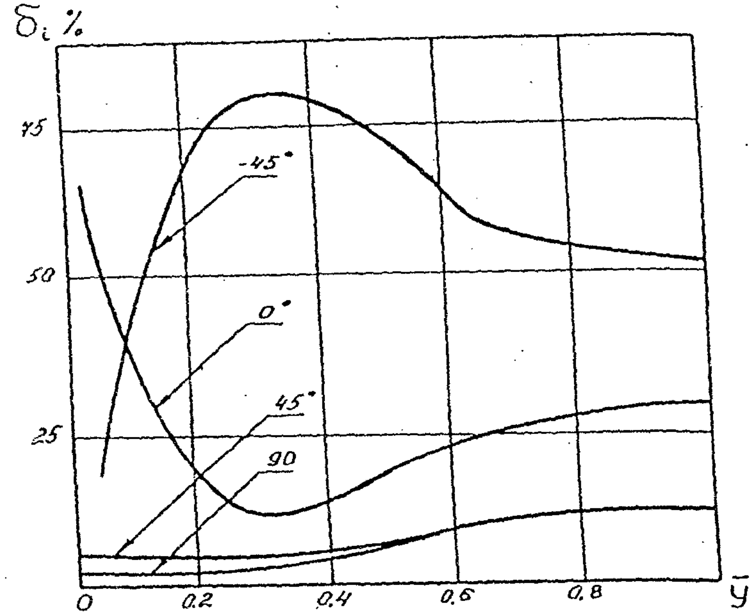
$$\bar{M} = \frac{M}{M_{\alpha=0}} \qquad \bar{\varphi}_K = \frac{\varphi_K}{\varphi_{K,\alpha=0}}$$

VERTICAL FIN OF A HIGH-PERFORMANCE AIRCRAFT

Level II model



Spanwise distribution of thickness



$$\bar{\delta}_i = \frac{\delta_i}{\delta}$$

COMPOSITE

PURPOSE

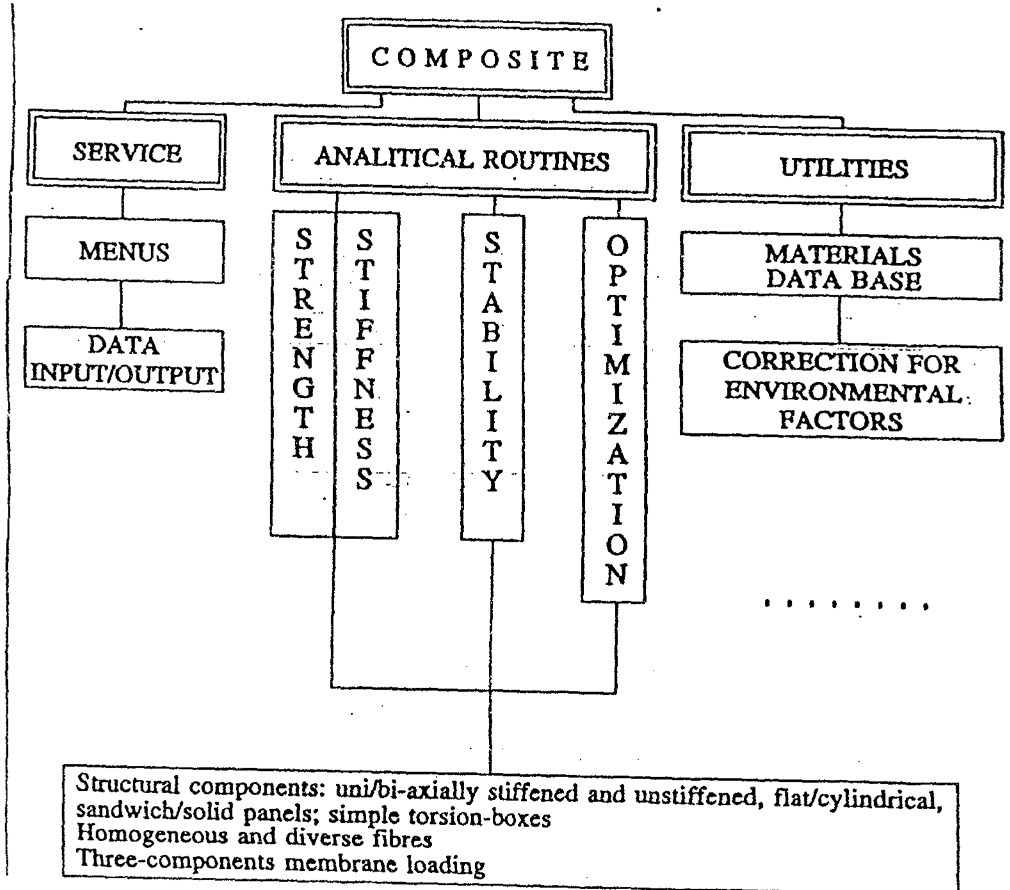
Automation of Strength Analysis at Stage of Designing Separate Composite Structures

LANGUAGE FORTRAN-77

AUTHORS

Mishulin I.B., Andrienko V.M., Zamula G.N., Sukhobokova G.P.,Ierusalimsky K.M.

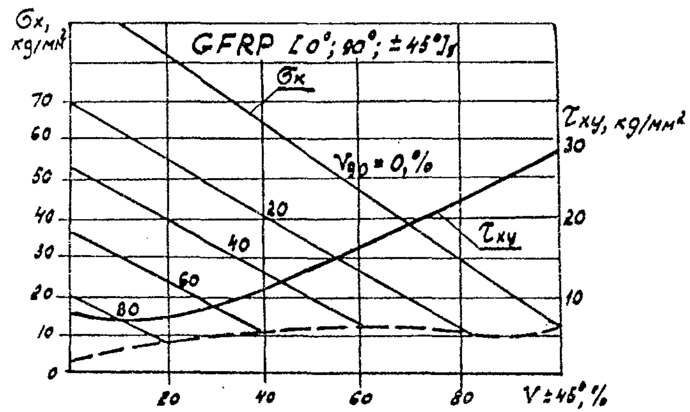
PACKAGE OF INTERRELATED PROGRAMS FOR ANALYZING THE COMPOSITE STRUCTURES



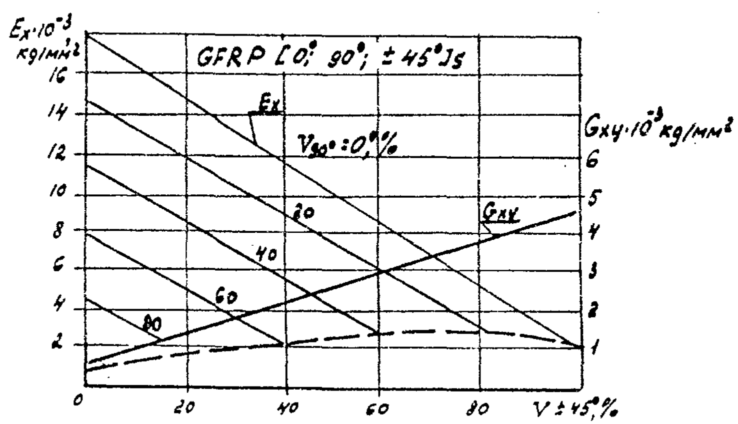
Interaction through multilevel menus

COMPOSITE

CALCULATION OF STRENGTH PARAMETERS

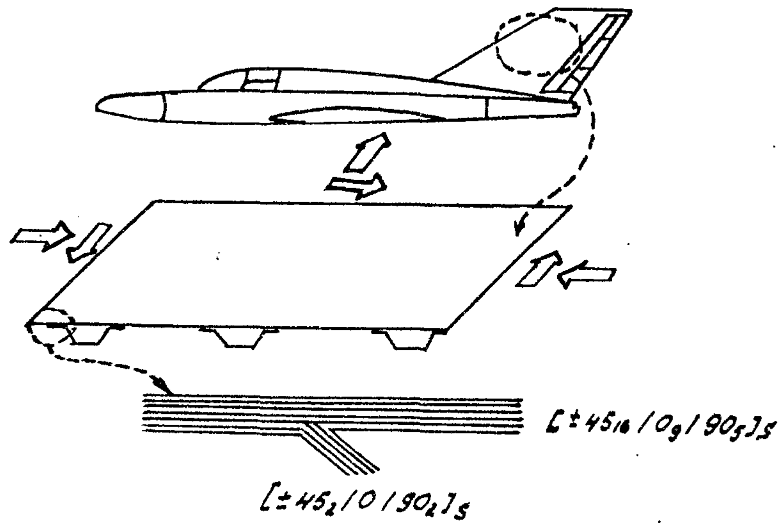


CALCULATION OF ELASTIC PARAMETERS



PACKAGE OF INTERRELATED PROGRAMS FOR ANALYZING THE COMPOSITE STRUCTURES

ANALYSIS OF STRUCTURAL MEMBERS



COMPOSITE is a convenient means for designers to use at their working stations.

OPTIMAL DESIGNING OF COMPOSITE

PANEL / SHELLS

Problem formulation

Initial data

Load:Pcompr
Geometry:R,L

Objective function

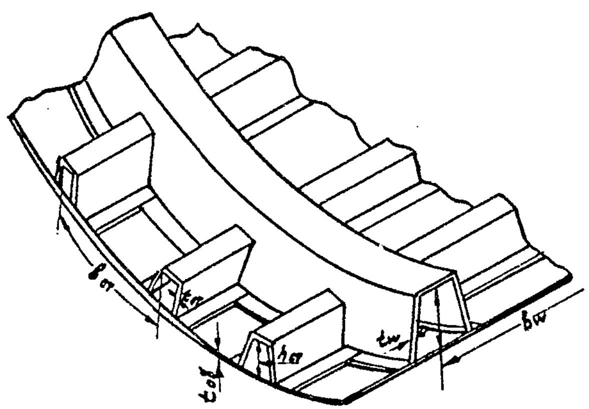
Min weight: G min

Constraints

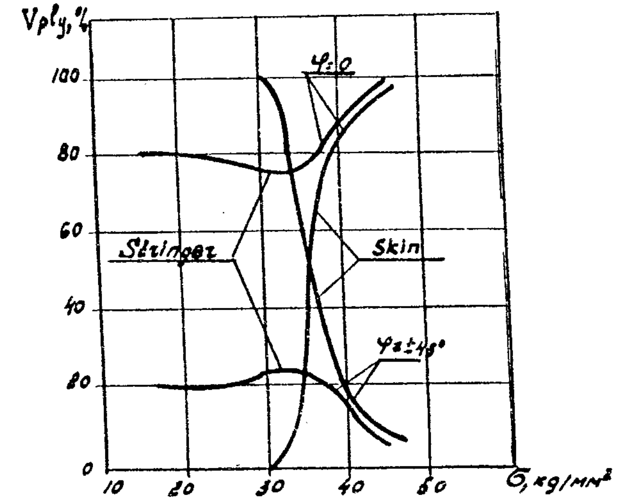
Stress:
$\sigma_i <= \sigma_b$; $\sigma_i <= \sigma$ crit
Manufacture:
(t_i / t_{lam})=integer

Design variables

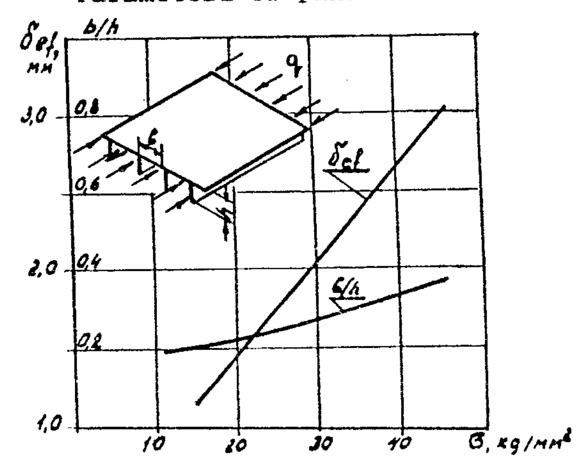
Detailed geometry
skin,web,str,fr,..

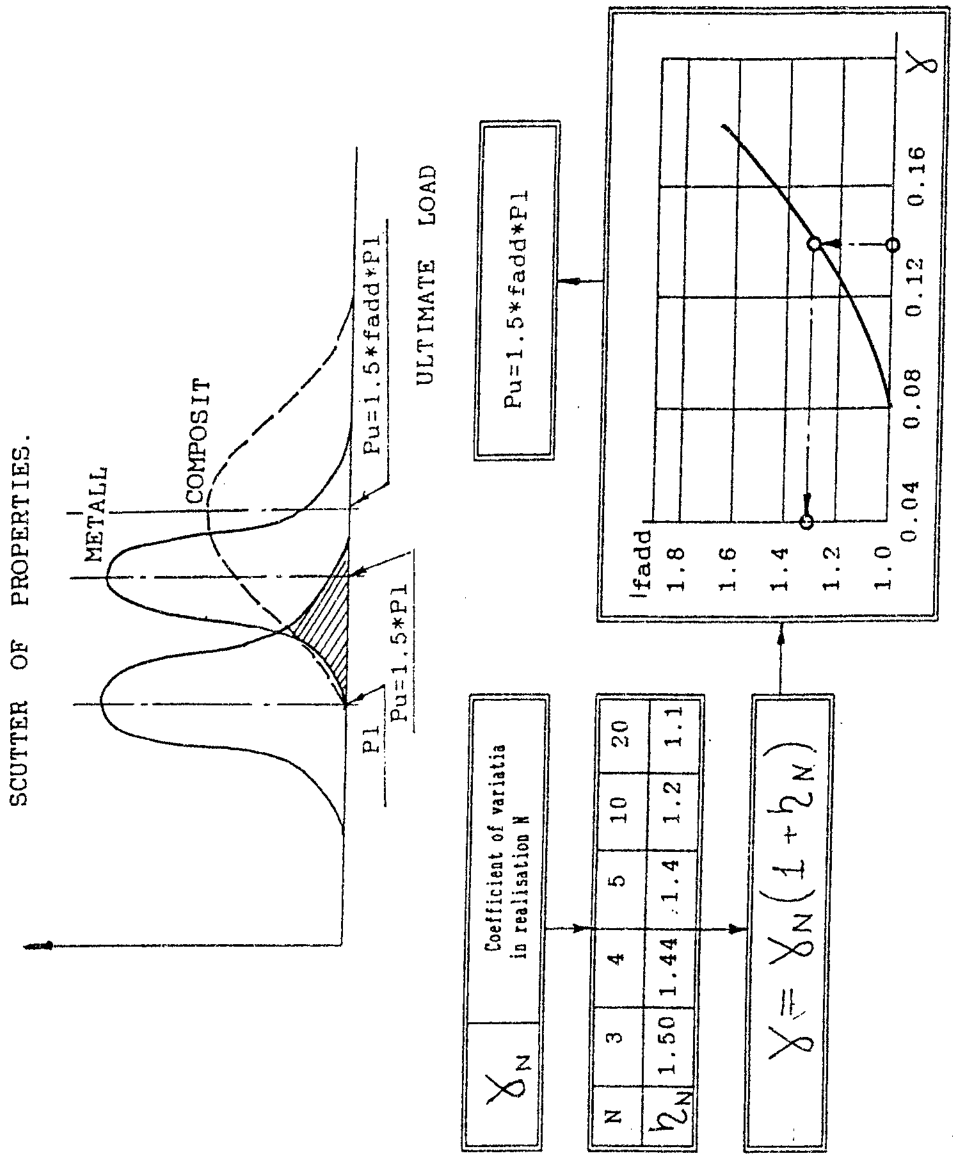


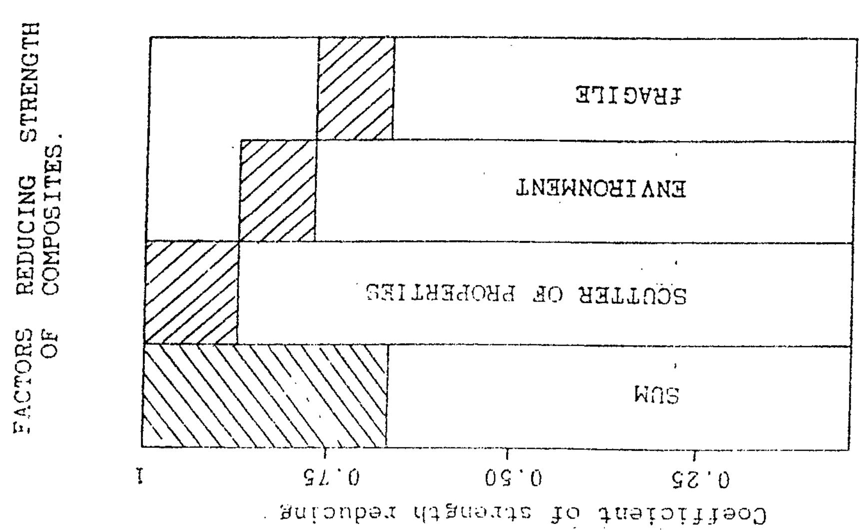
Type of material



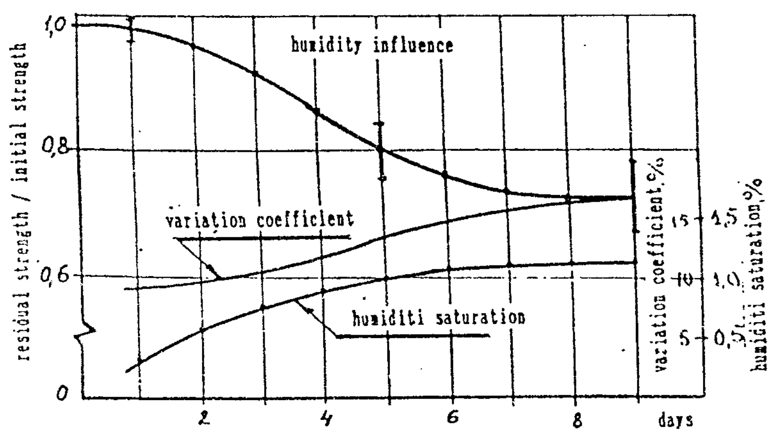
Parameters of panel member



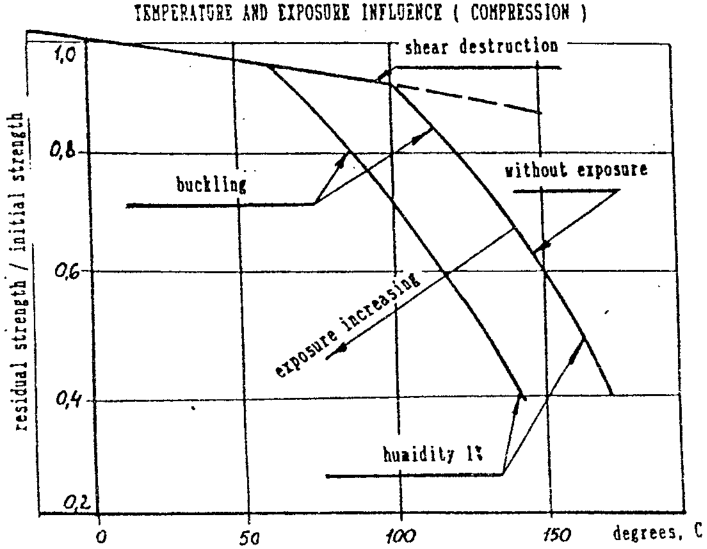

SCUTTER OF PROPERTIES.
METALL
COMPOSIT
P1
Pu=1.5*P1
Pu=1.5*fadd*P1
ULTIMATE LOAD
Pu=1.5*fadd*P1
fadd
1.0
1.2
1.4
1.6
1.8
0.04
0.08
0.12
0.16
γ
γN
Coefficient of variatia in realisation N
N 3 4 5 10 20
ηN 1.50 1.44 1.4 1.2 1.1
γ = γN(1 + ηN)


FACTORS REDUCING STRENGTH OF COMPOSITES.
FRAGILE
ENVIRONMENT
SCUTTER OF PROPERTIES
SUM
1
0.75
0.50
0.25
Coefficient of strength reducing

ENVIRONMENT INFLUENCE ON CARBONPLASTIC STRENGTH.

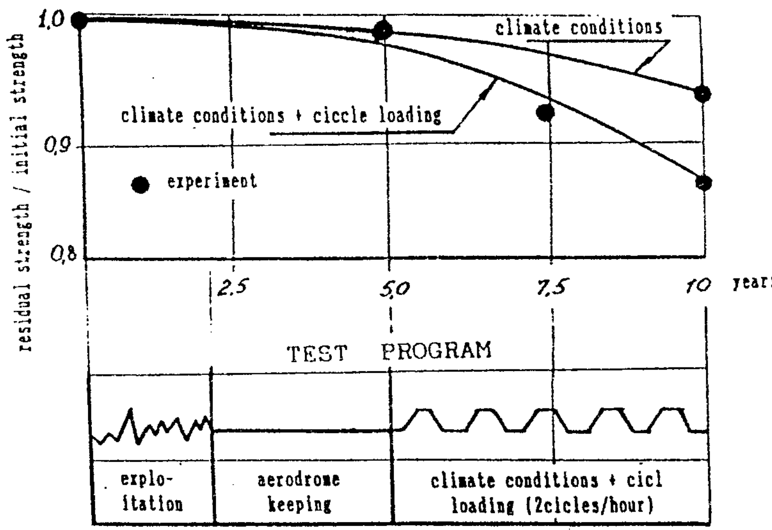


TEMPERATURE AND EXPOSURE INFLUENCE (COMPRESSION)



AIR - BRAKE

COMBINED ACTION OF ENVIRONMENT AND LOADS.



Ken < 1 - correction because of environment influence on design stress choice

Established on a basis of:
- analysis of exploitation experience,
- experiments.

CERTIFICATION OF COMPOSITE AIRFRAME STRUCTURES

CERTIFICATION APPROACH
- THEORETICAL STRTNGTH ANALYSIS,   -TESTS OF STRUCTURE SPECIMENS OF COMPLEXITY.

STATIC STRENGTH		
STRESS FIELD, LOAD CARRYING CAPABILITY DISPERSIONAL OF CHARACTERISTICS	DAMAGE TOLERANC	ENVIRONMENT INFLUENCK

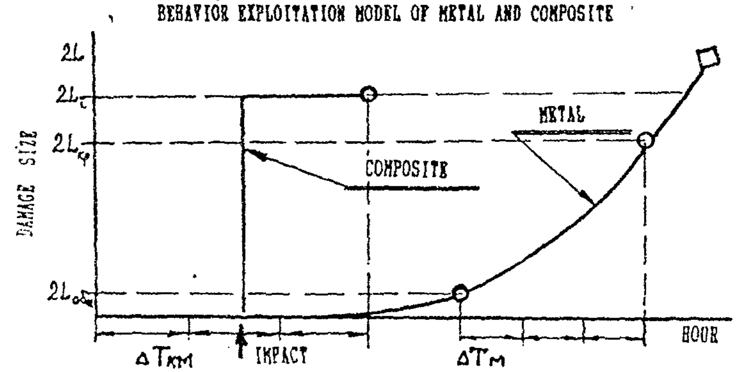
FATIGUE STRENGTH		
REPEATED STATIC LOADS DISPERSIONAL OF CHARACTERISTICS	DAMAGE INCREASE VELOCITY	ENVIRONMENT INFLUENCK

KONTROLABILITY (methods and technologies)

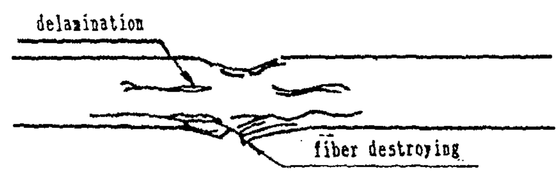
REPAIRABILITY (tipical repair technologies on exploitation		
STATIC STRENGTH	FATIGUE STRENGTH	ENVIRONMENT INFLUENCE

F R A G I L E D E S T R O Y I N G.

BEHAVIOR EXPLOITATION MODEL OF METAL AND COMPOSITE



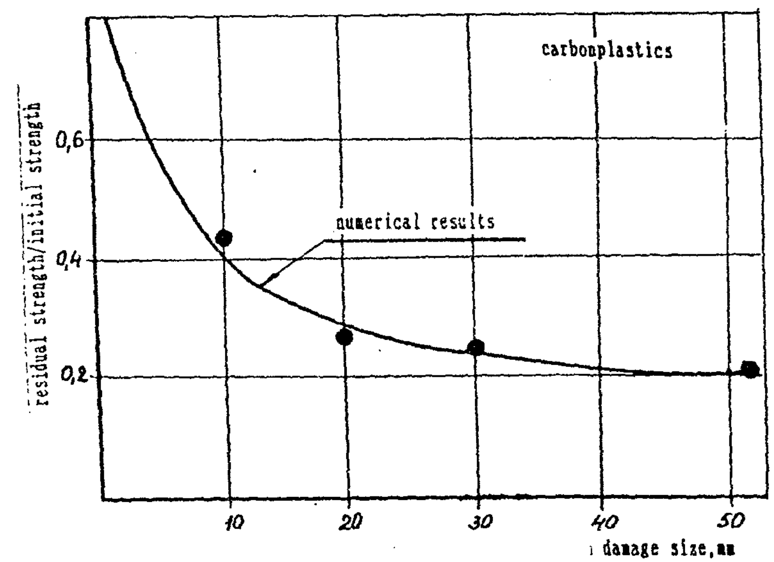
a) HIGH - ENERGY IMPACT.



b) MODERATE - ENERGY IMPACT.

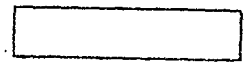
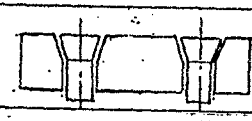
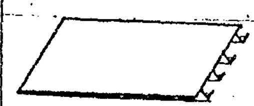
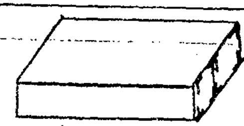
c) LOW - ENERGE IMPACT.

COMPRESSED PLATE STRENGTH DECREASE FOR IMPACT DAMAGE.



- DESIGN APPROACH - FAIL-SAVE STRUCTURE.
- CHOICE OF DESIGN STRESSES IN GIEW OF REGULATED DAMAGES (Kr.d <1).
- PROVISION WITH RELIABLE CONTRAL:
  - a. VISUALLY DURING EXAMINATION BEFORE FLIGHT.
  - b. INSTRUMYNTAL, NONDESTRUCTIVE DURING REGULATIVE EXAMINATION
- TYPICAL EXPLOITATION REPAIR TECHNOLOGIES DEVELOPMENT.

SPECIMEN TYPES FOR CERTIFICATION TEST.

N	TYPE	EXAMPLE /—— L ——/	L,M	NUMBER OF SPECIMENTS	PURPOSE	· CORRESPONDENCE DETERMINATION METHOD
1	COUPON		UP TO 0,3	LARGE	STRENGTH CHRACTERISTIES; MEAN ONES, DISPERSION, INFLUENCE OF EXPLOITATION CONDITIONS (HUMIDITY, TEMPERATURE IMPACT DAMAGES ETS.)	TESTS IN CONDITIONS WHICH ARE CORRESPONDED TO EXPLOITATION ONES.
	STRUCTURAL DETAIL		UP TO 1,0	LARGE		
3	SUB - COMPONENT		UP TO 3,0	MODERATE (5 - 10 pieces)	REAL STRENGTH AND FATIGUE CHARACTERISTIES. TECHNICAL SOLUTIONS CHECKING..	TESTS IN CONDITIONS WHICH ARE CORRESPONDED TO EXPLOITATION ONES. OR TEST IN NORMAL CONDITIONS WITH RECALCULATION TO EXPLOITATION ONES.
4	COMPONENT		UP TO 10,0	1 - 2 pieces	TOTAL CHECK OF ALL TECHNICAL SOLUTIONS AND PROCESSES RELATED TO AIRFRAME CREATION.	
5	FULL SCALE AIRCRAFT		UP TO 30,0	1 piece		TEST IN NORMAL CONDITIONS WITH RECALCULATION TO EXPLOITATION ONES.

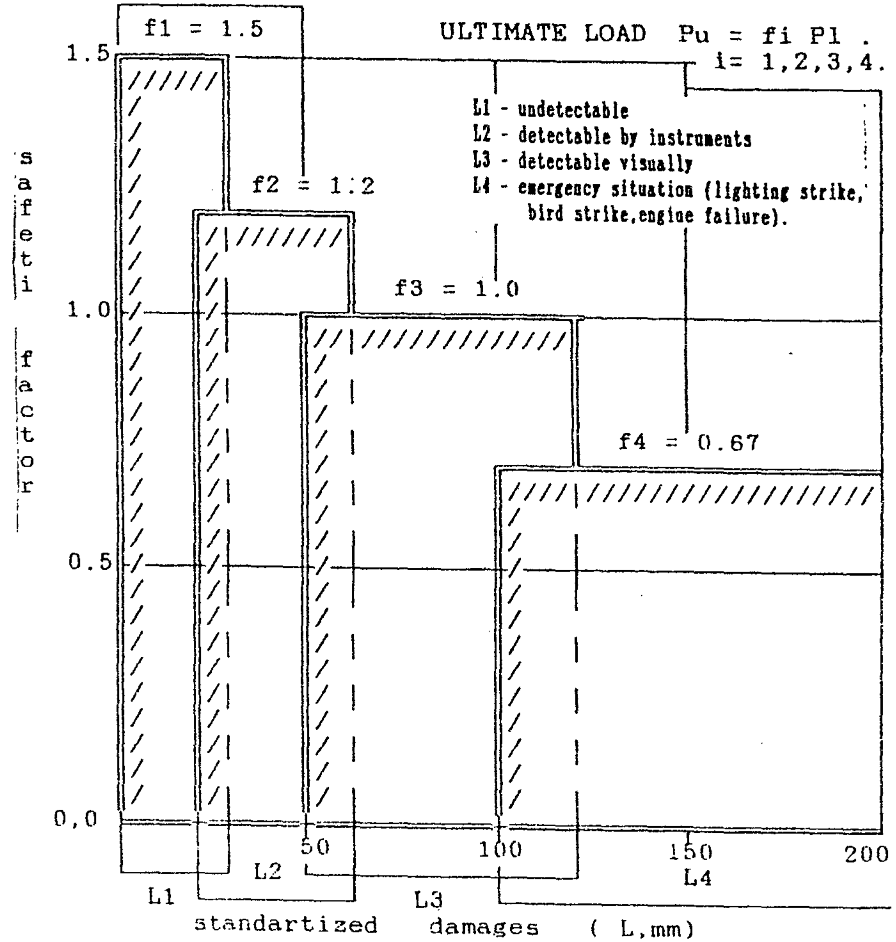
TEST FOR ALL SPECIMENT TYPES ARE OBLIGATORY PART OF AIRPLANE FLIGHT MAINTENANCE CERTIFICATION.

ENVIRONMENT INFLUENCE.

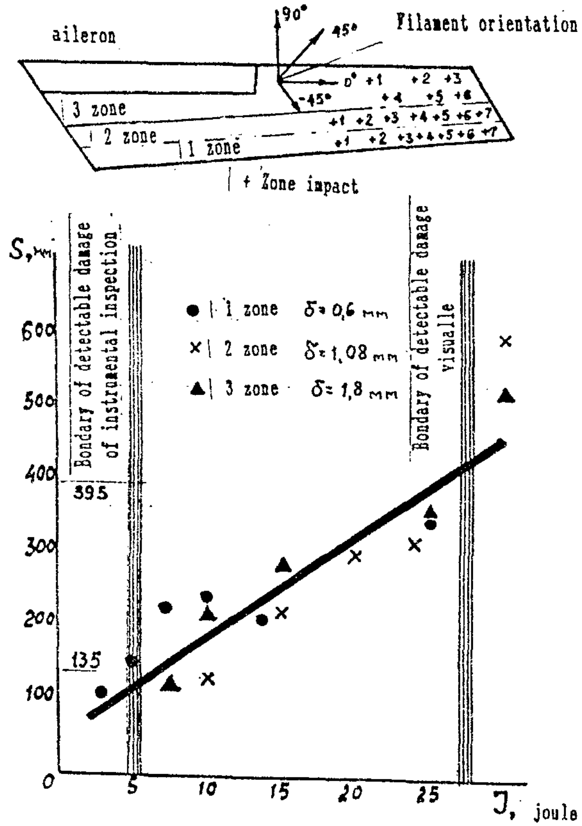
PARAMETERS / AIRCRAFT	EXTREME PARAMETERS Tmax, C	Tmin, C	Wmax, %.	DURATION OF EXPOSITION T, C	F, %
TU - 204	80	- 60	1.0	35	70
B - 737	82	- 56	1.0	21	70
A - 310	70	- 50	1.2	20	80
	STATIC TEST			FETIGUE	

W - moisture content level
F - humidity in exposition

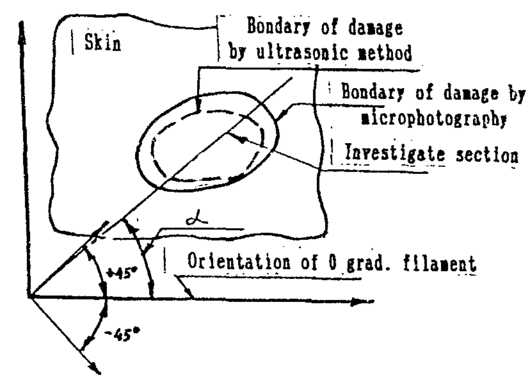
FAIL - SAFE DESIGN CONDITION.



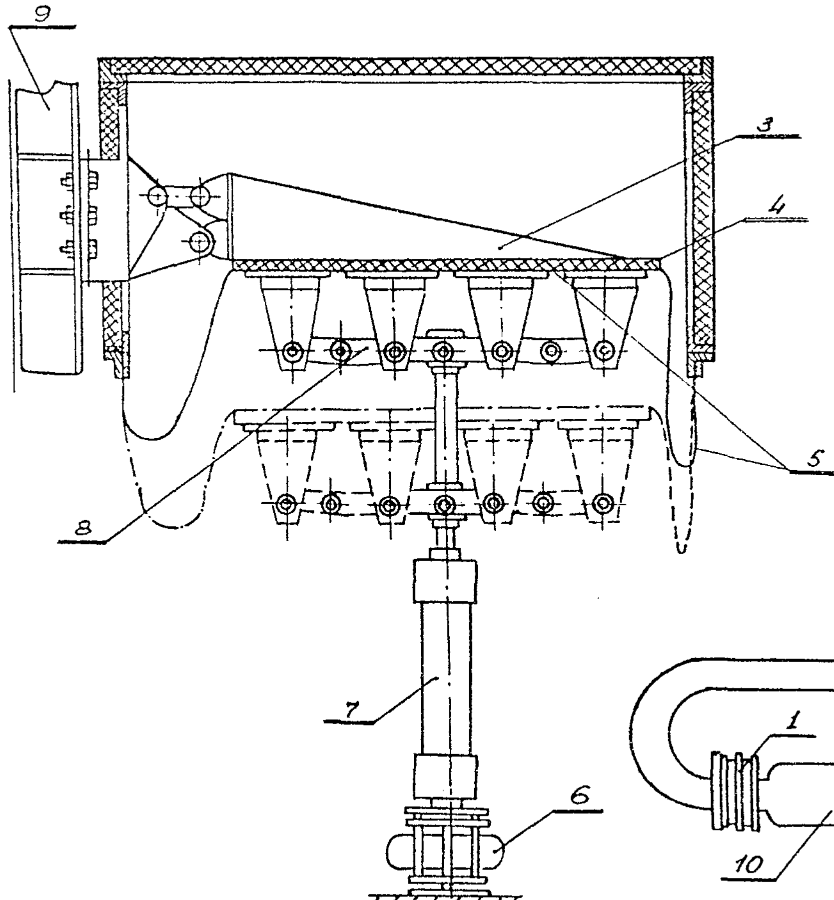
DAMAGE DETECTABILITY TEST.



VERIFICATION OF ACUSTIK METHOD.

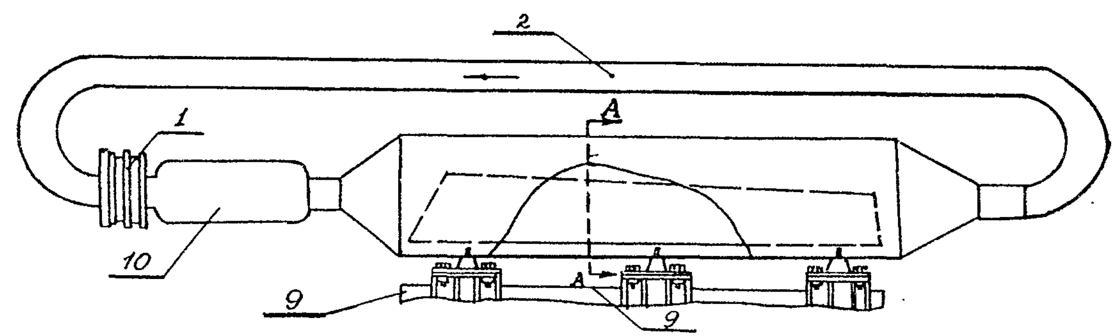


AILERON TEST STAND.

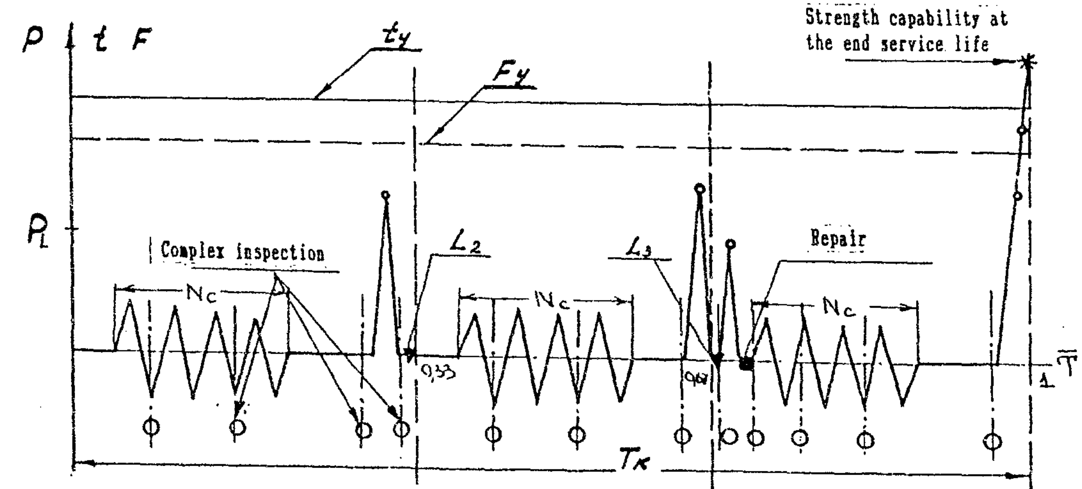


———— Membrane during loading.
– – – – Membrane during moisture accumulation and inspection.

1. Fan.
2. Return duct.
3. Aileron.
4. Rubber washer.
5. Elastic membrane.
6. Load cell.
7. Hydraulic force exciter.
8. Levers.
9. Support.
10. Steam generator.



E N D U R A N C E   T E S T I N G   P R O G R A M .

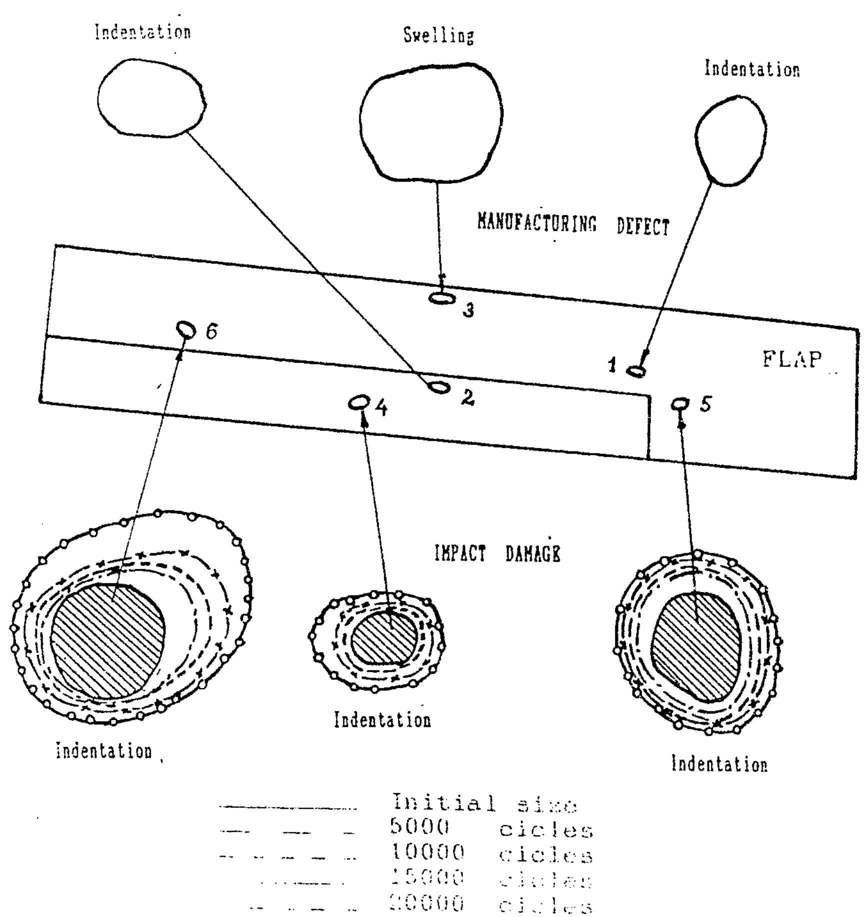


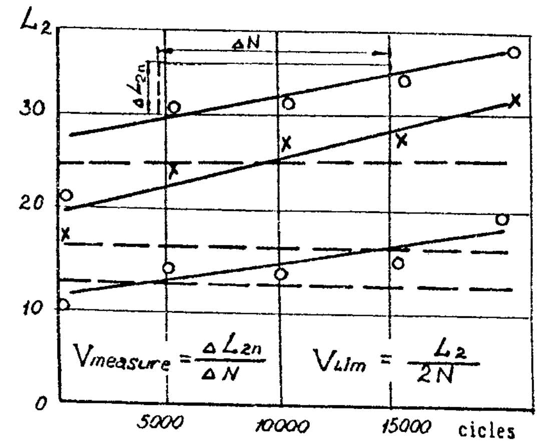
STAGE	1	2	3
DURATION	0,33 Tk	0,33 Tk	0,33 Tk
DAMAGE	L1	L2	1)L3 ,2)REPAIR
CYCLE NUMBER	Nc	Nc	Nc
STATIC LOAD	1.2 Pl	1.2 Pl	1)Pl,2)1.2Pl,3)1.5Pl
TEMPERATURE	ty	ty	ty
HUMIDITY	Fy	Fy	Fy

Tk - cervice life.
Pl - limit loada.
Nc - cicling loading for one service life.

L1 - undetectable.
L2 - detectable by instruments.
L3 - detectable visually.

GROWTH RATE OF DAMAGES AND DEFECTS





$$L_2 = 2\sqrt{\frac{1.4S}{\pi}}$$

N point	Type of damages	Damage growth rate, мм/c Measure	Limit
1	Indentation	0	40 E -4
2	Indentation	0	
3	Swelling	0	
4	Indentation	3.4 E -4	
5	Indentation	5.6 E -4	
6	Indentation	5.9 E -4	

V measure V limit

CERTIFICATION OF PREFORMS AND STRUCTURE ELEMENTS OF COMPOSITE MATERIALS

V.L.Raikher, V.Ya.Senik and Y.P.Trunin
TsAGI, Zhukovsky, Russia

Abstract

The cost-efficient approach to certification of preform materials and structural elements is suggested. The approach relies on static and fatigue tests of specimens from one material batch that has the worst average properties. The per ply thickness effect on strength and fatigue of polymer composite materials lies in substantiation of the certification approach.

It is known that a great number of specimens manufactured from several prepreg batches is necessary to test for estimation of the design values of material, structural element and structural detail. For example, less than five batches cannot be enough for A- and B-value estimation of material properties [1]. If the question about evaluation of design values of preform materials, elements and details of aircraft structure with such degree of confidence is raised, then test extent becomes excessive, because the number of elements and details that influence the flight safety is too great. Therefore, the known suggestion about tests of one worst batch of preform material, element and detail from the acceptable range is the alluring prospect.

The selection of the worst material batch is suggested to be realized on the basis of the relations between fracture resistance and per ply thickness. These relations are established by the supplier or the user on the basis of strength and fatigue test results of lamina or laminate under tension and compression. These tests are better to be carried out in advance. For receiving the relations between strength and per ply thickness the test results of process control specimens may be also used. On the basis of relations received the user sets the allowable range for per ply thickness alteration in preform material, between-batch variance and critical per ply thickness that is equal to one from two range bounds. As the worst material batch having the worst average properties we suggest to consider the one produced with critical per ply thickness. The test specimens of preform material, element and detail (further called model specimens) intended for estimation of the batch mean and the within-batch variance should be manufactured with the critical per ply thickness or close to it. In the last case a per ply thickness deviation is taken into consideration during test result analysis.

The estimation of A- and B-values is suggested to be conducted only on model specimens that have been subjected to the environmental exposure and have been tested at the most critical temperature. The influence of different factors on strength and fatigue of model specimens is suggested to be evaluated on the basis of the averages.

The influence of per ply thickness upon strength and fatigue of composites is necessary to be considered for two aspects. At the stage of material and process development the optimum of per ply thickness and the allowable range of per ply thickness alteration are needed to determine the maximum of unit static and fatigue strengths that are equal to the ratio of static failure stress or fatigue failure stress under given life to unit weight of composite. In this case the static failure stress and the loads for fatigue tests should be calculated under the actual thickness in the specimen failure area.

The second approach to the estimation of per ply thickness influence on a composite failure resistance is proper for the structure certification stage. In this case the static failure stress and the loads for fatigue tests should be calculated under the nominal per ply thickness under which the analysis of stresses acting in service have been conducted. The nominal specimen thickness is equal to the nominal per ply thickness multiplied by the ply number ($t_n=p_n n$). Under application of the nominal per ply thickness the static and fatigue failure loads will be directly proportional to the corresponding stresses. The relations of the static failure stress and life versus per ply thickness from the allowable range are required to establish the fit curves and the between-batch variances.

Most full investigation of per ply thickness influence have been performed with glass/epoxy [2]. The thickness of dry fabric of 8 harness satin weave construction is equal to 0.23 ± 0.02 mm. After deletion of the coating away from glass fibers by burning it out the fabric thickness becomes equal to $t_1 = 0.25$ mm. It is explained by that the coating promotes to sticking together the fibers. The lay-up type is [0]. The maximum pressure during the curing is equal to 0.8 MPa.

The life values for unnotched specimens of this material versus per ply thickness are given in fig. 1 [3]. The data are approximated by curve of $\lg N = -26.8 + 278p - 587p^2$. The unit stress amplitude is presented in fig. 2 versus per ply thickness. For comparison the curve for stress amplitude under nominal specimen thickness $p_n = 0.23$ mm is also demonstrated in fig. 2. It can be seen in the figure that maximum value of unit fatigue strength is attained under per ply thickness of

0.23 mm. The maximum nominal stresses under fatigue failures and, consequently, corresponding maximum loads take place under per ply thickness of 0.25 mm.

The fatigue tests of glass/epoxy with fabric having a thickness of $t_1=0.3$ mm have shown that the maximum unit fatigue strength is attained under per ply thickness of $p=0.3$ mm. Thus, under $p<t_1$ when a compression of fabric sheets occurs the glass/epoxy resistance to failure decreases.

Carbon/epoxy test results seem to have the same view. The mean values of tension strength at the nominal per ply thickness of $p_n=0.157$ mm for unnotched carbon/epoxy specimens versus per ply thickness are presented in fig. 3. Unlike the above mentioned glass/epoxy the per ply thickness decrease has been provided by the pressure increase from 0.1 MPa up to 1 MPa. The thickness of dry carbon tape is equal to $t_1=0.157$ mm. It can be seen that the composite strength decreases when $p<t_1$.

Rejection of elements with poor strength promotes to appearance of the truncated distribution. The empirical truncated distribution that is plotted by the batch averages of the process control specimens for the accepted elements of acrobatic aircraft is presented in fig. 4. The truncated distributions can also appear when the per ply thickness limitations are introduced for specimens. The empiric distributions for logarithm of life are presented in fig. 5 for the above mentioned glass/epoxy with different ranges of per ply thickness alteration.

The Weibull formula fits for approximation of distribution of logarithm life for specimens with the most large range of per ply thickness alteration of $p=0.18$—0.27 mm. The narrowing of per ply thickness range down to $p=0.19$—0.27 mm provides lognormal distribution. Some more narrowing of per ply thickness range leads to the appearance of truncated lognormal distribution.

Experience of the analysis and experimental strength studies of composite structures with stress concentrators such as cutouts, mechanical joints, delaminations and impact damages has shown that characteristics of the fracture toughness can be used to analyze structure strength. In the case of fracture under plane stress state the critical stress intensity factors (SIF) under tension, compression and shear are fracture criteria and consequently design characteristics. The critical values of SIF for carbon/epoxy laminates with close layups versus a failure stress of unnotched specimens under tension and compression are presented in fig. 6. As it can be seen, there is nearly linear relation between these characteristics. The tension test results of glass/epoxy flat specimens with a hole having a diameter of $D=5$ mm (fig. 7) substantiate that their ratio to the unnotched specimen strength does not practically depend on per ply thickness.

If fatigue curves were plotted at the fixed loading rate rather then fixed loading frequency, then these curves are well approximated by the exponent function with constant exponent, Fig.8.

The time-effective procedure for *A*- and *B*-values estimation is based on the application of test results of lamina and/or laminate and/or process control specimens from different material batches and also on test results of model specimens from one material batch. In addition the regularities received on the basis of generalization of investigation results of strength and fatigue of composite coupons and elements are considered. On the basis of these regularities the following assumptions are made:

1. The variance model satisfies the requirements of the one-way variance model with balanced random effect. This model presents each observation as the sum of three components

   $x_{ij} = \mu + b_i + \varepsilon_{ij},\ j = 1,\ldots,n.$

   μ is the overall average of the population. The error terms ε_{ij} are assumed to be independently distributed normal random variables with a mean of zero and a variance of S_e^2 (within-batch variance). It is considered that the variance S_e^2 is the same for all batches. The random effect b_i is assumed to be independently distributed normal random variable with a mean of zero and a variance of S_b^2 (between-batch variance). If the large preform number was rejected on the basis of the test results of process control specimens, then the distribution of b_i describes by truncated normal distribution (fig 4).
2. The ratio of static (fatigue) strength values for the process control specimens and for the model specimens does not depend upon per ply thickness (fig. 7).
3. In accordance with assumption 2 it follows that the between-batch variation coefficients of static strength (fatigue strength) for the process control specimens and the model specimens are the same.
4. The fatigue curves for the element can be described by power function with the constant exponent (fig. 8).
5. A composite element life has the two-parameter lognormal distribution.

Taking into account of the limited extent of certification tests, the parameters that are necessary for computation of *A*- and *B*-values by this method are estimated step by step in the process of test result analysis of the process control specimens and the model specimens.

The test results of process control specimens are used for validation of some assumptions and definitions: relationship "nominal failure stress — per ply thickness"; empirical distribution of failure stress means; between-batch variance; overall per ply thickness average of the population of taken batches.

The linear regression method is used for coefficient computation in the relationship "failure stresses (or life) — per ply thickness" . All test results are used including the ones of the rejected preforms. The experience of analysis has shown that these relationships are approximated by the quadratic equation in the sufficient wide per ply thickness range.

On the results of certification static tests the mean values and the variation coefficients are calculated by standard procedures [4]. The variation coefficient of the fatigue limit is estimated as following

$$\gamma_e = \sqrt{\exp\left(\left(S_{\lg N} / m\right)^2 / 0.189\right) - 1}\,,$$

where $S_{\lg N}$ is the estimation of the standard deviation for logarithm life, m is the exponent of fatigue curve function.

The overall average of population for element strength (life) is estimated using the relationships of properties with per ply thickness and the assumption 2.

The *A*- and *B*-values are calculated as

$$A = \bar{\bar{X}} - T_A \sqrt{S_e^2 + S_b^2}\,,$$

$$B = \bar{\bar{X}} - T_B \sqrt{S_e^2 + S_b^2}\,,$$

where T_A and T_B are the tolerance limit factors.

If the distribution of the random effect b_i is normal, then the T_A and T_B values are determined in accordance with MIL-HDBK-17 [1]. In the case of the truncated normal distribution of the random effect b_i the statistical modeling procedure is used for determination of the T_A and T_B values.

The verification of suggested procedure performed using the data of Ref.[1] has showed that deviation of the received estimation of the *B*-values from the *B*-values presented in Ref.1 does not exceed 5%.

The certificate of preform material and element contains the following sections:

1. Short description of preform and element (trade-marks of applicable composites and other materials, aircraft component, design values).

2. Input materials for preform processing trade-marks of input materials, fabric or tape thickness).
3. Processing and main parameters of process (pressure, temperature, curing time).
4. Preform batch for certification (natural or model preform, copy number, geometry, layup, average and variation coefficient of per ply thickness, nominal per ply thickness).
5. Batches of model specimens and process control specimens for certification (specimen type and property, fabrication, number).
6. Acceptance criteria for strength of process control specimens (calculated at nominal per ply thickness) and for per ply thickness.
7. Physical properties of preform material (unit mass, fiber volume, void content, equilibrium moisture content at given relative humidity and temperature, actual moisture content in model specimens, glass transition temperature of dry material and material with equilibrium moisture content).
8. Within-batch and between-batch variation coefficients for analysis of A- and B-values.
9. Table of properties (specimen type, property, specimen state, test temperature, batch average, within-batch variation coefficient, specimen number, A- and B-values).

All sections of the certificate should be provided with documentation references.

References

1. Military Handbook "Polymer Matrix Composites", MIL-HDBK-17, 1992.

2. Трунин Ю.П. "Исследование выносливости стеклотекстолитов для лонжеронов лопастей несущего винта вертолета", кандидатская диссертация, 1973.

3. Трунин Ю.П. "Влияние плотности упаковки стеклоткани и условий нагружения на выносливость стеклотекстолита", Труды ЦАГИ, выпуск 1417. Изд. отдел ЦАГИ, стр.114-120, Москва, 1972.

4. Brownlee K.A. "Statistical Theory and Methodology in Science and Engineering", John Wyley and Sons, INC., New York — London — Sidney, 1965.

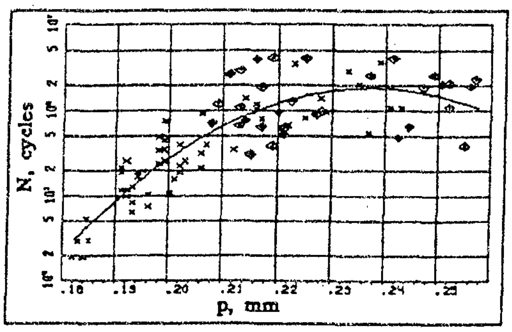


Fig. 1. Life of glass/epoxy unnotched specimens cut from sheets (small points) and helicopter rotor blade spars (large points) versus per ply thickness. Cycle stress under actual specimen thickness are σ_m = 100 MPa and σ_a = 120 MPa.

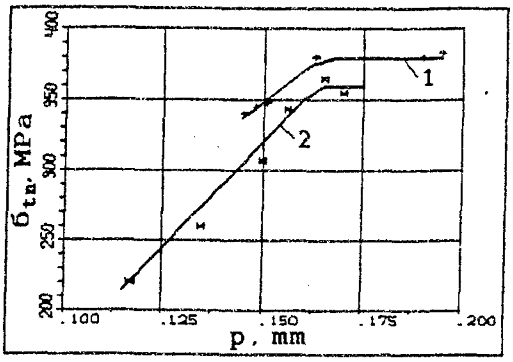


Figure 3. Variation of average tension failure stress under nominal per ply thickness of p_n = 0.157 mm for carbon/epoxy on the Lu-3 tape (1) and the Lu-3P tape (2) with per ply thickness. Layup type is [0/90].

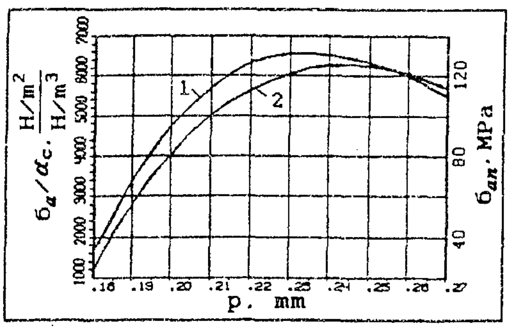


Fig. 2. Variation of unit stress amplitude (1) and stress amplitude under nominal thickness (2) of glass/epoxy unnotched specimens under N = 10^6 cycles and stress mean of σ_m = 100 MPa with per ply thickness.

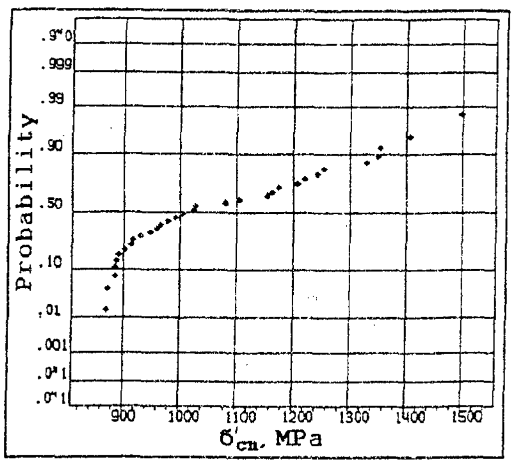


Figure 4. Empiric distribution of batch average values of nominal failure stress in carbon/epoxy longitudinal plies of hybrid composite (small points in Figure 6)

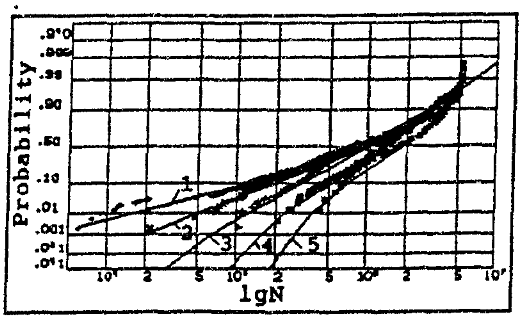


Figure 5. Empiric distribution of life logarithm for glass/epoxy unnotched specimens tested under the actual stresses of σ_m = 100 MPa and σ_a = 120 MPa.
1 - p =0.18 - 0.27 mm, n=270 specimens
2 - p =0.19 - 0.27 -"- 253 -"-
3 - p =0.20 - 0.26 -"- 213 -"-
4 - p =0.21 - 0.25 -"- 212 -"-
5 - p =0.22 - 0.24 -"- 45 -"-

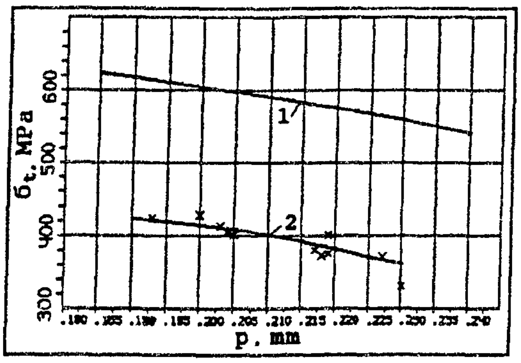


Fig. 7. Actual tension failure stress (brutto) of glass/epoxy flat unnotched specimens (1) and specimens with hole of d_0 = 5 mm and by width of 25 mm (2) versus per ply thickness.

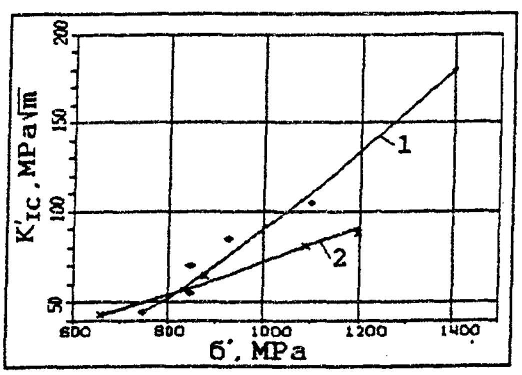


Fig. 6. Critical values of SIF for carbon/epoxy under tension (1) and compression (2) versus strength of unnotched specimens. All values are evaluated in longitudinal carbon/epoxy plies.

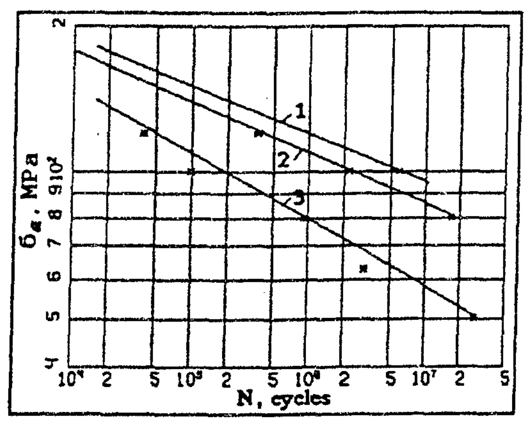


Fig.8.Comparison of predicted and experimental values of fatigue life of glass-fabric/epoxy specimens with central hole under uniaxial loading, d=8 mm,

B=50 mm, A_l=5.47 mm, $\dot{\sigma}$ =4500 MPa/s, σ_m=100 MPa (net);
1) p = 0.23 mm, σ_t=580 MPa, $m(r=O)$ = 14.8, s = 0.024;
2) p = 0.21 mm, σ_t=600 MPa, $m(r=O)$ = 13.5, s = 0.024;
3) p = 0.195 mm, σt=615 MPa, $m(r=O)$ = 10.4, s = 0.03.

TESTS AND CERTIFICATION OF AIRCRAFT METALLIC MATERIALS AT THE STAGE OF THEIR DEVELOPMENT

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QUALITY CONTROL SYSTEM OF AVIATION MATERIALS Q C S A M

QCSAM ESTABLISHES REQUIREMENTS TO THE DEVELOPMENT, CERTIFICATION, TESTING, APPLICATION AND ALSO TO THE PRODUCTION TECHNOLOGY AND MATERIALS TREATMENT.

QCSAM REGULATES:

— PLANNING THE SEARCH AND DEVELOPMENT OF NEW MATERIALS;

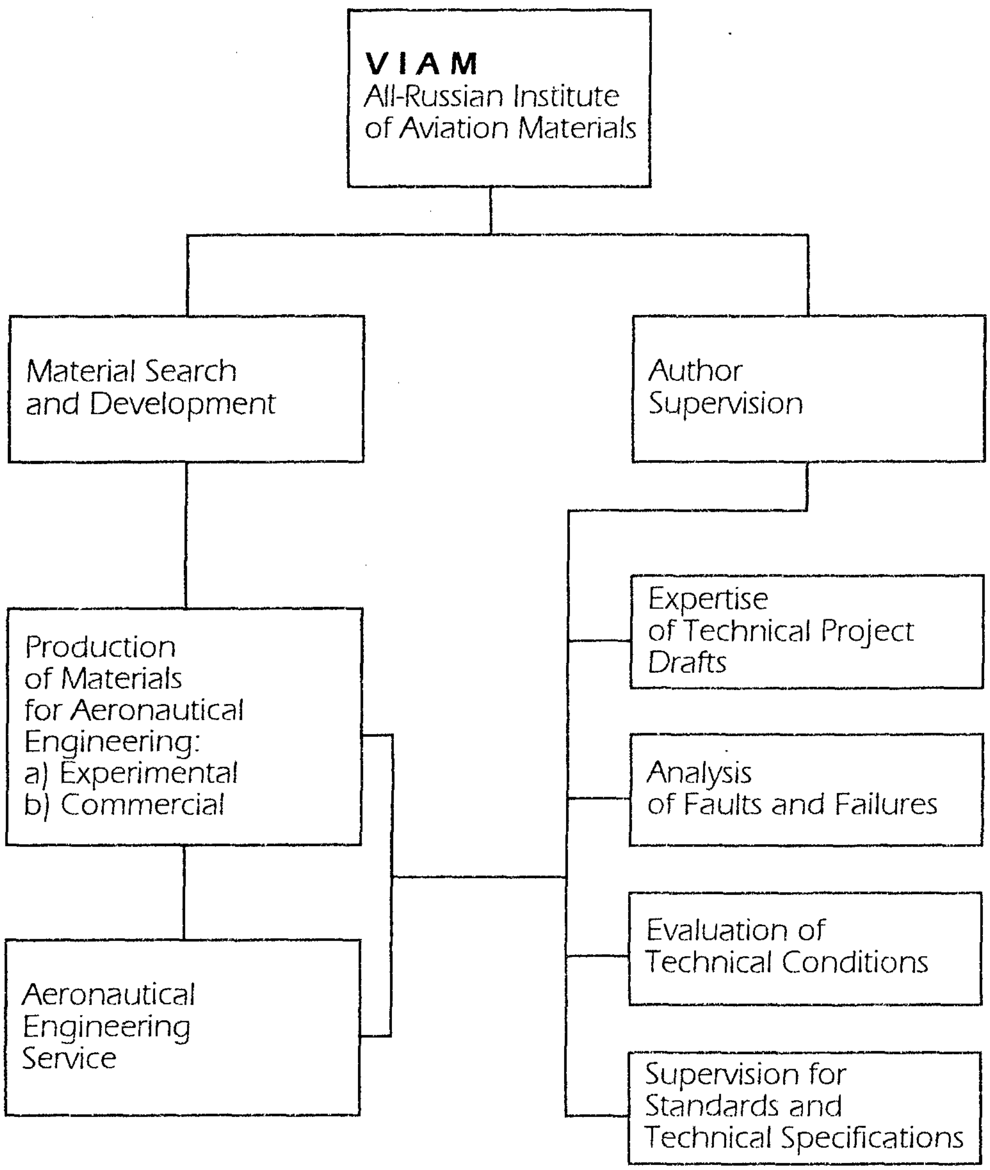
— PASSPORTIZATION OF NEW MATERIALS;

— PUTTING NEW MATERIALS INTO THE PRODUCTION;

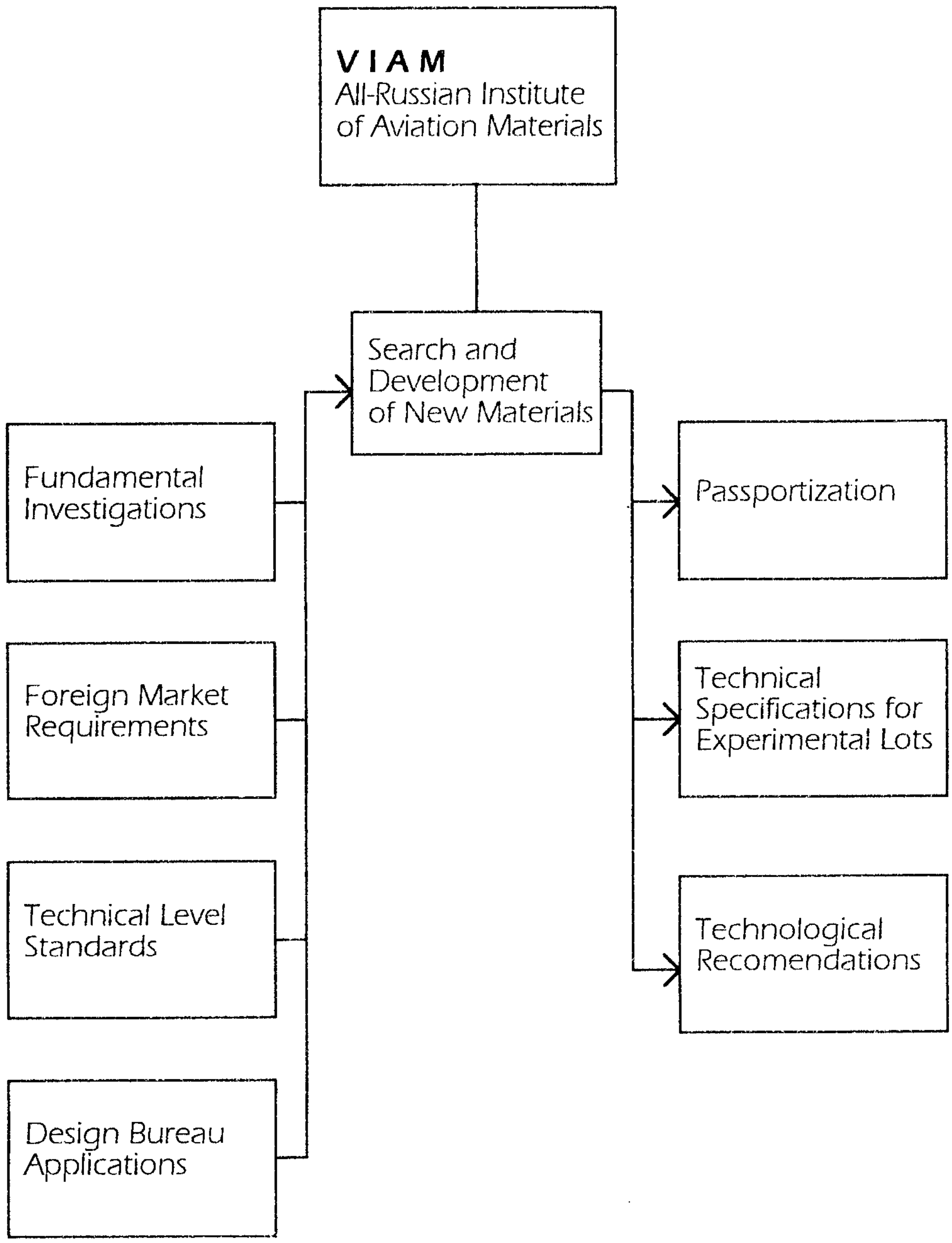
— CERTIFICATION OF NEW MATERIAL SEMIFINISHED PRODUCTS, THEIR PRODUCTION, CONTROL AND TESTING;

— THE OF APPEYING NEW MATERIALS;

— DEVELOPMENT OF THE NORMATIVE — TECHNICAL DOCUMENTATION FOR METHODS, MEANS, VOLUME AND THE ORDER OF CARRYING OUT THE CONTROL OF SEMIFINISHED PRODUCTS CHARACTERISTICS DURING THEIR PRODUCTION.



QUALITY CONTROL SYSTEM OF AVIATION MATERIALS



QUALITY CONTROL SYSTEM OF AVIATION MATERIALS

96% OF ALL THE MATERIALS, USED FOR AIRCRAFT-BUILDING OF RUSSIA HAVE BEEN DEVELOPED BY VIAM

VIAM DEVELOPS:
- STEELS;
- ALUMINUM ALLOYS;
- TITANIUM ALLOYS;
- MAGNESIUM ALLOYS;
- SUPERALLOYS;
- COMPOSITE AND HYBRID MATERIALS;
- GLASSES;
- RUBBERS, SEALANT, ADHESIVES, DYES AND OTHER NONMETALLIC MATERIALS.

GENERAL POSITIONS

ALL STRUCTURAL MATERIALS, USED IN AERONAUTICAL ENGINEERING, SHOULD BE PASSPORTIZED.

PASSPORTIZATION IS THE FINAL STAGE OF VIAM WORK, WHEN DEVELOPED THE NEW MATERIALS AND ALSO EVALUATING THE MATERIALS OF OTHER DEVELOPERS WITH THE AIM OF THESE MATERIALS APPLICATION SUBSTANTION IN AERONAUTICAL ENGINEERING.

100% OF THE MATERIALS, USED IN AERONAUTICAL ENGINEERING HAVE BEEN PASSPORTIZED.

GOALS OF PASSPORTIZATION

1. **PASSPORTIZATION** IS THE CONSTITUENT PART OF DEVELOPING NOVEL MATERIALS WITH THE GIVEN PROPERTY LEVEL.

2. OBTAINING THE SET OF DATA ABOUT THE MATERIAL, WHICH IS NECESSARY AND SUFFICIENT FOR TAKING THE DECISION CONCERNING ITS USE IN AERONAUTICAL ENGINEERING.

3. REVEALING THE MATERIALS DISADVANTAGES AT ITS DEVELOPMENT STAGE.

PASSPORTIZATION FUNCTIONS

1. DETERMINATION OF THE MATERIAL DEFINING CHARACTERISTICS — COMPOSITION AND STRUCTURE.

2. DETERMINATION OF THE WHOLE SET OF PROPERTY AND QUALITY PARAMETERS OF THE MATERIAL.

3. FIXATION OF THE BOUNDARY CONDITIONS OF THE MATERIAL PRODUCTION TECHNOLOGY.

4. ANTICORROSION AND EROSION PROTECTION REGULATION.

5. FORMATION OF THE MATERIAL USAGE CONDITIONS IN AERONAUTICAL ENGINEERING.

6. CONFIRMATION OF THE MATERIAL READINESS FOR PILOT PLANT PRODUCTION AND SUBSTANTIATION OF TECHNICAL SPECIFICATIONS REQUIREMENTS.

PASSPORTIZATION STAGES

1. **TECHNICAL TASK** (ON THE BASIS OF VIAM OR OTHER ENTERPRISES DEVELOPMENTS, APPLICATION OF DESIGN BUREAU.

2. **PASSPORTIZATION PROGRAM**, INCLUDING TYPES, VOLUMES AND TESTING TIME.

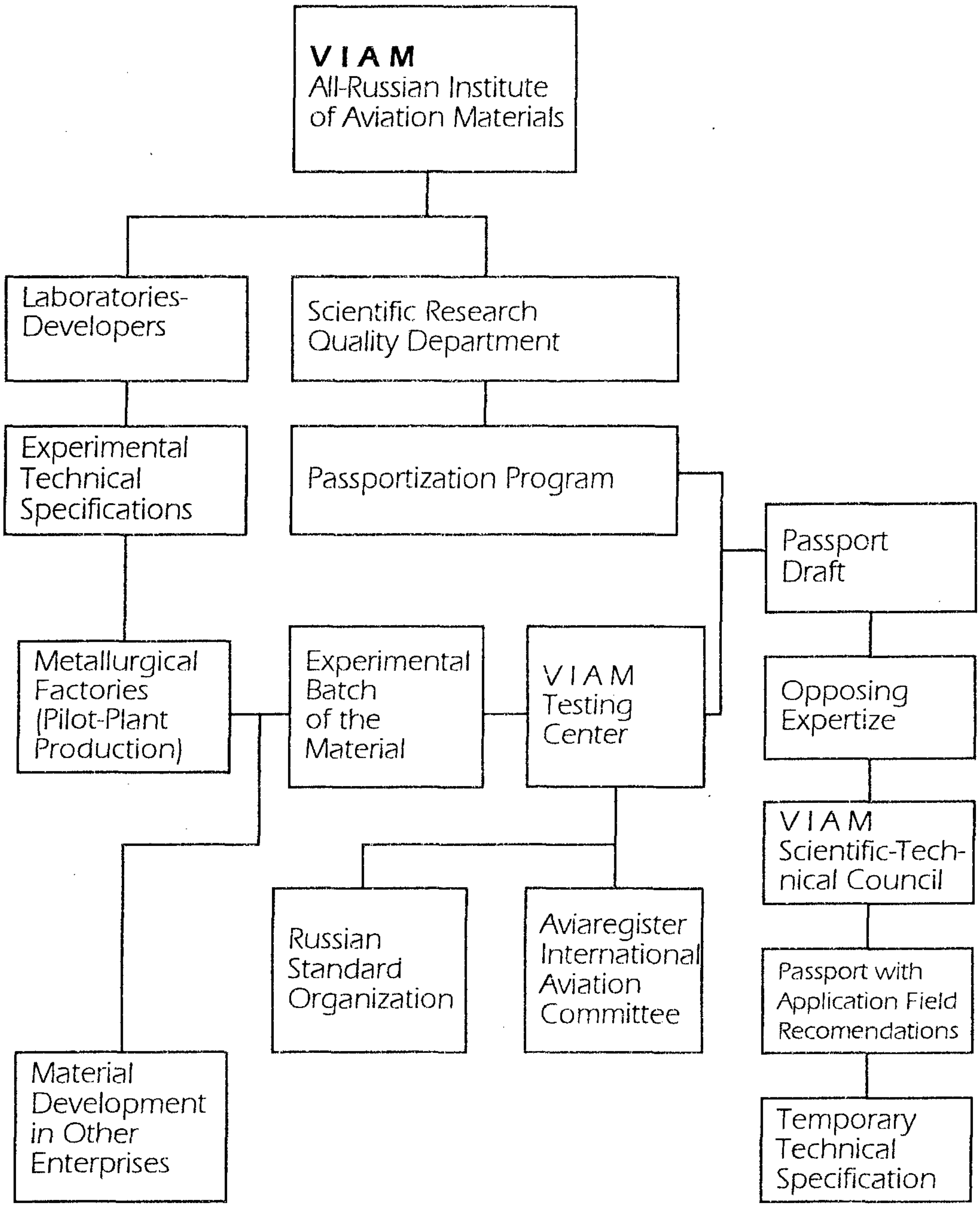
3. **CONTRACT** (EXPERIMENT, TECHNICAL SPECIFICATIONS) FOR THE MATERIAL DELIVERY FOR PASSPORTIZATION.

4. **MATERIAL STUDY** AT VIAM TESTING CENTER INDEPENDENT OF THE DEVELOPERS.

5. **PASSPORT DRAWING UP**.

6. **EXPERTISE** AND OPPOSING.

7. **PASSPORT APPROVAL** AT VIAM SCIENTIFIC-TECHNICAL COUNCIL. THE APPROVED PASSPORT ALLOWS MATERIAL TESTING IN COMMERCIAL PART AND ASSEMBLIES.



FUNCTIONAL INTERRELATION DURING PASSPORTIZATION

PASSPORT CONTENT

THERE ARE 21 PASSPORT FORMS AVAILABLE, INCLUDING 8 FORMS FOR METALLIC MATERIALS, WHICH ALLOWS TO ENVELOP THE WHOLE SPECTRUM OF THE MATERIALS, USED IN AERONAUTICAL ENGINEERING.

PASSPORT FOR STRUCTURAL METAL MATRIX COMPOSITE CONTAINS:

1. BRIEF ANNOTATION.
2. COMPARATIVE PARAMETERS WITH THE ANALOGIES.
3. CHEMICAL COMPOSITIONS.
4. RAW MATERIALS.
5. DESIGNATION BY SEMIFINISHED PRODUCT TYPES.
6. RECOMMENDED APPLICATION FIELDS.
7. MECHANICAL PROPERTIES.
8. WELDING JOINTS MECHANICAL PROPERTIES.
9. CORROSION RESISTANCE.
10. TECHNOLOGICAL PROPERTIES.
11. PHYSICAL PROPERTIES.
12. CONCLUSION OF VIAM SCIENTIFIC TECHNICAL COUNCIL.

EVALUATION CRITERIA AND PARAMETERS INCLUDED INTO PASSPORTS

1. STRENGTH PARAMETERS —

$\sigma_{0.2}$; $\sigma_{0.2}^{compr}$; $\sigma_{2\%}^{bearing}$; σ_B; $\sigma_B^{bearing}$; σ_B^n; E; τ_{aver}; σ_B^T; $\sigma_{0.2}^T$.

2. ENDURANCE AND DURABILITY PARAMETERS —

$\sigma_{\max}(N)$; $N_{LCF}(\sigma)$; σ_{-1}; σ_{-1}^n; σ_τ^T; $\sigma_{0.2/\tau}^T$; τ_σ^T.

3. FRACTURE CHARACTERISTICS —

K_C; K_C^{cond}; K_{1C}; $\sigma_{netto}^{friction}$; dl/dN.

4. THERMOPHYSICAL PROPERTIES —

α (1/degree) — THERMAL EXPANSION COEFFICIENT;

C (kJ/kg degree) — THERMAL CAPACITY;

λ (W/m • degree) — THERMAL CONDUCTIVITY.

5. CORROSION RESISTANCE PARAMETERS —

σ_{CR}; K_{1SCC}; $dl/d\tau$ (mm / hour);

ICC (INTERCRYSTALLINE CORROSION); EC (EXFOILIATION CORROSION); GK (GENERAL CORROSION).

6. TECHNOLOGICAL PARAMETERS — FORGEABILITY, WELDABILITY, CASTING PROPERTIES.

7. SPECIAL PROPERTIES — ADHESION, FRICTION COEFFICIENT, TOXICITY, FLAMMABILITY A. O.

8. ELECTROMAGNETIC PROPERTIES — CURRENT CONDUCTIVITY, ELECTRICAL RESISTANCE, MAGNETIC PROPERTIES.

9. TECHNICO-ECONOMICAL VALUES — RAW MATERIAL DEFICIT, WASTES UTILIZATION, YIELD, LABOUR CONSUMPTION FOR PARTS PRODUCTION.

REGULATION OF TEST TEMPERATURE LEVELS

1. FOR METALLIC ALLOYS, OPERATING AT NORMAL TEMPERATURES —

σ_B; $\sigma_{0.2}$; δ; KCU

ARE DETERMINING AT THE FOLLOWING TEMPERATURES: −70°C, +20°C, +80°C.

2. FOR THE ALLOYS, OPERATING AT CRYOGENIC TEMPERATURES —

σ_B; $\sigma_{0.2}$; δ; KCU

ARE DETERMINING AT THE FOLLOWING OPERATING TEMPERATURE RANGE: (−70°C, −130°C; −196°C).

3. FOR THE ALLOYS, OPERATING AT ELEVATED TEMPERATURES

σ_B; $\sigma_{0.2}$; δ

ARE DETERMINING IN THE OPERATING TEMPERATURES RANGE WITH THE INTERVAL:

25 — 50°C FOR ALUMINUM AND MAGNESIUM ALLOYS
50 — 100°C FOR TITANIUM ALLOYS AND SUPERALLOYS
100 — 300°C FOR REFRACTORY ALLOYS

MAXIMUM TEST TEMPERATURE SHOULD BE 25—100°C HIGHER, THAN THE RECOMMENDED ONE, IN ORDER TO REVEAL THE ALLOY CAPABILITY RESERVES.

4. STRESS RUPTURE AND ULTIMATE CREEP STRENGTH ARE DETERMINED AT 2−3 TEMPERATURES.

ADDITIONALLY THE STRESS RUPTURE OF NOTCHED SPECIMENS IS DETERMINED AT THE MAXIMUM OPERATING TEMPERATURE AND AT THE TEMPERATURE OF PLASTICITY DROP.

TEST VOLUME REGULATION

1. SPECIMENS FOR ALL TYPES OF SHORT — TERM TESTS

$(\sigma_{0.2};\ \sigma_{0.2}^{compr};\ \sigma_{2\%}^{bearing};\ \sigma_B;\ \tau_{aver};\ \delta;\ \psi;\ KCU;\ \sigma_B^T)$

ARE SELECTED OF NOT LESS, THAN 3 MELTS; IN THIS CASE NOT LESS, THAN 20 SPECIMENS ARE TESTED FROM EACH MELT.

2. ENDURANCE LIMIT $(\sigma_{-1};\ \sigma_{LCF})$ IS DETERMINED AT THE MAXIMUM AND ROOM TEMPERATURES AND FOR OF INTERMEDIATE TEMPERATURES, AND FOR NOTCHED SPECIMENS IT IS DETERMINED AT THE ROOM AND MAXIMUM TEMPERATURES.

LOW CYCLE FATIGUE IS DETERMINED ON THE BASE OF:

— $10^5 \ldots 2 \cdot 10^5$ FOR STRUCTURAL MATERIALS

— $5 \cdot 10^3$ FOR SUPERALLOYS.

LOW CYCLE FATIGUE FOR EACH TEMPERATURE LEVEL IS DETERMINED ON NOT LESS, THAN 15 SPECIMENS.

3. STRESS RUPTURE AND CREEP

$\sigma_\tau^T;\ \sigma_{0.2/\tau}^T$

ARE DETERMINED AT 2–3 TEMPERATURES, INCLUDING ALSO THE MAXIMUM RECOMMENDED ONE. NOT LESS, THAN 12 SPECIMENS ARE TESTED FOR EACH TEMPERATURE LEVEL.

4. CRACKING RESISTANCE CHARACTERISTICS K ARE DETERMINED AT THE ROOM TEMPERATURE AND FOR HIGH-STRENGTH STEELS THEY ARE DETERMINED ALSO $-70°C$.

THE LIST OF CHARACTERISTICS IS ESTABLISHED BY **QUALITY DEPARTMENT**, DEPENDING ON THE PART DESIGNATION AND SEMIFINISHED PRODUCT TYPE.

5. CORROSION TESTS ARE CARRIED OUT, DEPENDING ON THE MATERIAL SEMIFINISHED PRODUCT TYPE. THE VOLUME AND TEST CONDITIONS ARE SPECIFIED FOR EACH DEFINITE SEMIFINISHED PRODUCT.

RATIONING OF ADVANTAGE QUATUM (METALLIC MATERIALS)

N	PROPERTY	PARAMETER DESIGNATION UNITS	PROPERTY LEVEL	Quata not less than (%)
1	STRENGTH	σ_B $(\mathrm{kgf/mm^2})$	140	6
			100—140	8
			40—99	10
			25—39	10
		τ_{aver} $(\mathrm{kgf/mm^2})$	ALL VALUES	$5-10$
2	ENDURANCE AND DURABILITY	σ_{-1} $(\mathrm{kgf/mm^2})$	ALL VALUES	10
		$\sigma_{MAX}(N)$ $(\mathrm{kgf/mm^2})$		5
		N (cycle)	for high-strength materials	50
			for medium-strength materials	100
		σ_{τ}^{T} $(\mathrm{kgf/mm^2})$		$5-10$
		τ_{σ}^{T} (hour)	depending on the temperature level	$50-100$
3	FRACTURE CHARACTERISTICS	K_{1C}, K_C, K_C^{cond} $(\mathrm{kgf/mm^{3/2}})$	ALL VALUES	10
		dl/dN (mm / cycle)		10
4	CORROSION RESISTANCE	σ_{CR} $(\mathrm{kgf/mm^2})$	ALL VALUES	$5-10$
		K_{1SCC} $(\mathrm{kgf/mm^{3/2}})$		5
		$OK, \Delta\sigma_B, \Delta l$		40

DECISION OF V I A M SCIENTIFIC TECHNICAL COUNCIL

APPROVAL OF THE PASSPORT IS CARRIED OUT FOR NEWLY DEVELOPED MATERIAL WITH THE MATERIAL RECOMMENDATION FOR TESTING UNDER PRODUCTION SERVICE CONDITIONS WITH THE SPECIFICATION OF THE FLYING VEHICLE PARTS OR ASSEMBLIES FOR WHICH THE MATERIAL HAS BEEN DEVELOPED, AND THEIR OPERATING CONDITIONS.

TECHNICAL SPECIFICATIONS (TS)

1. ASSORTMENT AND ALLOWANCE FOR SIZES.

2. CHEMICAL COMPOSITION, LIMITATIONS FOR IMPURITIES AND CONTENT.

3. TECHNOLOGICAL CONDITIONS FAULTS ULTRA-SONIC CONTROL.

4. MECHANICAL PROPERTIES (min) — σ_B; $\sigma_{0.2}$; FOR DIFFERENT DIRECTIONS (L AND T).

5. TEMPORARY CONTROL FOR CRACKING RESISTANCE (K_{1C} and LCF) CAN BE USED FOR THE MOST CRITICAL SEMI-FINISHED PRODUCTS OF A GRADE.

TESTING EQUIPMENT
V I A M

1. **INSTRON** P_{max} = 5 tf, 10 tf, 20 tf.
   (Static tests) $T = -196$°C ... 450°C.
2. **MTS**; **810** MODEL.
   P_{max} = 5 tf, 25 tf, 50 tf, 100 tf, 250 tf.
   (ALTERNATIVE LOADING)
   Temperature cabinet: $T = -196$°C ... 350°C.
   High-temperature furnace: $T = 1000$°C.
3. **ZST 3/3** P_{max} = 3 tf, $T = 600$°C;
   P_{max} = 3 tf, $T = 1000$°C.
   VIETA (**ZST 2/3**)
   P_{max} = 3 tf, $T = 1200$°C.
   (STRESS RUPTURE AND CREEP).
4. SPECIAL TESTS TREATMENTS.

DETERMINATION OF METALLIC MATERIALS DESIGN CHARACTERISTICS KC AND CRACK GROWTH RATE ACCORDING TO AP-25 REQUIREMENTS

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The development of new Airworthiness Regulations AP-25 where fatigue strength ensuring for the airframe at the stages of design and operation is based on the damage tolerance concept required standardizing of design characteristics acquisition for the materials used for residual strength and crack growth rate evaluation. This report contains the approaches to form basic and design characteristics K_C (K_{1C} and K_{Cy}) and crack growth rate for metallic materials according to AP-25 requirements. The algorithms are presented for basic and design characteristics analysis in a case of necessary primary data lack.

The basis characteristics are understood as the average ones.

The design characteristics are determined from the basic ones by means of introducing appropriate factors. When the design characteristics are specified, the following things are used:

a). statistical parameters, characterizing material properties — average and scatter;
b). parameters considering the representativeness and the volume of input data.

Statistical characteristics K_{1C}, K_{Cy} and crack growth rate are evaluated from the assumption of normal distribution laws for K_C and K_{Cy} and lognormal distribution for crack growth rate (v). Here the requirements to the minimum volume of prepreg and specimen tested is no less than 3 items for prepregs and no less than 3 items of each prepreg — for specimens. The requirements mentioned are valid in the case of stable material production.

When there are less than three samples, scatter received S (S — scatter defined as root mean square deviation) can be corrected as follows. For the case of one sample from the assumption of S_m=S_b, where S_m and S_b are, respectively, inter-and in-sample scatter, the correcting factor is 1.4. Then using linear interpolation we get

one sample — $S_{anal} = 1.4\,S$,

two samples — $S_{anal} = 1.2\,S$,

three and more samples $S_{anal} = S$ where S_{anal} is the scatter value considered; S is the scatter value from the statistical processing.

The performed check of this assumption for several materials showed rather satisfactory results for using such approach.

Basic and design characteristics of crack growth rate are assumed to be parameter values of approximating expressions, establishing the functional relationship $dl/dN = f(K)$, where K stress intensity factor scatter in the loading cycle. Three most common expressions are selected.

Erdogan equation. At $A_p K_{th} < K_{eff} < K_c/2$, where K_{th} — threshold stress intensity factor

$K_{eff} = K/(1-R)^p$,

where: R — asymmetry;

p — parameter.

The first approximation of A can give $A = 2.5$.

$$v = C_1 K_{eff}^{m_1}$$

Forman equation. At $K_{eff} > A_p K_{th}$

$$v = C_2(K^{m_2}/((1\text{-}R)K_{Cy}\text{-}K)$$

Collypriest equation. At any K_{eff}

$$v = C_3\left(\lg\left(K_{eff}/K_{th}\right)/\lg\left(K_{cy}/K_{eff}\right)\right)^{m_3}$$

When producing crack growth rates the following three asymmetry values are advisable: R=0; R=0.5; R=–1.

There are two procedures for crack growth rate analysis using characteristics received. Either the analysis is carried out using basic (average) characteristics and then the resultant life is divided by the factors given below, or at first the values C_1, C_2 or C_3 are multiply by the factors given below and then crack growth rate is calculated. The first procedure is preferable as any error is excluded in this situation dealing with nonlinear accumulation when the models are used taking into account crack growth retardation after long tension cycles.

Safety factors being the multiplys either of C_1,C_2,C_3, or of crack growth rates are calculated as follows:

$K = k_1 k_2 k_3 k_4$,

where k_1 — safety factor for material property scatter;

k_2 — safety factor for test data representativeness;

k_3 — safety factor for environmental effect (k_3>1.5, when environmental effect must be considered);

k_4 — safety factor for structural element type (k_4=1 for structural elements having multi-path load transfer and k_4=1.5 for single path load transfer).

The value of k_1 is assumed due to design scatter S anal. according to the following table:

S_{anal}	0.1	0.2	0.25	0.3
k_1	2.0	3.0	3.5	4.3

Linear interpolation is used for intermediate values. The above table is formed in the following way. It is assumed that for the standard scatter typical for principal structural materials (S_{anal} = 0.1) safety factor value is k_1=2. This includes both material scatter safety factor and safety factor (1.37) for structural features of the element (joint) as compared to the specimen being the basis for evaluation of material characteristics. If the probability level to define the lower scatter boundary is assumed to be 95%, then the following relation is used to form k_1 value:

$$k_1 = 1.37 \cdot 10^{1.645 \cdot S_{anal}}$$

Safety factor k_2 is equal to 1, if all the requirements to the volume and the representativeness of test data are satisfied. Otherwise k_2 is defined as

$k_2 = k_{21} k_{22}$, where

k_{21} — safety factor for the approximate determination of the average. This factor is calculated by the 90% Student quantile; k_{22} — safety factor for insufficiently representative range K of test points in the diagram $v - f(K)$. The range is considered representative, if test points are situated from $K = 2.5 \cdot K_t h$ to $K_c/2$. Otherwise

$k_{22} = 0.95 + (lgK_{cy} - lgK_{th} - 0.7) / [20(lgK_b - lgK_n)]$,

where K_b and K_n are, respectively, maximum and minimum values of K in the whole totality of test data.

Basic characteristics of static crack resistance K_{1C} and K_{Cy} are determined as average values in the whole totality of test data. In case when to define K_{Cy} the specimens of different width are used, then due to K_{Cy} relationship with the specimen width it is necessary to recalculate K_{Cy} values to one specific width. For example, for skin materials the standard specimen width W is assumed to be 750 mm or 1200 mm. The following formula can be used for the recalculation:

$K_{Cy}(W_1) = K_{Cy}W_2)(W_1 / W_2)^B$, where

$K_{Cy}(W)$ — K_{Cy} value at the specimen width W; W_1 and W_2 are respectively, the 1st and the 2nd specimen widths. B is a test parameter for each material (the first approximation of this value can give $B = 0.5$).

The determination of statistical characteristics follows the recalculation by the width. Herein it is convenient to present the scatter in terms of variation coefficient $K_{war} = S_{anal} / K_{Cy}$ (or $S_{anal.} / K_{1C}$). Design characteristics $(K_{1C})_{des}$ and $(K_{Cy})_{des}$ are defined from the basic ones by means of the latter division by the following safety factors:

for K_{1C} $(K_{1C})_{des} = K_{1C} / (k_s k_p)$;

for K_{Cy} $(K_{Cy})_{des} = K_{Cy}/(k_s k_p k_w)$, where

k_s — safety factor for material property scatter;

k_p — safety factor for representativeness of test data;

k_w — safety factor for recalculation by the width for K_{Cy} values.

Safety factor ks is assumed depending on variation coefficient in accordance with the table.

K_{war}	0.05	0.08	0.12	0.20
k_S	1.1	1.2	1.3	1.6

The table is formed on the basis of assumption that a standard variation coefficient is equal to 0.05 and safety value corresponding to the probability 95%. Therefore safety factor is calculated by the formula

$$k_s = 1/(1-1.645\, K_{war})$$

Safety factor kp is introduced when test dada are insufficient in volume and it is connected with the inaccuracy of average determination. It is defined by 90% quantile for the average using the known formulas and tables.

Safety factor k_w is introduced when K_{Cy} is recalculated for different widths. It is defined as follows. For the structural elements whose width does not exceed the width range of the specimens used for K_{Cy} calculation, safety factor $k_w = 1$. Otherwise safety factor value is defined depending on the fact how much the considered structural element width exceed the tested specimen width range. After the analysis of possible error conducted using some most common materials the following table of safety factors was formed versus dW value, where

$$dW = W_{max(min)} - W_e ,$$

($W_{max(min)}$ — maximum or minimum test specimen width, W_e — width of the calculated structural element).

dW, mm	< 50	< 200	< 500	> 500
k_w	1.00	1.03	1.06	1.10

If the calculated element width exceeds the specimen width, then the safety factor k_w=1.

The approaches submitted above were used to form the design characteristics of principal structural materials utilized in Structurally Significant Items of IL-96 and TU-204 aircraft. These results and the accumulated statistics enable to apply the suggested approach for other aircraft structures.

ALTAI SYSTEM FOR STRUCTURAL DAMAGE TOLERANCE ANALYSIS

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Description is provided of the ALTAI system that analytically predicts residual strength and crack growth rates for complex structural components.

The experience in working with various structures convincingly shows that it is impossible to avoid occurrence of defects (both due to manufacturing errors and occasional) and cracks (due to fatigue and/or corrosion). Development of damage tolerant structure is a reliable method for ensuring safe operations. The damage tolerance analyses can be unified (which is extremely desirable at the stage of airframe certification under NLGS-3, AP-25, FAR-25, etc.), and the theoretical study labour requirements reduced, by using the ALTAI damage tolerance analysis system. It is composed out of four subsystems:

— stress intensity factor (SIF) calculation;
— system adjustment modules;
— residual strength calculation;
— crack growth prediction.

The system has a graphic menu interface. The code length is more than 12000 C++ operators; the object oriented programming technology is used.

SIF calculation subsystem

It relies upon a SIF database that makes it possible to form a solution for a SIF either in a numerical form or as an approximating function. Currently, the database includes about 70 solutions (for through, surface and corner crack in plates, pipes, bars, shells around stress concentrations and in stiffened panels). The subsystem menu may be seen in fig. 1. The user is allowed to build up complex SIF solutions from those included in the system (by compounding or superposing the previous solutions); this stage is performed by ALTAI automatically.

System adjustment modules

They enable the user:
— to obtain a copy of the data entered;

— to define the system of units for work, including the Russian technical, SI, and customary;
— to convert the calculation results to alternative units systems.

The subsystem menu may be seen in fig. 2.

Residual strength calculation subsystem

It computes the residual strength by using:
— the linear elastic fracture mechanics;
— the E.M.Morozov's approach [1];
— the G.I.Nesterenko's approach for stiffened panels [2].

The subsystem menu may be seen in fig. 3.

Crack growth calculation subsystem

The subsystem calculates the fatigue crack growth duration, taking into account interaction of loads. It can operate in the cycle-by-cycle and quick calculation working modes; the latter reduces the calculation time by a factor of 3 to 100. The user is offered the crack rate equations by:
— Forman;
— Walker or;
— Walker-Chang.

Load interaction can be estimated by the following models:
— no load interaction;
— Wheeler's;
— generalized Willenborg model;
— model by Willenborg and Chang.

Fatigue loading is assumed to consist of a sequence of various-type flights (subblocks); one can specify periodic flights (subblocks) in the sequence, as well as periodic load steps in any flights (subblocks).

The subsystem menu may be seen in fig. 4.

Conclusion

"ALTAI" damage tolerance analysis system is proposed. It computes stress intensity factors for sophisticated structural areas and then evaluates residual strength and crack growth duration.

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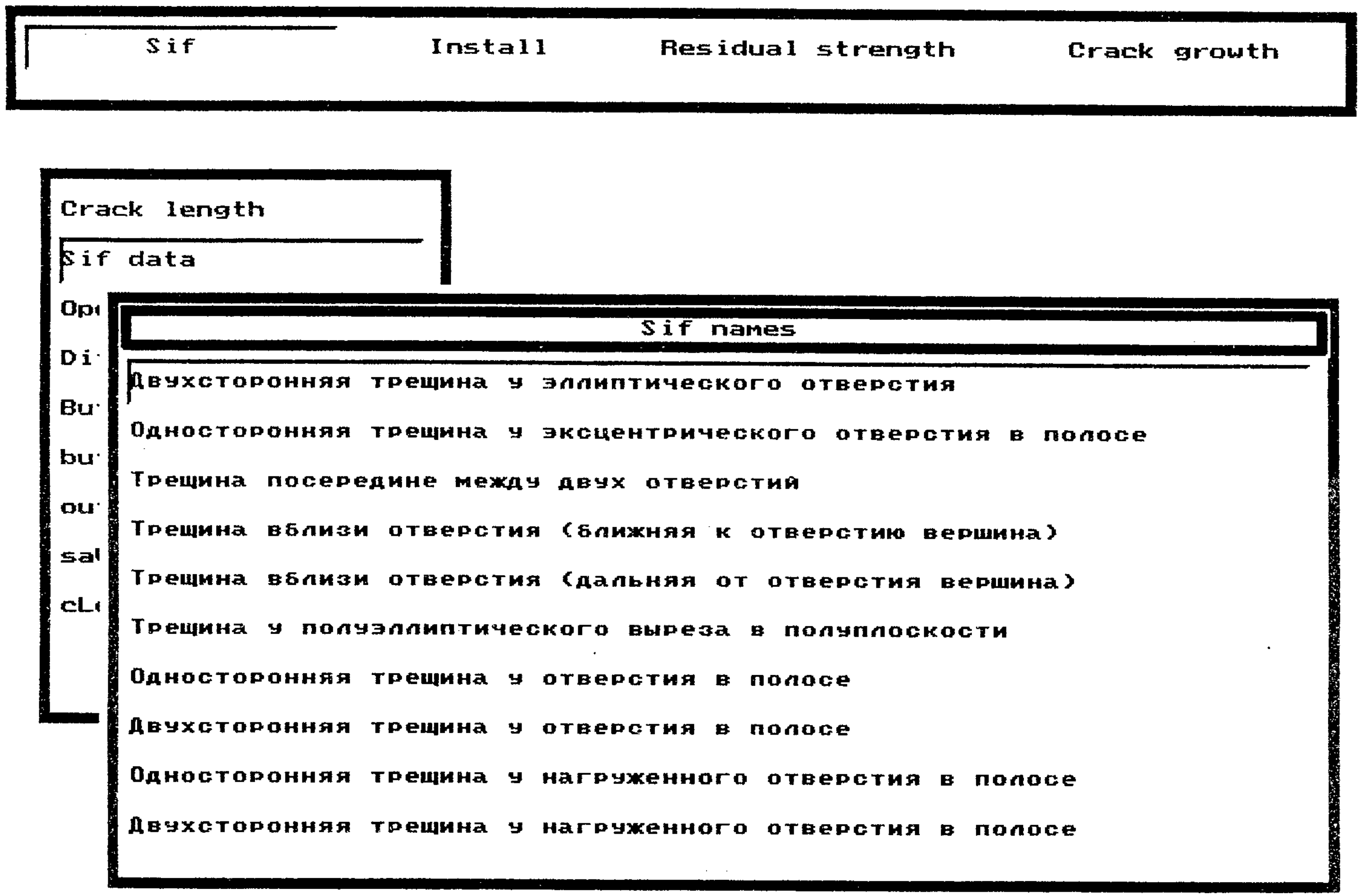
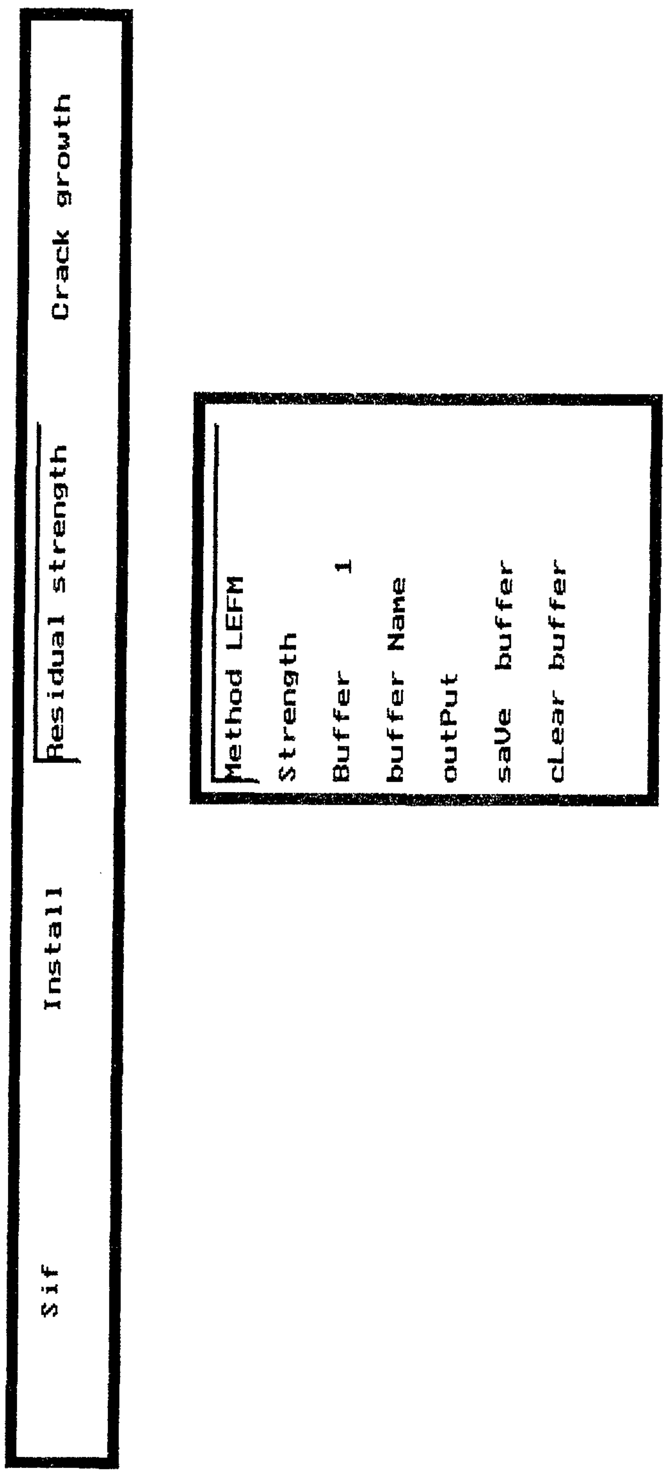


Перечень данных по КИН

```
 Sif        Install   Residual strength     Crack growth

           Protocol  ON
           Metric    RUS
           Sif DATA/FUN
           Copy      OFF
           saVe
           cHeck   ERROR
```

Установка протокола



Метод расчета остаточной прочности

```
Sif          Install      Residual strength      Crack growth

                                   loaD BLOCK
                                   load File
                                   load Scale              1
                                   Equation  FORMAN
                                   Model   LINEAR
                                   Regime CYCLE_BY_CYCLE
                                   Threshold               0
                                   Crack  life
                                   Buffer      1
                                   buffer Name
                                   outPut
                                   saVe   buffer
                                   cLear  buffer

Тип нагружения
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THE FEATURES OF AIRCRAFT STRUCTURE FATIGUE RESISTANCE CERTIFICATION AND AIRWORTHINESS MAINTENANCE IN CONTEMPORARY CONDITIONS

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INTRODUCTION

Well-grounded and practically proved normative state documents dealing with structure strength is one of the most important conditions to ensure aircraft safe operation. Standards in the countries with the developed aviation turned out to start the complete systems of requirements in all principal strength aspects such as static strength, aeroelasticity and since 1950th fatigue strength and damage tolerance. Normative strength requirements in our country were developed since 1920th [1] and present the complete system of normative documents [2,3,4].

Due to traditions and evolution features of every country national regulations had definite differences in "the philosophy", approaches and specific regulations. These differences were as a rule very small in such so to say "classical" field as static strength, but they increased when some new problems arose (e.g. aeroelasticity or so called "temperature" strength problems) and turned out to be rather substantial in such "youngest" direction as providing fatigue strength and damage tolerance.

Long-term and relatively independent development resulted in the definitely marked approaches: American one under "ideological" leadership of the USA, Western European one and Eastern European one having the USSR as a leader. The difference among these "technical regions" appeared most noticeable when normative approaches to provide fatigue strength and damage tolerance of the structure were developed. The present report is devoted to this most important and promoting most hot discussions problem.

BACKGROUND IN THE STANDARDS OF FATIGUE STRENGTH AND DAMAGE TOLERANCE

Western European approach was strongly affected by two accidents of English passenger airplane Comet I in the mid 50th caused by fatigue damage of the pressurized body. By this reason great attention was paid to fatigue strength problem for a well-grounded forecast of fatigue failure moment. Long-term and expensive full-scale fatigue strength tests were conducted during these years for many times of service life that compensate a huge life scatter typical for fatigue. It further turned out to form mainly by European efforts the concepts and approaches to generate unified programs for full-scale tests which then transformed into well-known nowadays programs — simulators of actual loading of TWIST type. Wide investigations of aircraft structure loading in actual operation were started; fundamental and well-grounded generalizations of these data were for instance included in the data bank ESDU well-known and widely used all over the world.

All these approaches affected significantly the "ideology" of standards. In particular, one of the principal roles in English normative requirements BCAR of that time [5] was played by the regulations in fatigue strength containing rather detailed and specific notes dealing with the reliability factor values for fatigue life relative to the test results. Some importance was also given to ensuring safety thanks to in-time detection of the initiated fatigue crack, but this "normative direction" was not developed so carefully as the first one called safe life.

Home approach in its early years (in the fifties) was greatly affected by the European "ideological" influence up to the fact that in the first versions of Civil Aircraft Airworthiness Regulations (NLGS-2) [2] the only concept for ensuring safety during long-term operation was safe life. There were however just doubts in the reliability of safe life forecast, in the maintaining during decades of those operational conditions that were assumed at such forecast and, finally, in the fatigue phenomenon as the only safety challenge. These doubts were answered by the system of reliability factors; by the increase in the number of the tested full-scale structures including those having operational life. For example some 4 – 6 copies of full-size airframes of such airplanes as Tu-104, Il-18, An-24 were tested. The collection and processing of load factor occurrence rate from the permanent board emergency recorders were extended for the principal types of passenger airplanes; cumulative operation time being the base for such statistics reaches almost 1.5 million flight hours [6].

Very effective in the situation of separation between design bureaus and manufacturers appeared the assumed in our country original monitoring procedure based on the idea of continuous step-by-step extension of operation time limitations, carried out in the form of establishing current (prolonged) values of the Specified Time Limits in the range of the claimed Design Goal Life and further. While the next (prolonged) Specified Time Limit is established "validity reassessment" (life and loading forecasts for the next stage) is performed. On the basis of actual technical state analysis for the aircraft fleet and its leading (candidate) fleet both more accurate correction of specific conditions for the current Specified Time Limit and the control of the adequacy and the reliability of the general approaches are conducted. Such approach possibilities were provided first by the features of our state evolution based on the rigid administrative command principles and second by the unique home civil aviation that was gathered together in the only airline "Aeroflot".

Due to "the world environment", particularly, American influence and own experience the home system of ensuring safety was systematically and significantly transformed. The most serious effect had the accident with the An-10A aircraft near Kharkov in 1972, caused by the multiple fatigue cracks in the wing lower surface skin and stringers in the joint along the aircraft centerline. Some of the most important stages in development of the home fatigue strength and damage tolerance standards were the Chapter 2 to Chapter 4 NLGS-2 dated Dec. 25, 1976 [3] as well as the third edition of Civil Aircraft Airworthiness Regulations (NLGS-3) [4], where Damage Tolerance concept was introduced as a normative approach together with (that is having equal rights) Safe Life concept. There was much freedom in the selection of any of these two concepts to ensure safety.

Damage Tolerance concept was extensively used in practice on the basis of home "Operational Survivability" concept including both Damage Tolerance and Fail Safe concepts. Fail safe properties are proved by the required residual strength control of the structure with standardized damages like two-bay cracks in the skins and panels or fully-broken spar cap with the adjacent skin and web details etc. "Operational Survivability" concept is used as Fail Safe concept for a great number of structural elements, where multi-path load transfer and redundancy of connections (fittings for high-lift wing structure and control surfaces, fittings for engines, empennage, double lugs, multi-strut landing gear etc.) are assumed. Our aircraft operation experience showed that the application of "comprehensive" concept of "Operational Survivability" in some cases helped to prevent catastrophic damages of aircraft in the situations when rather high requirements to damage detection services were not satisfied by some aircraft companies.

American approach to standards [7] absorbed the experience of very intensive and long-term operation of great aircraft fleets including very many different types operated all over the world by a large number of companies differing greatly in their level. The reaction to this experience was very active and gave the belief that "classic" fatigue is not the only problem connected with structure safety ensuring at long term operation. Two more factors are not less important, i.e. corrosion of various types (including its combination with fatigue) and accidental damages in operation (and in production) that are active initiators of fatigue crack growth.

The desire to cover all the three danger sources first gave rise to Fail Safe concept containing the provision of such reserve of load paths in the structure, the certainly detectable in operation full failure of one load transfer paths would not decrease strength below the tolerant level (as a rule, being 67% of ultimate load). This concept has long ago (since the late fifties) became dominant in the American approach that unfortunately immediately affected the practice of activities, particularly, experimental, held to ensure safety. It may be remembered, for example, the fatigue strength tests of the airframe (wing, body and empennage as a whole) of Boeing 747 were the shortest in the history of modern civil aviation and were completed after laboratory tests for only one Design Goal Life instead of common 3 to 5 lives. Full scale tests of Boeing 737 whole structure were not conducted at all, just the body having operational life was tested.

However, the assumption of the comprehensiveness of Fail Safe concept (like Safe Life concept was considered a panacea in the home requirements at the same time) again contradicted the service experience. In the seventies "ideological victory" was won by the abovementioned Damage Tolerance concept assumed in the home [3] and then in the American [7] regulations. As already stated it was used in our requirements on an equal footing with Safe Life concept, while the American approach (rather in the technical policy of state certification authorities than in the formal way) gives the Damage Tolerance concept practically absolute priority presenting as an example the opportunity to use Safe Life concept for landing gear structure.

CONTEMPORARY FATIGUE STRENGTH AND DAMAGE TOLERANCE STANDARDS

Now the essence of all the three above concepts of ensuring safety, their features and interaction is rather clear to all developed aircraft states of the world (including, of course, our country). Objective prerequisites for harmonization of national normative requirements dealing with safe operation of aircraft structures became urgent. International Civil Aviation Organization (ICAO) created in the seventies and its Technical Committee publishing the generalized normative documents [10], as well as Joint Aviation Administration (JAA) created in the eighties contributed this.

Harmonization of European and American normative requirements in the problem considered was based on the practically full adoption in the Joint Aviation Requirements (JAR) [8] of the general requirement text of 25.571 from Federal Aviation Regulations (FAR) of the USA [7]. This seemed to be the just recognition of the great and really "pioneer" contribution of the USA into the development and practical approbation of the methods for solving this serious problem. True, some practical important moments were added to JAR 25.571 that enable the decrease of the required residual strength in the case when the initiated damage is easily detected during short time of operation. Besides as it could have been expected all the previous experience of European countries was maintained in the form of specific (and often quantitatively different) "standards" of various aircraft manufacturers, in particular, reliability factors and other technical aspects.

NEW HOME AVIATION REGULATIONS

General

In the nineties the necessity arose to harmonize home standards with those that are today almost world-wide normative requirements to ensure aircraft structure strength and particularly the problem discussed herein. The activities in general were leaded by Aviation Register of Interstate Aviation Committee while specific developments dealing with strength standards were traditionally carried out by TsAGI experts in cooperation with experts from the leading aircraft design bureaus. Similar to the harmonization principles adopted earlier by the western countries for the requirements of FAR 25.571 and JAR 25.571 devoted to ensuring safety during long-term operation, it was decided to base home Aviation Regulations (AP 25.571)[9] on the American standards as the corresponding home requirements (section 4.9 of Civil Aircraft Airworthiness Regulations edition three [4]) were "ideologically" close to the requirements of FAR 25.571. Some significant changes were meanwhile introduced into the general "philosophy" of safety ensuring. First it was assumed necessary to consider possible corrosion and accidental damages besides the fatigue ones. Second Safe Life concept is only used when the Applicant proves that damages tolerance concept does not work in this particular structure.

It was simultaneously assumed valid that some principal moments approved by the years of home aviation history should be reflected rather in the basic general requirements than in the additional normative documents.

The following three moments are included into such most principal problems:

- home monitoring "technology" based on the idea of step-by-step establishing (prolonging) of Specified Time Limits alongside with the appropriate changes (if necessary) in the service documents considered;

— priority value of full-scale structure test results as compared with analytical methods;
— similarity of the inspection threshold (when Damage Tolerance concept is used) and safe life terms as far as the order and the methods of their definition and establishing are considered.

The rest more specific standards contained in Civil Aircraft Airworthiness Regulations and having up to now the position of obligatory requirements are, in general, maintained, however, noticeably changed to be included into the newly made normative document of "the second level" — Methods of Compliance Definition (MOS 25.571). This document has the position of an advisable one like the US Advisory Circular (AC 25.571) or European Advisory Circular (ACJ 25.571). The detailed consideration of this rather large document cannot be performed in the boundaries of this report. But some its most important things may be discussed in short in the last part of the report.

Monitoring procedure

The assumed home practice of step-by-step prolongation of the Specified Time Limits seems clear without any comments. However, is should be noted that it only differs in form from the similar foreign procedures meant to provide the opportunity for the required changes in the conditions of Design Goal Life (and longer) work out. The efficiency of obligatory and centralized actions during the step-by-step life prolongation must have been more significant in contemporary conditions if there were very many small airlines having no adequate experience and skills. But instead of its adequacy in home conditions approved by years some difficulties can arise in the mechanism of providing this procedure in the specific conditions when certification is performed for a foreign "user". That is why principal (obligatory) requirements of AP 25.571 do not insist on such procedure referring to MOS 25.571 for details, but regulate just the necessity of some monitoring procedure, providing equivalent level of safety. Let's quote the addition (a)(4) to the original (a) of FAR 25.571, included into AP 25.571:

— (4) The documentation developed in accordance with (a)(3) shall be periodically reviewed on the base of taking into account and analysis of investigations and test results and also the considered aircraft type service history being accumulated. The procedure shall be determined to provide the reliability and appropriate timing of this analysis. The recommendations contained in MOS 25.571, part 1 can be used to determine this procedure or the any other procedure can be used to provide the equivalent safety.

Full-scale structure tests

Home Airworthiness Regulations paid much attention to fatigue strength full scale laboratory test result themselves (including both fatigue crack initiation moment and their growth way and time). It can be said that there are no types of home aircraft whose full scale structure was not fatigue strength tested having large reliability factors in life relative to the Design Goal Life. As mentioned above several aircraft copies were tested in some cases including the structures having operational life. The experience showed that test costs (certainly, at the reasonable test volumes) are surely compensated by the significant increase in the structure quality and reliability assurance. First, more complete detection of its critical sites is provided and particularly of probable multiple site damages. Second, the reliability of actual life property definition increases greatly including crack growth stage considering structure features and its production technology. Analytical methods of life determination both at the stage before detectable crack initiation and at their growth stage always left much place for doubts especially in the complicated structural zones of primary joints under multi-component loading and multi-axial stress-strain state. All these considerations resulted in the most close maintenance in the obligatory requirements of AP 25.571 (and even more in recommendations of MOS 25.571) the very "experimental emphasis", typical for Civil Aircraft Airworthiness Regulations. The phrase "by the analysis supported by tests" was changed for another phrase "by tests and/or analysis" in all positions of AP 25.571 as compared to FAR 25.571. The dominations in MOS 25.571 are much better specified: the first significance is given to the direct full-scale test results (or structural members equal to them), the second one is given to the indirect full-scale test results recalculation for other critical zones of the same structure, and finally, analysis results.

Safe life

Home certification practice gained large experience of Safe Life concept utilization. This experience witnesses, for example, that Safe Life concept should be used for critical zones where multiple-site fatigue cracks can initiate having very small sizes and hardly detectable while their junction cam result in drastic failure. Such critical zones are longitudinal splices and joints of the pressurized body frame, transverse wing splices and other members. Safe Life concept usage is also reasonable for such structural zones whose damage and complicated repair in the range of the design operational life leads to the extreme increase of costs for the aircraft itself and those connected with the forced lost of time.

The concern dealing with drawbacks of this concept is quite natural. Home experience has a great number of examples where the "failures" took place during forecasting operational life. However this experience can be in general qualified as positive, as Safe Life concept usage correlated reasonably both with Damage Tolerance concept and Fail Safe concept. Really, the most important technical document being the basis of step-by-step monitoring methods is the Regulation for Specified Time Limits (initial or next prolonged) establishment. It contains three obligatory Appendixes, being the integral part of any Regulation.

The first Appendix is a list of members to be replaced during the prescribed life, the second one is a list of structural parts (zones) and members to be adjusted (repaired) during the prescribed life. The items of these two lists and schedules are defined by safe life analysis results. The third Appendix is a list of structural zones to be inspected during the established prescribed life. The items of this list, control methods and tools as well as schedules, i.e. the inspection threshold and inspection interval are defined by damage tolerance analysis results.

It should be noted that contradiction of three currently formulated safety ensuring concepts is not considered constructive even by the leading American certification experts [11]. Any of these concepts alone is unable to solve as a whole such most difficult problem as safety ensuring from the point of view of structural strength.

At the same time reasonable correlation of various aspects of one or other concept appears quite useful from the point of view both of safety and economics. For example, natural caution during the establishment of the inspection threshold for damage tolerant structure seems absolutely valid due to possible initial flaws (manufacturing or operational). Similar caution should also cover the procedure of safe life determination in case the structure allows these flaws. Strict recommendations about the necessity to consider the initial flaw during safe life determination are given in MOS 25.571. On the other hand the structure by the inspection threshold must maintain the residual strength enough for recognition of damage tolerance presence. Taking into consideration the above mentioned similarity the same structure state may be considered the end of safe life.

All these considerations are paid attention to during formulating home requirements of (c) in AP 25.571 in the following form:

(c) Fatigue (safe-life) evaluation.
Compliance with the damage-tolerance requirements of paragraph (b) of this section concerning inspection interval determination in accordance with (a)(3) is not required if the applicant establishes that their application for particular structure is impractical. This structure must be shown by analysis, supported by test evidence and analysis, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks de-

creasing residual strength and residual rigidity more than 25.571(b) indicates.

Unlike the same paragraph text in FAR 25.571:

(c) Fatigue (safe-life) evaluation.
Compliance with the damage-tolerance requirements of paragraph (b) of this section is not required if the applicant establishes that their application for particular structure is impractical. This structure must be shown by analysis, supported by test evidence to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks.

MOS 25.571 RECOMMENDATIONS

As it was already mentioned special attention at the harmonization of home and foreign normative requirements was paid to providing those principal methods and safety ensuring system having official reflection in Airworthiness regulations [3,4] and having proved its high efficiency during long-term operation of our aircraft fleets. Principal items of this system, maintained in MOS 25.571 . recommendations, were considered in a quite detailed way in Ref.[12,13]. Some important aspects of the home system are to be noted that are though similar to the foreign approaches however have some home special features.

Much attention was paid to the development of lists with Superduty Elements and Critical Structural Items. These terms are similar to foreign terms Principal Structural Elements (PSE) and Structurally Significant Items (SSI). Superduty Elements are those whose single failure (for example, fracture) leads to accidental or catastrophic situation. Critical Structural Item is a structural zone where dangerous damage can initiate during operation, therefore such zone requires special attention. The a priori version of this list used at design and generation of programs for full scale fatigue strength tests must be corrected by these test results and actual operation experience. The aircraft structure in operation is adjusted and repaired, so the list is changed and some critical zones appear that differ from those of the initial structure. These zones are to be analyzed like the initial ones.

The loading of critical structural elements is the principal information required for evaluation of any life limits including the first inspection moment and inspection interval. It is the loading that must be subjected to systematic monitoring for increasing forecast reliability during the procedure of step-by-step establishment (prolongation) of the Specified Time Limits, MOS 25.571 recommendations utilize the specific "tools" in the form of reliability factors to stimulate actual loading evaluation. In particular reliability factor 1.5 (or 2 in the case of insufficient forecast at the design stage) can be decreased down to 1, if the reliable system of actual loading evaluation is upgraded, for example, using the information that can be received from on-board emergency recorders [14] or specific means of "loading monitoring" [15].

It should be noted that home standards have a developed system of normative reliability factors. The formal "centralized" standards in this field of safety ensuring (contrary to the foreign practice) without large deviations by the design bureaus is the effective feature of normative practice in our country.. This feature is maintained in MOS 25.571 including the approved system of reliability factors [4], where the accumulated factor is the product of separate factors. Each of these factors is responsible for separate important aspect of the problem: possible errors in analytical methods, loading scatter, fatigue durability scatter etc.

MOS 25.571 has some corrections in the system of reliability factors as compared to the Civil Aircraft Airworthiness Regulations. For example reliability factors for life scatter during fatigue crack growth is decreased; but an additional factor 1.5 is introduced at the same stage to consider the unfavorable environmental effect.

During evaluation of life characteristics of Superduty Elements under one-path loading a factor 1.5 is introduced to consider the risk degree and damage controllability. So when Safe Life concept is used the established operational life due to this factor actually corresponds to the structure state close to the undamaged one or the one having quite small damages, when the residual strength corresponds to the ultimate load.

The text of MOS 25.571 as compared to the Civil Aircraft Airworthiness Regulations is significantly enlarged in other aspects. In particular, the section "Additional notations for the composite structures" is enlarged and corrected as the recommendations in the reliability factor values taking into account the significantly increased fatigue life scatter of such structures are of great importance.

Absolutely new section "Tolerant operational life of the critical structural element under corrosion strength" is introduced containing recommendations in evaluation methods and the order of establishing the schedules of the required in-service activities (replacements, repairs and inspections in the range of the Specified Time Limits). Safety ensuring connected with a possible corrosion damage initiation is based on the "Operational Survivability" concept. Average time values before the damage initiation and growth rate in the critical structural elements prone to corrosion are determined on the basis of data analysis and summation for service experience of similar structures under the conditions close to the expected. The analysis takes into account the environment aggressiveness degree, the expected efficiency of the specific protection and covers, the material sensitivity to all kinds of corrosion and in particular to stress corrosion etc. When the tolerant operational life before the inspection threshold and inspection interval are established the required reliability factor is assumed (relative to the time average values). Some requirements to production technological processes whose parameters affect the structural fatigue strength characteristics are introduced into MOS 25.571 text due to the great importance of this safety ensuring aspect.

Finally, the section "Damage tolerance evaluation (discrete source)" from AC 25.571-1B draft is transferred into MOS 25.571 text as a whole because the American experience of these requirement application is wider than the home one.

CONCLUSIONS

1. The harmonization of home and foreign requirements to structural safety ensuring by strength conditions during long-term operation appeared easily carried out, because the basic principal approaches to the problem including the most modern ones are very close. In particular an "Operational Survivability" concept is widely used for safety ensuring including the aircraft designed earlier officially (while much earlier in practice) since December 1976, i.e. since the introduction of Change 2 of chapter 4 of Civil Aircraft Airworthiness Regulations, edition 2.
2. The harmonization process is by no means the refusal from the accumulated large home experience in this field having demonstrated its high efficiency in our country. Therefore the adoption of the current essentially common world requirements as national ones is accompanied by some definite differences, taking this experience into account. Very small number of most important differences of methodological kind is formulated in the AP 25.571 principal requirements themselves, while technical and procedure features using the approved standards from the home Airworthiness Regulations are contained in MOS 25.571 after some modifications. Such approach to harmonization of home and foreign requirements seems most valid.
3. The performed harmonization of home and foreign requirements dealing with structural safety ensuring by the strength conditions during long-term operation is an important stage for introducing our aviation into the world economics and the necessary step on the way to mutual certificate recognition.

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FULL-SCALE AIRFRAME FATIGUE CERTIFICATION TESTS

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Currently, one can see the ever-growing significance of airframe testing for fatigue in order to improve the airframes, in both Russian and international aviation. The amount of structural analyses and component testing at the design stage is growing, but the full-size airframe tests are the only measure to improve elements whose service life is governed by — the structure manufacture/assembly process and/or — redistribution of forces in neighboring elements under the cyclic load.

The primary objective of the full-size airframe fatigue tests is to ensure safe operation; testing is the major requirement in aviation regulations in both Russia and abroad.

The other important objective of the full-size tests is to prepare the inspection schedule and the in-service maintenance procedures.

In the airframe fatigue resistance improvement program the major point is to test the airframe as a unit. Such tests make it possible, first, to better reproduce the typical in-service loads and, second, to avoid development of special adapters that would represent the stiffness of the removed portion of the structure.

The development of the airframe loading program for lab tests relies on two postulates:

(1) the external loads corresponding to the actual operations should be reproduced to a maximum possible accuracy, and
(2) the abilities of the test facilities/equipment must not be exceeded.

These requirements are completely met by programs such as "flight-by-flight" concept with:

— random sequence of flight with various load levels, and
— random alteration of in-flight variable loads.

The development of such program proceeds from the spectrum of variable loads expected in flight and ground conditions (fig. 1). To derive a lab test spectrum, the whole of the load factor range significant in terms of fatigue accumulation is subdivided into 5 levels. These 5 levels are included into 5 flight types: **A**, **B**, **C**, **D**, and **E**. In so doing, the "least severe flight", **E**, includes only the minimum loads, whereas the "most severe flight", **A**, includes all he five levels; these are changed in a random fashion. These five flight types are the basis for a block of flights; the flights in a block are mixed at random. The total number of flights in a block is 0.1 of the airframe service life. Further, the loading program derivation procedure (fig. 2) includes

- the development of the program layout,
- the development of a logic of the multichannel load-generation/application system, and
- computation of concentrated loads to be generated by the channels.

As a rule, the program layout reflects all phases of a typical flight and incorporates all loads to occur in ground-handling and flight conditions.

When developing the logic of the multichannel load-generation/application system, the task is to find out a compromise between the desire to minimize the total number of channels and the necessity to reproduce

- aerodynamic and inertia loads acting on the wing, fuselage, tail, wing high-lift devices, and
- loads applied to such units as the landing gear and engines.

For example, testing the Il-96 airplane required 120 channels.

The concentrated loads for each channel during each segment redefined by minimizing the root-mean-square deviation of a load parameter, subject to equilibrium conditions and load limitations. The forces are determined from the condition of minimum functional:

$$F = \left(K_{test} - K_{oper}\right)^2,$$

where K is the load parameter which can be assumed

a) concentrated forces P,

b) shear force Q, bending moment M_{bend}, torsion moment M_{tors} or combinations thereof,

c) normal stresses σ, or tangential stresses τ,

d) equivalent stresses ,being a combination of normal and tangential stresses.

The minimum functional is evaluated under the condition of equilibrium of the airframe as a whole:

$$\sum X = 0, \quad \sum M_X = 0,$$

$$\sum Y = 0, \quad \sum M_Y = 0,$$

$$\sum Z = 0, \quad \sum M_Z = 0,$$

and under the restriction of the concentrated loads:

$$P_i < [P]_i, \; 1 < i < n,$$

where n is the total number of the channels.

The sequence of concentrated loads derived in this manner is generated cyclically in lab conditions by using the automated complex (fig. 3) that provides:

— programmed cyclic loading through 200 channels,
— vibration loading at frequencies up to 50 Hz,
— real-time monitoring of loads at 1000 channels,
— permanent inspection of test structure integrity, and
— 16000 channel strain-gauging.

Monitoring of loads and airframe displacements relies on:
— force monitoring for each channel,
— measurement of extreme displacements of the airframe as a whole, and
— permanent determination of equivalence of the actual loading to the reference loading (fig. 4).

The force monitoring is conducted by the multichannel system. It ensures reproduction of the prescribed force-time function (a sine represented by 25 points per a half-period) for each of the channels. If extreme forces turn out to exceed limits then the system unloads the structure by using a special subroutine. In addition, the system envisages emergency unloading if some parameters become unacceptable. Between the extremes of each segment the overshoot (or undershoot) of forces causes the system to either retard or accelerate loading. Also, emergency unloading is made if an airframe violates limits with respect to angles of pitch, roll and/or yaw.

There is another supervision option, the real-time stress/strain monitoring system. It measures forces at extreme points and the displacements (by means of calibrated strain-gauge bridges mounted at characteristic points of the structure). These data are the basis for computing integral (resulting) factors and a fatigue amount accumulated during a complete flight. Thereafter, an equivalent is determined the ratio of fatigue for an actual complete flight to the fatigue amount for a reference flight. Deviation of the equivalent from the unity allows to estimate an error in load mode reproduction.

The experience in full-size airframe fatigue testing shows that over 75% of calendar time is spent for structure repair after fatigue damages. To reduce the idle time for repairs, use is made of "damaged structure zone release" method (fig. 5). With a fatigue damage revealed, the system forms a new vector of concentrated loads which causes a lower stress around the crack (so as not to develop the crack), whereas the rest of the structure is loaded with an acceptable deviation. Test are continued, and simultaneously the causes of the damage are investigated, documentation is prepared, and relevant structural parts are manufactured. The tests are only ceased when the structure itself is being recovered. After the repair the tests are continued obeying the original program.

In the case of detecting a fatigue crack the effective method for arresting the cracks is employed, as shown in fig. 6. The method is based on the drilling of a stop hole at the crack tip in addition to which a special device that keeps the crack opened during the cyclic loading is mounted on the crack. This option makes it possible to stop the 20—180 mm long cracks in various wing components for periods of 600 up to 4000 lab flights.

Timely detection of fatigue cracks is supported by the flaw detection system; it is based on the development of inspection schedule that stipulates for individual zones the minimum detectable crack length, inspection intervals, method and labour amount. The schedule is developed on the basis of

— structural features,
— results of testing and using similar existing airframes, and
— results of fatigue and damage tolerance evaluation.

The reliably detectable crack lengths and the inspection methods and labour requirements are established proceeding from the previous experience in inspecting the airframes during fatigue tests. Detectable crack lengths for various inspection methods are represented in fig. 7.

After the structure is tested for as long as 2 or 3 design service life, the residual strength is evaluated. The major objective of the tests is to show compliance with requirements to residual strength of the structure with natural or artificial cracks. The challenge is that a single structure should be utilized to evaluate the structure damaged at a number of points. There exists danger of a complete failure before the limit load is attained for all zones. The task is solved managed by employing the method protecting the airframe against complete failure (fig. 8). The essence is to load the airframe stage-by-stage and monitor the crack in the stable growth area. This method allows to cease the loading before the crack becomes critical.

As for pressurized bodies, there exists a possibility to arrest a crack at unstable growth stage (fig. 9): a structural area near the crack tip should be heated so as to generate compressive thermal stresses; the unstable crack stops propagating if its tip approaches this area.

With the test work package fulfilled, the permanent joints are disassembled to detect damage in the zones difficult to access. Fig. 10 demonstrates the procedure of such studies for an Il-86 wing. These studies revealed 650 cracks in longitudinal joints; the crack growth data were summarized to adjust the in-service inspection schedule.

Conclusion

The present technique for fatigue tests makes it possible to use a single full-size airframe to fulfill the entire set of for fatigue resistance and damage tolerance testing and to adjust the maintenance and flaw detection schedules.

STRUCYURE LOADING SPECTRUM FOR IL-96 FATIQUE TEST

- "FLIGHT-BY-FLYGHT" TEST CONCEPT
- 5 TYPES OF FLIGHTS FOR DIFFERENT SERVICE CONDITIONS;

  FLIGHTS ARE MIXED AT RANDOM IN A BLOCK OF 1200 FLIGHTS

Flight type	Number of flights in block
A (severest loads)	1
B	9
C	65
D	435
E (lowest loads)	690

- 5 LEVERS OF VARIABLE LOADS MIXED AT RANDOM IN EACH FLIGHT

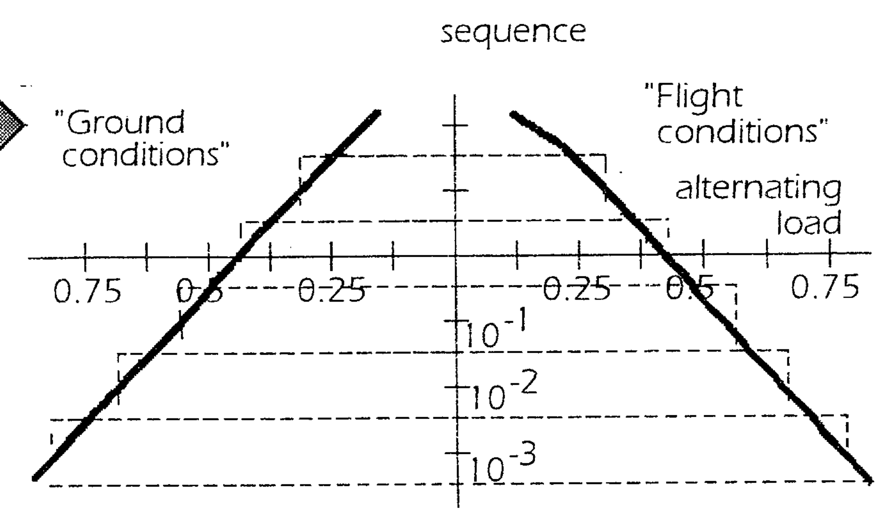
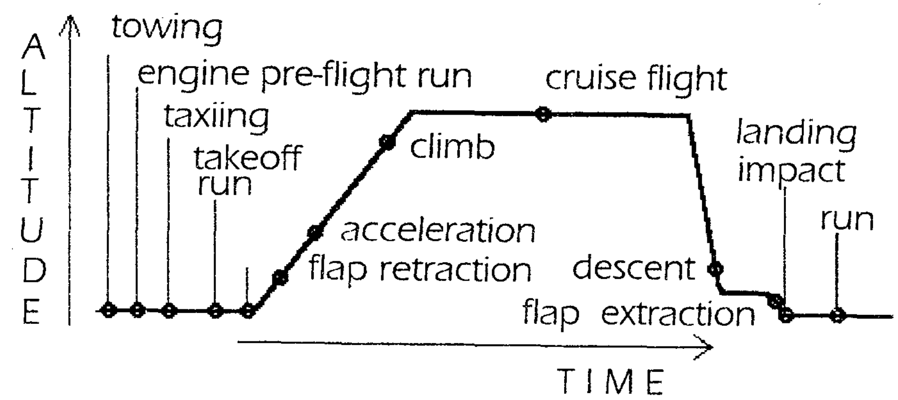


Figure 1.

COMPOSING A LOADING PROGRAM

- PROGRAM COMPOSITION REFLECTS TYPICAL FLIGHT STAGES, INCLUDING:
  - GROUND LOADS;
  - FLIGHT LOADS.



- THE LOADING SYSTEM APPLIES CONCENTRATED LOADS TO ALL AIRFRAME UNITS IN THREE DIRECTIONS

- CONSENTRATED LOADS TO EACH FLIGHT SEGMENT ARE DETERMINED TO MINIMIZE ROOT-MEAN-SQUARE DEVIATION OF LOAD GENERATION LIMITATIONS

$$F = (K_{test} - K_{oper})^2 \Rightarrow \min$$

$$K = \left| \begin{array}{l} S^* \\ Q, M_{bend.}, M_{tortion} \\ \textit{NORMAL AND SHEAR STRESSNES} \end{array} \right.$$

$$\Sigma X = 0 \qquad \Sigma mom_x = 0$$
$$\Sigma Y = 0 \qquad \Sigma mom_y = 0$$
$$\Sigma Z = 0 \qquad \Sigma mom_z = 0$$
$$P_i \langle [P]_i$$

Figure 2.

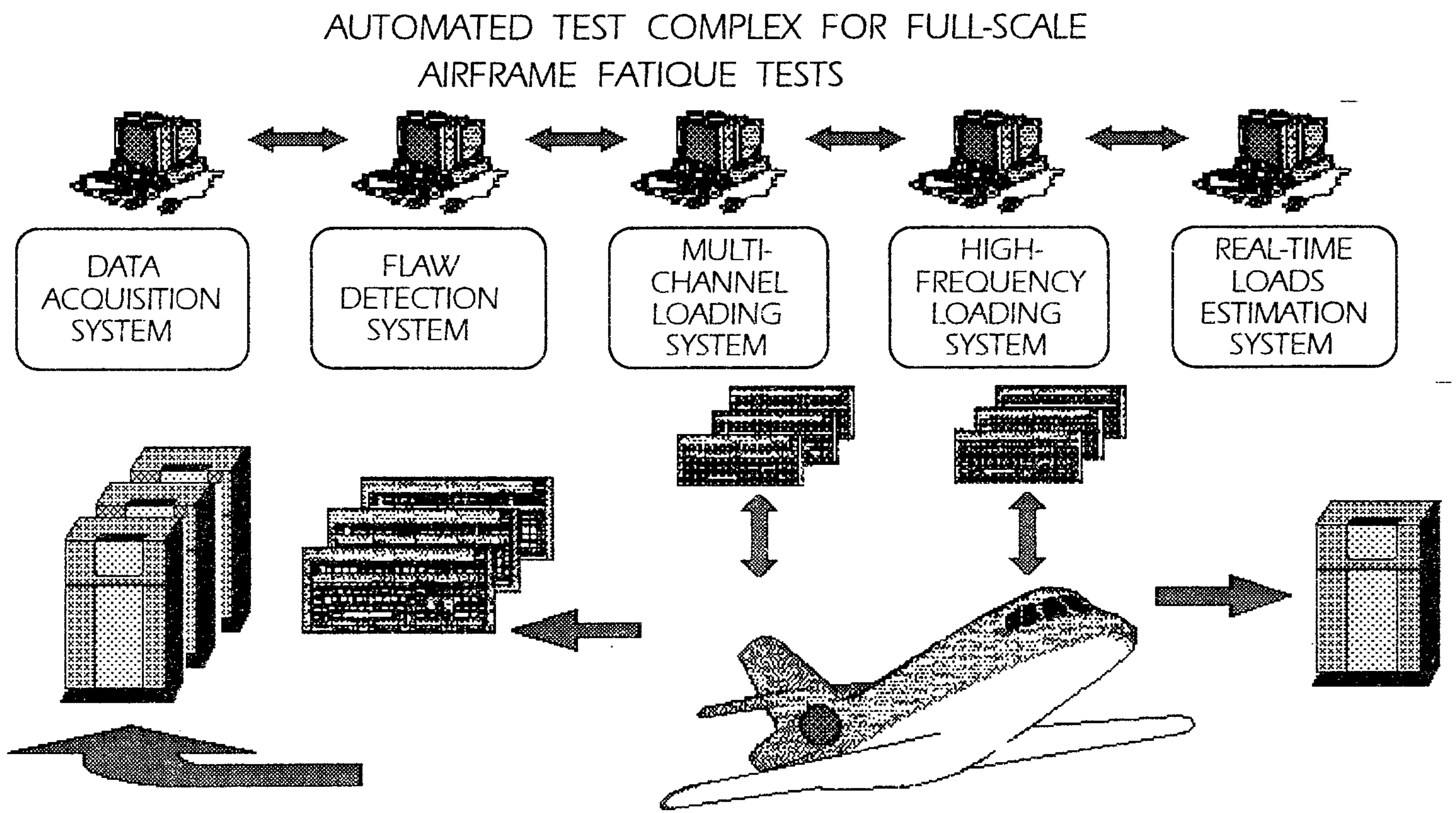
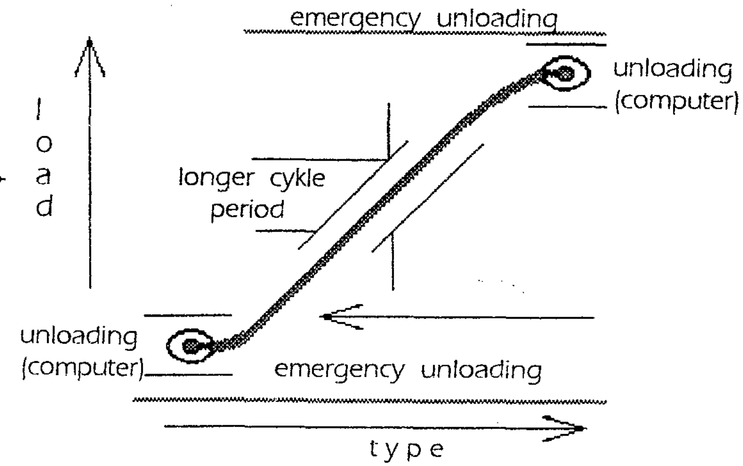


Figure 3. THE COMPLEX PROVIDES:

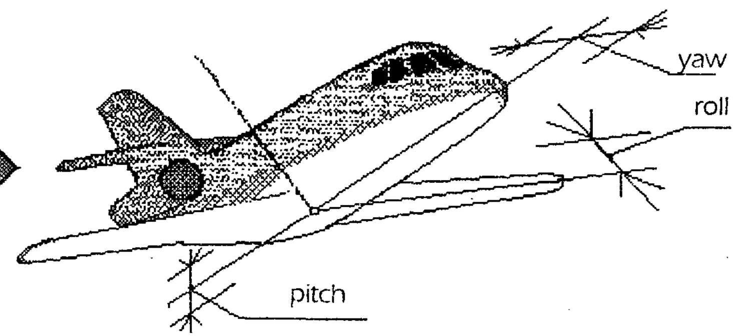
-programmed cyclic loading through 200 channels
-vibtation loading at frequencies up to 50 Hz
-real-time monitoring of loads at 1000 channel
-16 000 channels strain gauging
-permanent inspection of test structure integrity

AIRFRAME LOAD/DISPLASEMENT MONITORING

- FORCE MONITORING FOR EACH CHANNEL OF AUTOMATIC LOADING CONTROL SYSTEM



- MEASUREMENT OF EXTREME AIRFRAME DISPLACEMENT IN ROLL, YAW AND PITCH



- PERMANENT DETERMINATION OF EQUIVALENCE OF THE ACTUAL LOADING TO THE REFERENCE LOADING

$$E = \frac{N(K_{test})}{N(K_{oper})}, \quad K = \left\| \begin{matrix} P \\ Q, M_b, M_t \\ STRESSES \\ S^* \end{matrix} \right.$$

Figure 4.

UNLOADING OF DAMAGED AIRFRAME STRUCTURIAL ZONES

- REPAIRS TAKE UP 75% OF CALENDAR TIME INCLUDING 40% FOR PREPARATION
- MODIFIED PROGRAMME INCLUDING THE DAMAGED AREA UNLODING
- TESTS WITH PARTIAL UNLOADING AREA CONTINUED UNTIL REPLACEMENT PARTS ARE SUPPLIED

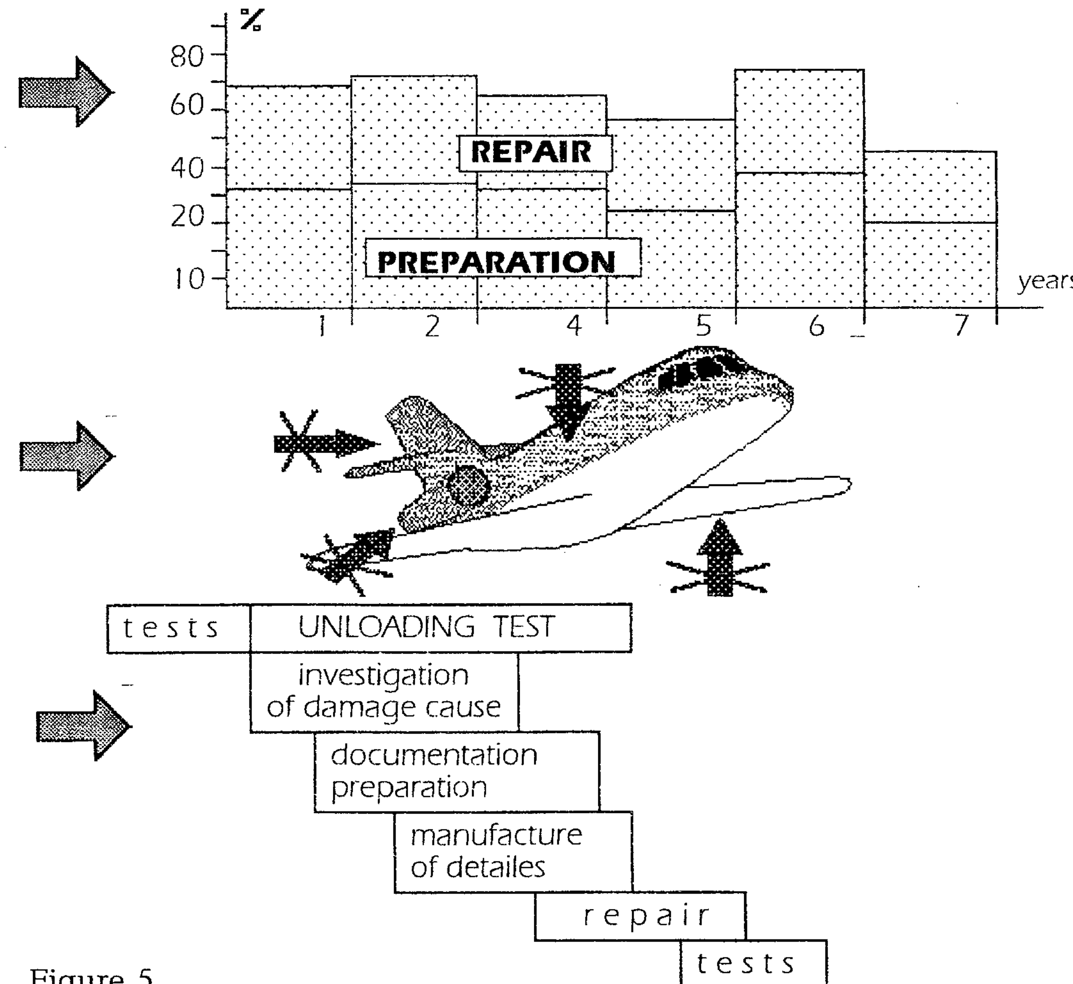
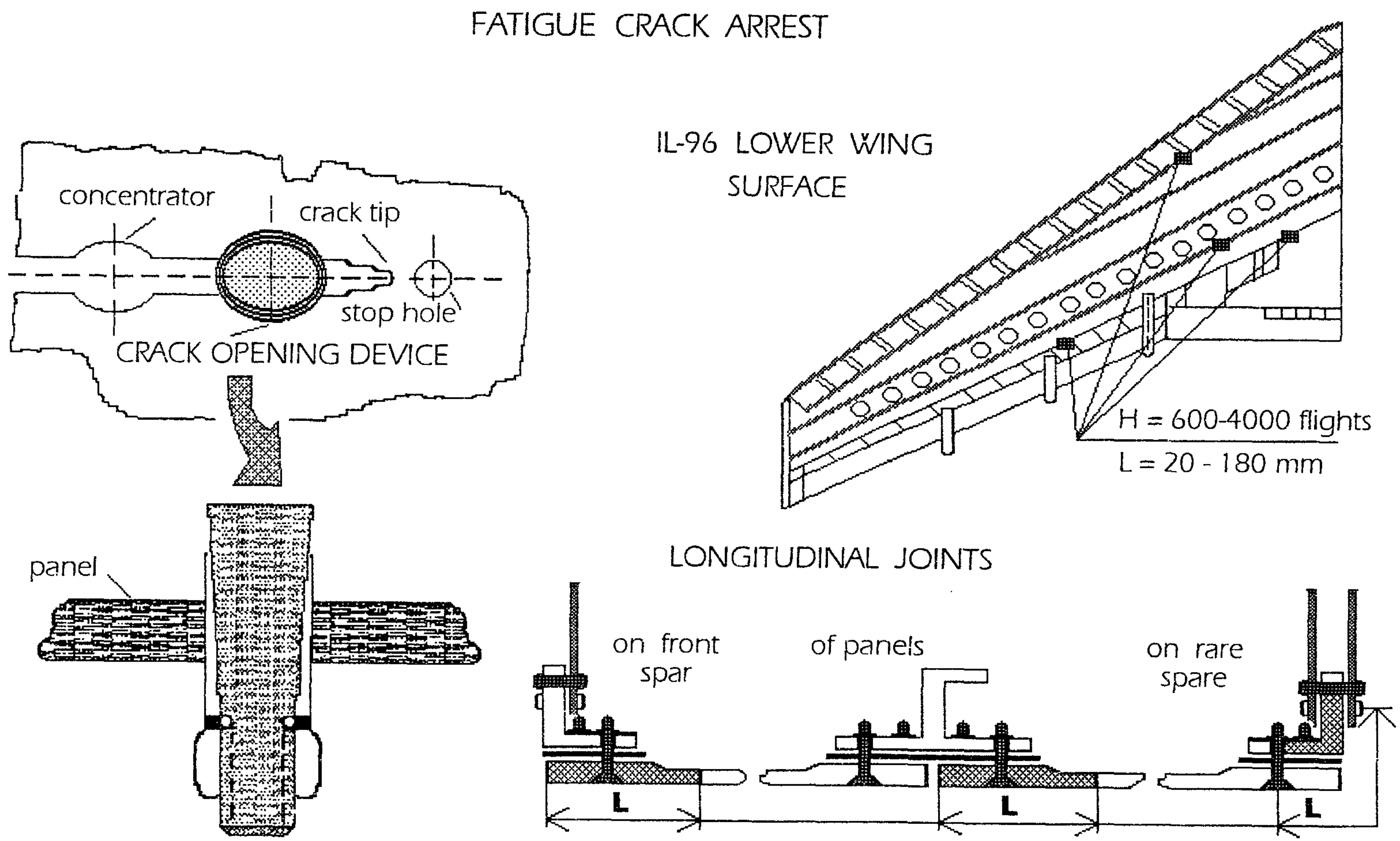


Figure 5.



FATIGUE CRACK ARREST
concentrator
crack tip
stop hole
CRACK OPENING DEVICE
panel
IL-96 LOWER WING
SURFACE
H = 600-4000 flights
L = 20 - 180 mm
LONGITUDINAL JOINTS
on front
spar
of panels
on rare
spare
L
L
L

Figure 6.

RELIEBLY DETECTABLE CRACK LENGTHS

INSPECTION METHODS	TYPICAL CRACK LENGTH MM		
	surface crack	internal crack	under doubler
VISUAL INSPECTION	25-50		
ULTRASONIC INSPECTION	5-10	5-10	10-15
EDDY-CURRENT INSPECTION	5-7		15-20
X-RAY INSPECTION		10-15	15-20
MAGNETIC POWDER INSPECTION	3-5		
LIQUID PENETRAITE INSPECTION	1		

Figure 7.

RESIDUAL STRENGTH EVALUATION WITH STRUCTURE FAILURE PREVENTION

- RESIDUAL STRENGTH OF A WING WIDTH STANDARDIZED DAMAGE IN 8 ZONES

- STEP—BY—STEP LOADING IN SRABLE CRACK GROWTH AREA

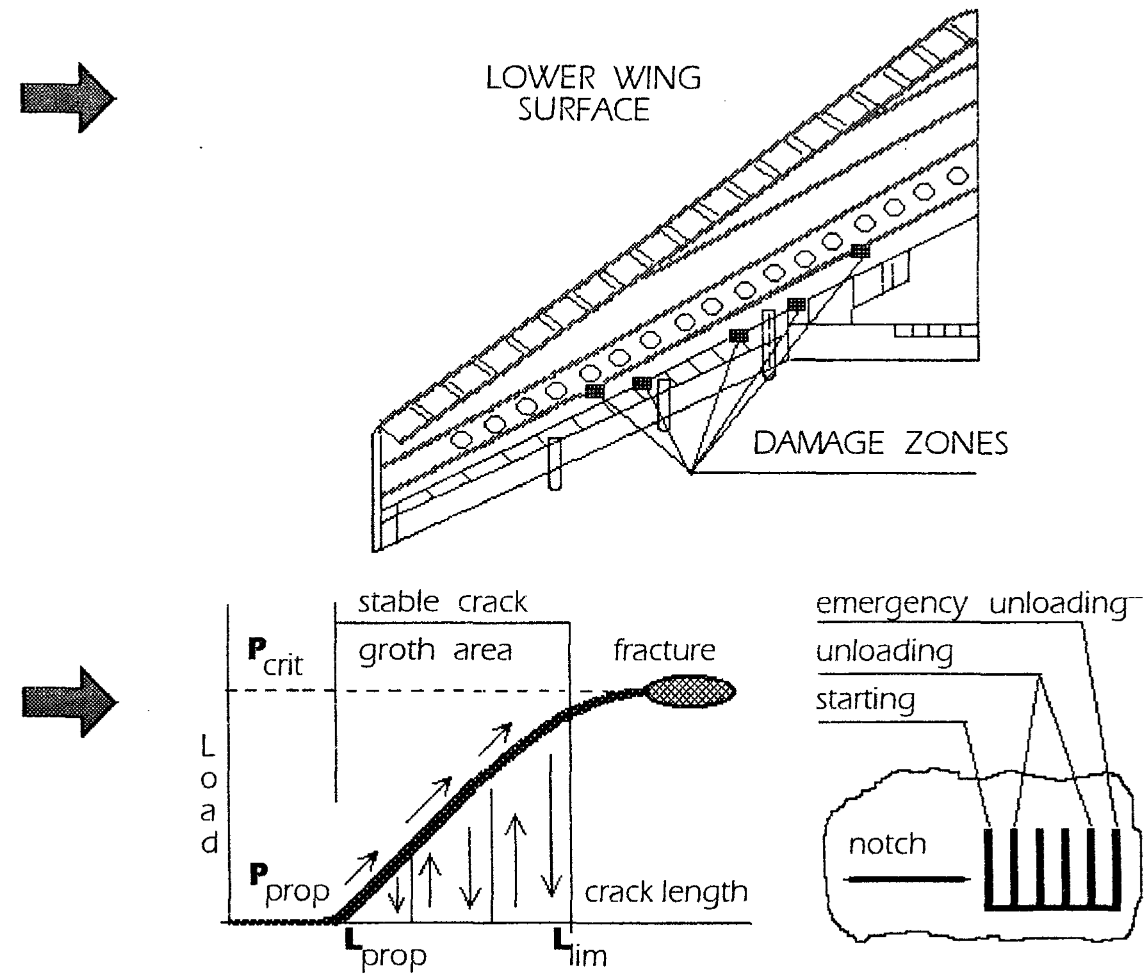
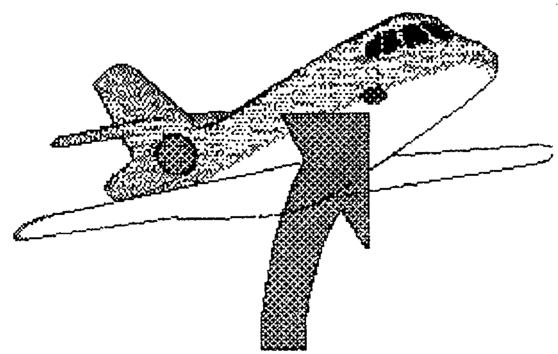
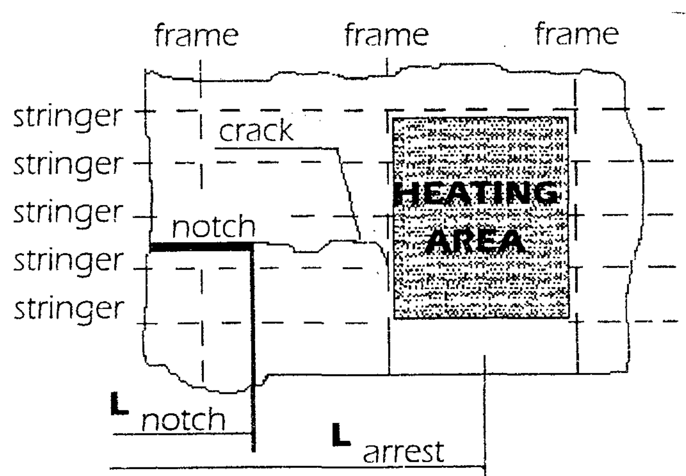
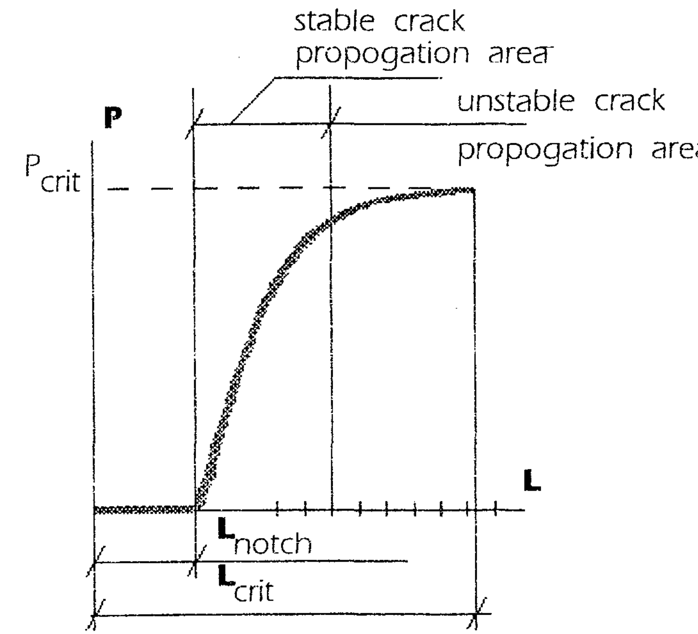


Figure 8.

UNSTABLE CRACK ARREST







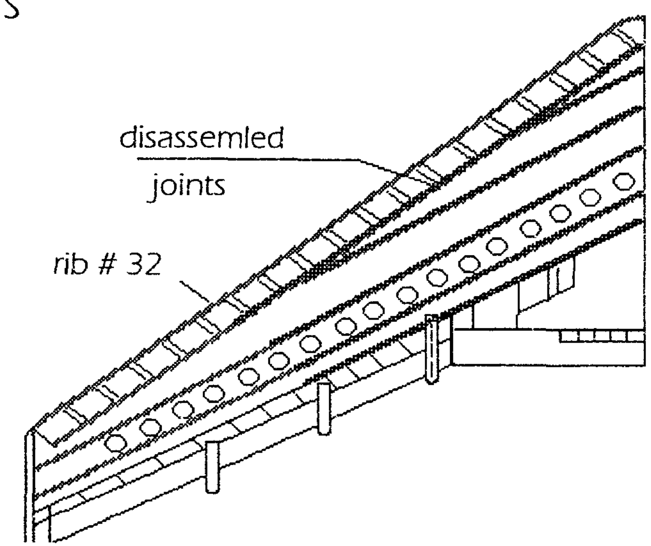
METHOD ENABLE

- Precluding explosive decompression of fuselage under damage tolerance tests
- establishing limit load
- detarmining critical crack length for several zones of one structure

Figure 9.

DISASSEMBLING THE IL-96 WING AND INSPECTING THE PERMANENT JOINTS

- DISASSEMBLING THE LONGITUDINAL JOINTS OF LOWER AND UPPER PANELS FROM THE WING ROOT TO RIB # 32



- EDDY-CURRENT INSPECTION OF 101 000 HOLES; 650 CRACKS DETECTED

- FRACTOPHRAFY OF 27 CRACK FROM LOWER SURFACE AND 54 CRACKS FROM UPPER SURFACE

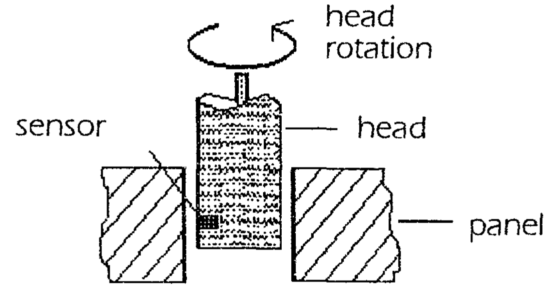


Figure 10.

LANDING GEAR STRESS ANALYSIS

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ABSTRACT

Stressing of Landing gear is done for the loadings arising during various landing and take off cases set by the Government or Civil Airworthiness approval organisations. There are wide detail differences in the requirements in the military and civil spheres in various countries. Hence it becomes difficult to compare the different specifications. Also due to phenomenal geometric differences, there is no unique stress analysis for landing gears. Engineering judgement have to be applied for analysing the different element. This paper present an analysis for MiG-21 landing gear for different landing configuration. Tyre and shock absorber efficiencies were estimated by drop test and overload factor was calculated based on design sinking velocity defined in MIL 8862 (ASG). Different landing loads were calculated for the load factor and stresses for landing gear elements have been estimated. The study has been restricted mainly to main landing gears.

SYMBOLS and NOTATIONS :

W_L	= *Aircraft Landing Wt Kg*
G	= *Static load per wheel in Kg*
V	= *Vertical velocity during landing m/sec*
N	= *Ratio of dynamic load to static load per wheel (overload factor)*
P_x, P_y, P_z	= *Load arising during landing at wheel axle (Kg)*
F_x	= *Load arising at tyre end (Kg)*
η_t, η_{sa}	= *Efficiency of tyre and shock absorber*
δ_t, δ_{sa}	= *Deflection of tyre and shock absorber*
E	= *Energy to be absorbed during landing*
H	= $V^2/2g$ *vertical height of drop testing*
h	= *Height of a/c c.g. above ground*
p_o, p_s, p_n	= *Air pressure in the strut in extended, static and compressed condition.* Kg/cm^2
x, y, z	= *Co-ordinate system.*

INTRODUCTION

The energy the landing gear requires to absorbs depends on the landing weight and vertical velocity of descend of the aircraft(sinking velocity). It absorbs this energy during landing by the compression of the tyre and the shock absorber system . While the efficiency of the tyre is below 0.5, shock absorbers could be designed for efficiency of the order of 0.85 -0.9. Since the total deflection ($\delta_t + \delta_{sa}$) is limited , the force generated due to the impact is much in excess to static load. This creates an overload factor (N) of about 2 to 3.

An investigation was conducted to analytically determine N for MiG-21 landing gear system. The tyre load and deflection was estimated based on empirical formulation. Shock absorber load was calculated based on Hadkel formulation with polytropic compression in the oleo -strut . Subsequently actual drop tests were carried out on landing gear system assuming 75% lift available as per MIL -T - 60538 . η_t, η_{sa} and N were evaluated . For evaluating the strength, landing gear loads were calculated based on N for the various loading cases as per the three specifications. One to one correspondence of landing cases do not exist. Though MIL gives lower axial load, side loads are higher compared to USSR loads. This gives higher flexural and torsional stresses in the two planes. Principal stresses were therefore compared. Both MIL and USSR indicate stresses in the plastic zone. Considering Bending Modulous of Rupture the struts could take up higher landing weights.

Fatigue testing on two landing gears were conducted to evaluate life. Load spectrum was decided by statistically apportioning the different landing cases. Using a fatigue scatter factor of 12 the gears were cleared up to 3000 landings. This paper discusses the analytical formulation, experimental results and fatigue testing of MiG-21 Landing Gear System.

BRIEF DESCRIPTION OF THE MIG-21 LANDING GEAR

MiG-21 Aircraft has a nose wheel tri-cycle type landing gear. Main wheels are installed on the wing and while retracted are accommodated in the wing and fuselage. Nose wheel retracts backwards in the fuselage. The main landing gear has oleo pneumatic shock absorber of telescopic type fitted with cantilever offset wheel. Both tube less as well as tyre with tube could be fitted to the wheel. While the pressure of the tyre is 9 kg/cm^2 the pneumatic pressure in shock absorber is 30 kg/cm^2. The schematic diagram of the landing gear is shown at Fig-1.

ENERGY ABSORPTION AND OVERLOAD FACTOR

Total vertical energy that has to be absorbed by the each landing gear unit is given by

$$E = 1/2 \ G/g \ V^2 \ .... (1)$$

Static load on any wheel at any landing weight can be calculated from the location of main and nose gear with respect to aircraft center of gravity. Vertical velocity are to be taken from Airworthiness standards. While Def Stan 00-970 (UK) indicates likely percentile exceedance of various design sinking speeds, MIL-8862 (ASG) specifies following sinking speed.

Table 1 : Design Sinking Speed (Land Planes)

	At design landing weight (m/sec)	Maximum design gross wt (m/sec).
Non-carrier based		
Navy primary and basic trainer	5.18	
USAF Primary and basic trainer	3.96	2.59
All other class	3.048	1.828

From the reverse calculation of operational energy as given in the Russian Album and the aircraft landing weight, the sinking speed used by Russian works out to be 4.2 m/sec. In the present analysis V=3.048 m/sec is used.

EXPERIMENTAL DETERMINATION OF η_t, η_{sa} and N

TYRE EFFICIENCY

The load deflection curve for main under-carriage tyre against static load is shown at Fig-2. Tyre efficiency calculated from the above curves gives :

a) For tubeless tyre η_t = 46%

b) For tyre with tube -

at charging pressure 8.5 kg/cm^2, η_t=44%

9.0 kg/cm^2, η_t=48%

DROP TESTING

Drop testing of landing gear was done to determine the overload factor and shock absorber efficiency. For this impact testing was conducted for different drop weights at a constant drop height (H) of 470 mm (V2/2g for V=3.048 m/sec) on a main landing gear. The time history of loading, shock absorber travel δ_{sa} and total travel of the LG (h) were recorded in a U-V recorder. The speed for paper travel of the recorder was 50 cm/sec. The $\delta_{sa\ max}$ and h_{max} were measured using potentiometer. The overload P was measured using load cells.

However, as per MIL-T-60538 , 75% lift can be considered available during landing. Considering this and neglecting static load on nose wheel (2 point landing) the landing weight of the aircraft W_L could be calculated as :

$$2\ G\ (H+h) = W_L.H + W_L\ h - 0.75\ W_L.h$$

$$W_L = \frac{2\ G\ (\ H+h)}{(H + .25\ h)} \qquad (\ 2)$$

Then overload factor $N = \frac{2P}{W_L}$ ....(3)

The $P - \delta_{sa}$ is shown at Fig- 3 for various landing wt. The shock absorber efficiency is calculated as the ratio area under the curve and the actual area as :

$$\eta_{sa} = \frac{\Sigma \Delta P \cdot \delta_{sa}}{P \cdot \delta_{sa\ max}} \qquad (4)$$

TABLE-2 : Experimental Results (Drop Testing)

W_L (kg)	P_{max} (Kg)	$\delta_{sa\ max}$ (mm)	h_{max} (mm)	η_{sa}	N
6210	8207	189	255	.87	2.639
6710	8850	198	276	.86	2.63
7300	10345	204	292	.80	2.8
8700	12850	216	324	.79	2.954

It is seen that η_{sa} decreases and overload factor increases with increase in aircraft landing weight. As the LG is designed for a maximum W_L of 7300 kg, the shock absorber efficiency and the overload factor obtained are very close to values normally assumed in the design.

THEORETICAL ESTIMATION OF OVERLOAD FACTOR

The total energy of impact during landing is absorbed by the tyre and shock absorber by their compression. The travel of shock absorber is calculated in a iterative manner. For this purpose 'Shock Absorber Calculations' by R.Hadkel, is discussed below :

Total energy to be absorbed per wheel $= \frac{G V^2}{2g}$

Energy absorbed by tyre and Shock absorber -

$E = \eta_t \cdot \delta_t \cdot N \cdot G + N \cdot G \cdot \eta_{sa} \cdot \delta_{sa}$ ...(5)

From a study of actual load deflection curves of typical tyres, $\delta_t = \delta_{t\ max}\ (N\ G/P_t)^{0.9}$ ... (6)

Where $\delta_{t\ max}$ = Max.tyre deflection (Taken as 2/3 tyre width in the absence of any other data).

P_t = Dynamic load to squash the tyre (taken as three times design static load in the absence of exact data).

($\delta_{t\ max}$ = 117 mm and P_t = 12280 Kg given in tyre data.)

For the shock absorber N and δ_{sa} are both unknowns but they are inter-related. The relation can be worked out from the polytropic compression of the air in the strut generally taken as $pv^{1.3}$ = Const.(i.e. the compression is not fully adiabatic).

$$p_o \times l^{1.3} = p_n\ (1 - \delta_{sa}/\ l\ )^{1.3} \ldots \quad (7)$$

But the static compression of strut will be isothermal,

$p_o \times l = p_s \times b$ ....

Where , l = length of the air column in the extended condition

b = length of air column in static condition ..

$$\text{Then ,} \quad N = p_n/p_s = \frac{b/l}{(1 - \delta_{sa}/l)^{1.3}}$$

$$\text{Or,} \quad \delta_{sa} = \left[1 - \frac{(b/l)^{0.77}}{N^{0.77}} \right] . l \quad (8)$$

Usually b/l is defined in the design of the shock absorber. It is usual to provide an available stroke equal to 3/4 l (i.e. a compression ratio of 4) and to inflate the strut so that the static position corresponds to one third extension of the leg. This corresponds to b/l=.5 .In some design b/l=0.625 is chosen which will correspond static position at half extension with compression ratio of 4 .

Thus substituting for δ_{sa} from eqn 8 we can get an expression for N as given below :

$$\frac{V^2}{2g} = \eta_t N^{1.9} \delta_{t\,max} (G/P_t)^{0.9} + \eta_{sa} N\, l \left(1 - \frac{b/l^{.77}}{N^{.77}}\right). \qquad (9)$$

For the present case operational travel shock absorber is 200 mm. Considering one third travel as clearance volume and for a compression ratio of 4 and b/l = 5, N calculated from eqn (9) is 2.5. The deflection calculated from eqn (8) is 189 mm. This value of N is low;thus thus experimental value of N=2.8 has been used for stress calculation.

LANDING GEAR LOAD CALCULATION :

The various landing cases as per Russian, MIL and British Specifications are shown below :

TABLE-A : Russian LOADING CASES

1. 3 point landing - Ew
2. 2 point landing
   a) 2-point landing - Ew
   b) 2-pt landing with forward hit (high drag)-Gw
   c) 2-pt ldg with wheel not rotating (skidding)- $(Ew' + G'w)_1$
   d) 2-pt ldg with anti-drag - (E'w + G'w)2
   e) 2-pt ldg with side force R^{+P2}_{1W} and R^{-P2}_{1W}

3. Ground Loads
   a) Taxi with brakes R^{CT}
   b) Taxi without brakes R^{BT}
   c) Torque about strut when ac is rotating MW

TABLE-B : MIL 8862 ASG

1. Spin up loads
   a) 3 pt ldg
   b) 2 pt ldg
   c) 2 pt ldg with high angle
   d) Drift landing.

2. Spring Back Loads
   a) 3 pt ldg
   b) 2 pt ldg
   c) 2 pt ldg with high angle
   d) Drift ldg

3. Ground Loads :
   a) 3 pt braking
   b) 2 pt braking
   c) Turning loads
   d) Taxing loads
   e) Piloting loads.

4. Handling Loads
   a) Towing loads
   b) Jacking loads

TABLE-C : DEF STAN 00-970

1. Combined drag and side load
2. Side load in board
3. Side load outboard
4. High drag and spring back
5. One wheel landing
6. Rebound of unsprung parts

Case 2,3, & 5 do not apply nose wheel.

The loads calculated for Russian and MIL cases for the main wheel at aircraft landing wt of 7300 with design factor of safety of 1.5 is shown below :

Table 3: Loads as per MIL-A- 8862 ASG

	Loading Case	Py(kg)	Px(kg)	Pz(kg)
1.	3 pt SU	12983	9088	–
2.	3 pt SB	12983	-8114	–
3.	2 pt SU	16647	5300	–
4.	2 pt SB	14831	-6029	–
5.	Drift ldg	1483		-11489 on one wheel 8902 on other wheel

Table 4 : LOADS AS PER RUSSIAN CASE

		Py	Px	Fx	Pz
1.	Case - E	12976	–	–	–
2.	Case - E'	14823	3627	–	
3.	Case - G	5278	6290	–	
4.	Case - R1+P_z	6932	1696	–	3812 on one wheel 2425 on other
5.	Case - R_2CT	6205	–	3257	6205
6.	Case - T	7789	–	2874	

STRESS ANALYSIS

The schematic diagram for landing gear loading is shown at Fig-1 .The various stress components coming in the analysis are :

a) 2-bending stresses in xy and yz planes

b) 2-shear stresses in xy and yz plane

c) Compressive stress due to vertical load P_y

d) Shear stress due to torque

e) Hoop stress due to internal oil & nitrogen pressure

f) Hoop stress in the air bottle area of the strut.

The sectional diagram of the strut is shown at Fig-4 . Forces and reaction on the strut is shown at Fig- 5. The force 'P' shown in fig 5 is the force in the landing gear actuating jack. For the purpose of the stress analysis the strut has been divided into four sections as shown in Fig-4 . Various stress component for the loads as per MIL-8862 were calculated. For the purpose of strength analysis resultant principal stresses for all these stresses were calculated. Critical stresses are given below :

Table 5: MAX PRINCIPAL STRESS

	Loading Cases	Zone-1	Zone-2	Zone-3	Zone-4
1.	3 pt SU	115.19	122.47	179.24	181.73
2.	2 pt SU	124.5	130.93	143.90	118.07
3.	2 pt SB	70.92	17.12	-33.37	-38.85
4.	Drift ldg	172.4	184.13	194.25	196.01

It is seen that drift landing stresses are very high. This is because this case use 0.8 times the vertical loads as side loads. This high side loads give rise to high bending stresses.

STRENGTH COMPARISON

Since the design stresses with design factor of safety of 1.5 is beyond the material strength of 180^{+10} kg/mm^2, the stresses are expected to reach plastic region. The material is a ductile material and bending modulus of rupture for the section is calculated as follows :

$$\sigma_{br} = \sigma_m + k.\sigma_o$$

Where σ_{br} = Bending Modulus of Rupture

σ_o = Yield strength = 140 kg/mm^2

σ_m = Ultimate strength = 180 kg/mm^2

$$k = \text{Factor} = \frac{d \int y \, dA}{2 \int y^2 \, dA} - 1$$

(d = outer dia of strut)

For a inner dia 102 mm and min wall thickness 6 mm, k works out to be 0.28 and thus σ_{br} = 219.2 kg/mm^2.

$$\text{Margin of Safety} = \frac{219.2}{196.01} - 1 = + .118$$

FATIGUE TESTING OF THE LANDING GEAR :

Initial life prescribed for the landing gear was 1800 landings. This value was very much conservative .To ascertain actual life potential of the strut fatigue testing was conducted. In the Russian method , low amplitude loads are neglected and thus four landing cases were considered.Landing loads used for the testing was as given in the Russian Album. The loading block is shown below :

Table 6: Fatigue Load Spectrum as per Russian Method

Sequence	Loading Cases	Load Level	No of Cycle
1.	2 pt ldg with high drag $(E'+G')_1$	60 % of Design load	100
2.	2 pt ldg with tail down $(E'+G')_2$	,,	100
3.	3 pt ldg (E)	50 % Design load	400
4.	2 pt ldg in cross wind R_1	,,	50

The above table shows one block of 650 cycles. In the block 3 point landing has been assumed for 400 cycles which appears to be very severe.In actual practice 3 point landing case may have low occurrence.

Two landing gears were chosen for the fatigue testing. These gears had completed approximately 1700 landings before subjecting to testing. Testing was discontinued after 25 blocks with inspection after every block.There was no crack or nay other defect reported . A scatter factor of 12 was used for evaluating the life. This gave one block corresponding to 55 landings .Thus:

Fatigue Life = 1700 + 55 x 25 = 3075 ldg.

Based on above fatigue life of the landing gears was assigned as 3000 landings.

DISCUSSION AND CONCLUSION :

Stressing of landing gear is done for the landing cases as well as design sinking speed prescribed in the airworthiness standards. Load estimation as per Russian and MIL specification are shown. It is seen that Russian specification is more severe. Based on MIL load and stress analysis , it is seen that landing gear could take up addition landing loads. Life assigned to landing gears initially are conservative. Based on actual fatigue testing life could be increased. Experimental values obtained for the tyre and shock absorber efficiencies as well as the dynamic load factor for the oleo strut were close to the theoretical values assumed in the design.

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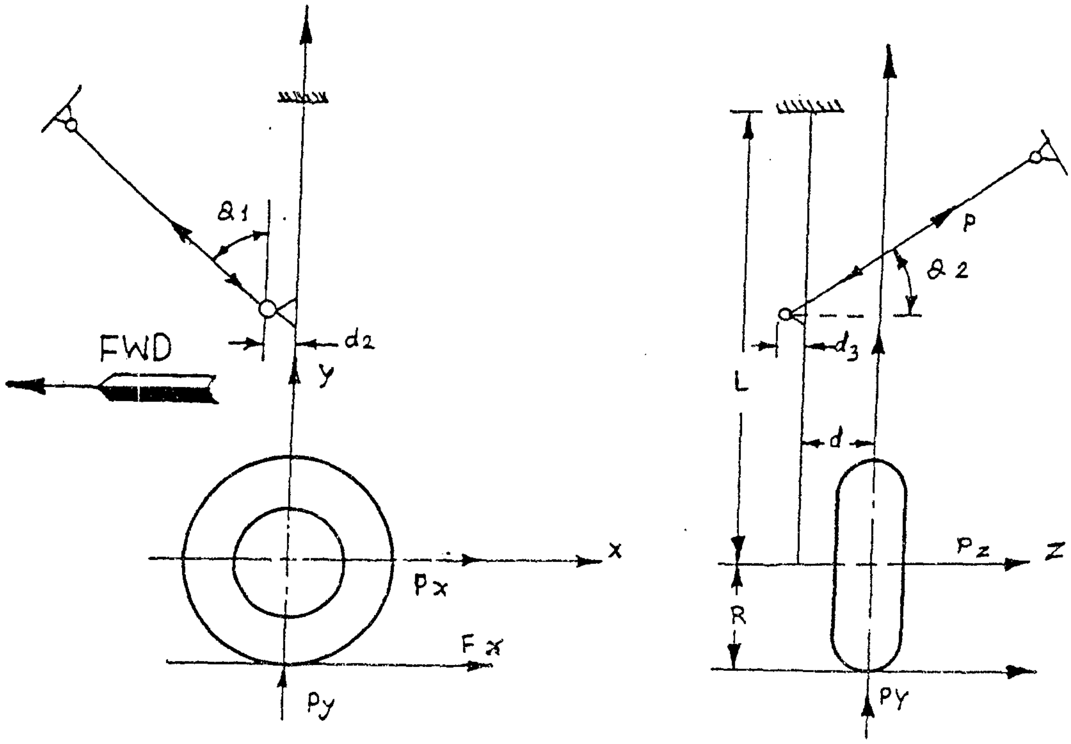


U/C X-Y PLANE

U/C Y-Z PLANE

VARIOUS PARAMETERS

L = Free length from u/c axle to wheel axis=1223mm

R = Radius of the tyre = 400 mm

θ_1= angle 30^{o} , θ_2=angle 28^{o}

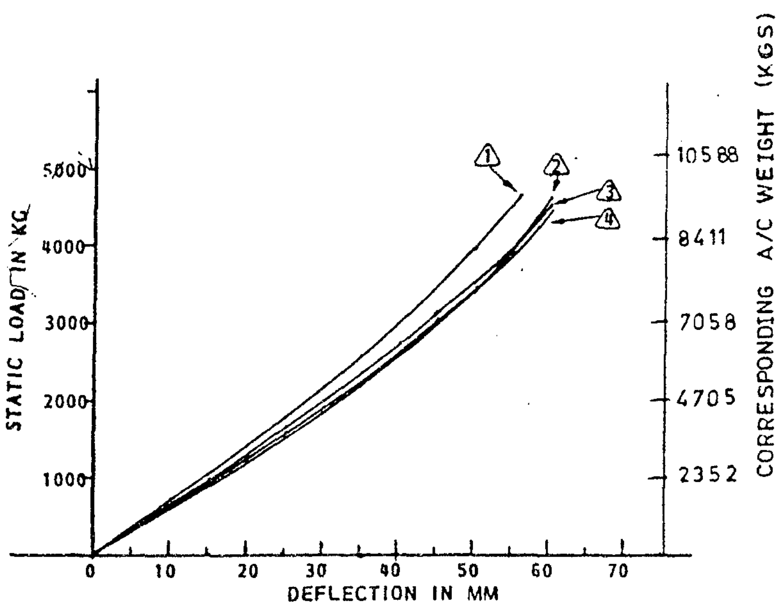
d = 180mm ; d_2 = 72mm , d_3 = 106mm

Px and Fx = Landing load in X direction

Py = Vertical load

Pz = Landing load in Z direction

Fig.1 SCHEMATIC DIAGRAM ON LANDING GEAR LOADING



1 USSR TYRE WITH TUBE CHARGING PRESSURE 9.0 Kg/CM^2
2 USSR TYRE WITH TUBE CHARGING PRESSURE 8.5 Kg/CM^2
3 DUNLOP TUBELESS TYRE CHARGING PRESSURE 9.0 Kg/CM^2
4 DUNLOP TUBELESS TYRE CHARGING PRESSURE 8.5 Kg/CM^2

Fig.2 LOAD DEFLECTION CURVE OF TYRE

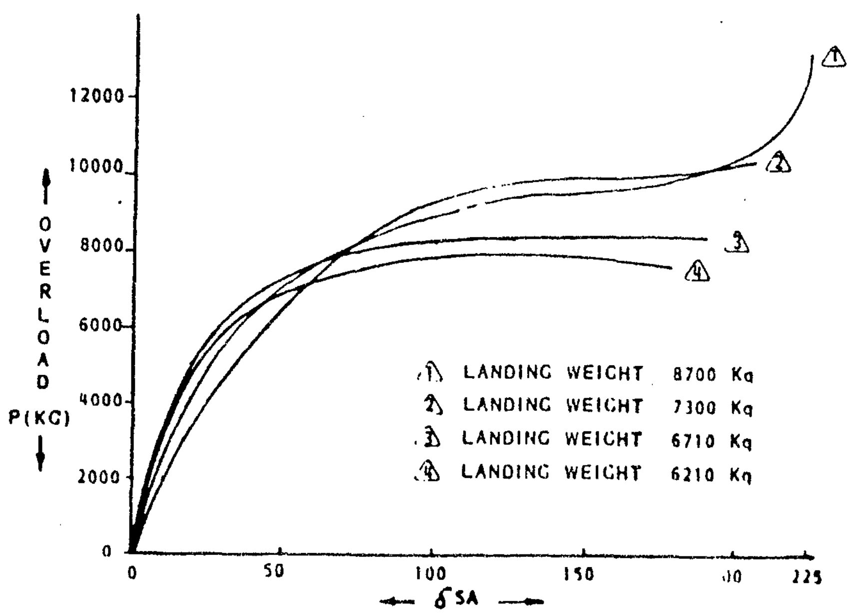


Fig.3 LOAD DEFLECTION CURVE OF SHOCK ABSORBER

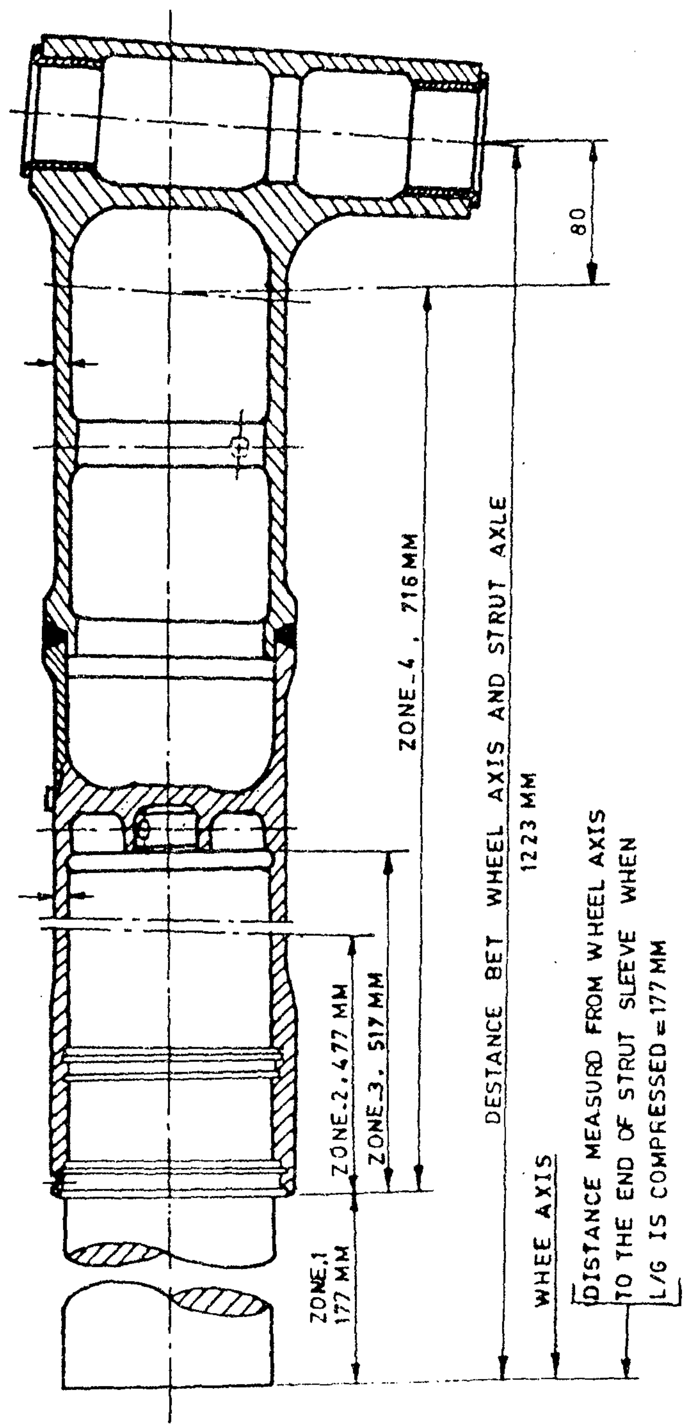


Fig.4 SECTIONAL VIEW OF THE STRUT

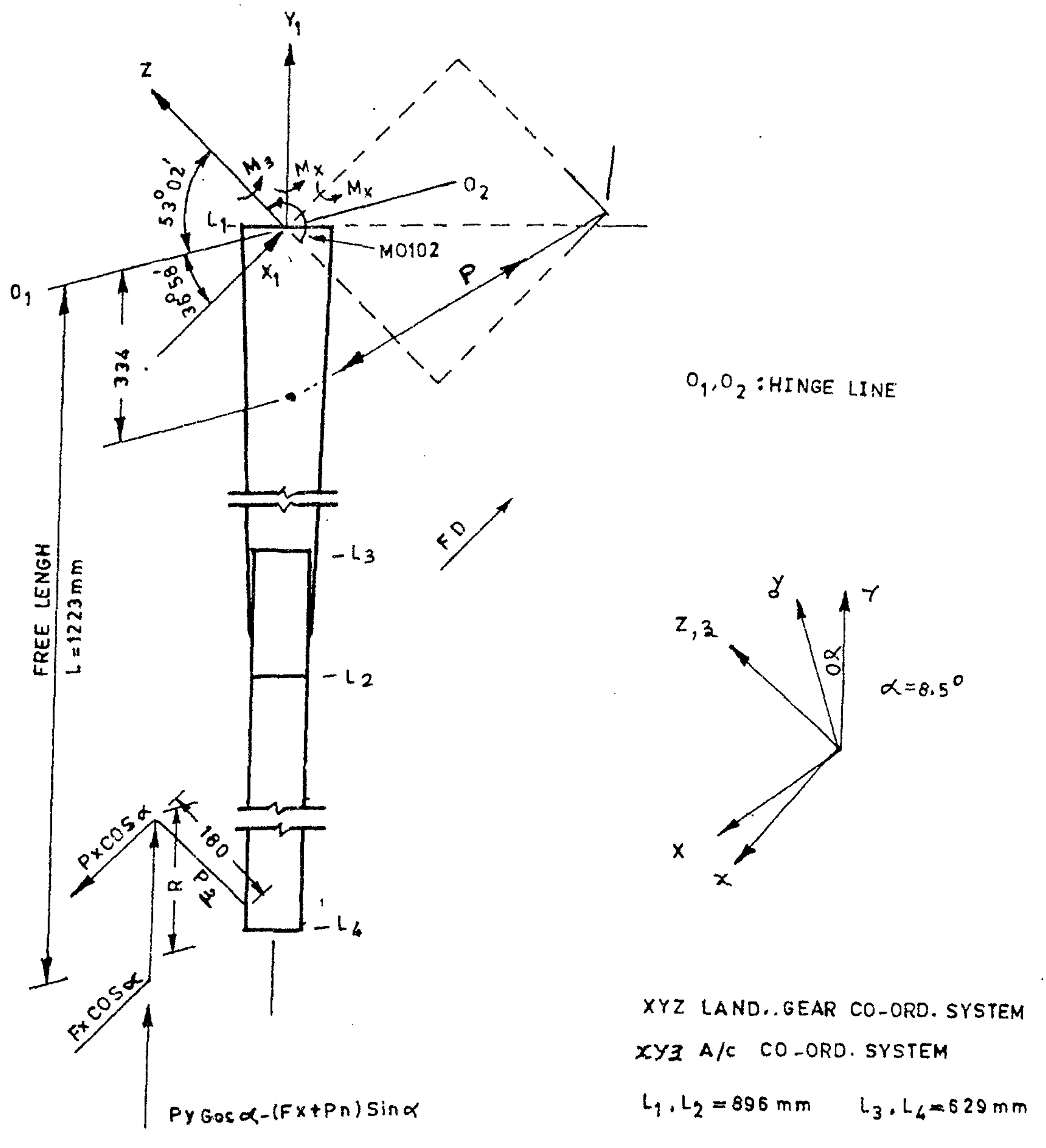


Fig.5 FORCES AND REACTIONS MAIN LAND.GEAR

HARDWARE AND SOFTWARE FOR FUNDAMENTAL RESEARCH IN FATIGUE

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Fatigue failure under multiaxial cyclic loading and effects of loads with relatively small amplitudes are of basic ones for fatigue fundamental research. To carry out required investigations in TsAGI hardware and software were developed that provided some fundamental works in these directions.

For research under multiaxial cyclic loading an electrohydraulic rig DAKON was created allowing testing of tubular specimens under simultaneous action of axial load P, torque M_t and internal pressure p ($P = \pm 400$ kN, $M_t = \pm 1000$ kN • m, $p = 100$ MPa). An original design to get and to control high pressure gave us a possibility to use industrial hydraulic aggregates without an expensive multiplier. The usage of industrial hydraulic aggregates provides high reliability of this system, simplicity of its manufacturing using features of a scientific or factory laboratory and accordingly low cost of its development. Moreover this design allows to control high pressure in large volumes, that is very difficult to provide using a multiplier. The design gives us a possibility to get cyclic pressure up to 2 Hz and to test without time limitations. At usage of a multiplier it is impossible because of oil leakage. Computer control and software provided unique experiments allowing investigations of failure criteria under multiaxial cyclic loading including ones, where fatigue failure was got owing to rotation of surface of principal stresses with constant modules of principal stresses [1]. Conducted tests have shown high reliability of developed system, simplicity of its service. Accuracy of high pressure control was 1% of range used.

The one of main problems at experimental estimations of small load effect is to provide reliability of fatigue test results. We have an experience when due to random two-times lowering of test loads, that was not detected at periodic load monitoring using oscilloscope, fatigue lives of specimens were drastically increased. Even for test equipment of leading manufacturers (MTS, Schenk and so on) guaranteed accuracy is 1—3% load range taken for a test. For loads with relatively small amplitudes those are 5—10% of this range accuracy will be 10—60% range of these loads. Taking into account that part of damage caused by these loads is up to 50% whole damage it is easy to estimate that fatigue life deviations from mean values due to this reason may be up two times.

The second problem of this science direction is a dependence of small load effect on load spectra. Together with theoretical methods based on local stress-strain approach at fatigue failure point according to GECON method [2] an approach to a forming of typical service load conditions was developed providing programs for laboratory tests [3]. Tests using these programs are a base to evaluate of parameters of so called "design S—N curves" that allow us to estimate with high accuracy some required values of equivalents between various service load conditions and fatigue life at full-scale fatigue test programs [4].

To estimate experimentally an effect of loads with relatively small amplitudes on computer controlled electrohydraulic machines software SAMUM has been developed allowing us to carry out fatigue tests at practically any load sequences. The main distinction of this software is a log of real cyclic load conditions as a "rainflow matrix". The usage of "design S—N curves" in this software gives us a possibility to evaluate an effect of loads with relatively small amplitudes from tests with load conditions similar to service ones. Moreover the usage as load conditions a real load history instead of given load history (usually taken) allows us to increase by an order the metrological accuracy of test at random and program loading with small amplitudes.

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DTIC COULD NOT GET MISSING

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DOMESTIC AND FOREIGN AIRCRAFT AVIONICS CERTIFICATION PROCEDURES

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IAC Aviation Register

Lately a number of documents specifying the civil aircraft components certification procedure has been developed. The main of them are the Aviation Rules AR-21 (Section 9) and Advisory letters of IAC AR Nos. 10-94 and 11-95. In the process of the a/m documents development the attempt was made to harmonize the certification procedures under development and those applied in USA and West Europe, preserving the peculiarities of Russian approach to the components design and test processes. On the account of the above it was suggested to assign all the components to two categories: A and B.

The category "A" comprises the components significantly influencing the aircraft airworthiness level as a whole. The table shows the relationship between the suggested classification and those applied in the West and preceding Certification Rules.

AR - 21	Analogue	
	FAR - 21	1989 Certification Procedures
Category "A" Components	Products approved under TSO	Components to be certified "prior to installation on an aircraft"
Category "B" Components	Products approved under RMA or certified on an aircraft	Components certified onboard an aircraft

The introduced procedures require, that any component to be installed on an aircraft under certification shall have an approval. This means that the product has undergone necessary test procedures conducted with the participation of the independent experts, its generic design is specified and suits the requirements set. The category "A" components approval is issued by IAC AR, and the category "B" components are approved by the General Designer Independent Expert Group.

Let's consider the Category A components approval procedures. Here it should be noted that these procedures are a little bit different for the component made in CIS and made in the West.

Types of Approvals by IAC AR for the Category A Components

Category A Component Approval issued by IAC AR

- Component Worthiness Certificate
- IAC AR Letter of Approval

A Worthiness Certificate is issued for the CIS-made developed components in case:

- the component is installed on an aircraft under certification (or already certified aircraft) and is qualified as a Category A Component in compliance with the IAC AR Advisory Letter N11-95,
- the component is designated (or may be suitable) for the installation onboard of the two or more types of aircraft,
- the Developer of the component applied for the Worthiness Certificate.

The Letter of Approval is issued for the CIS-made components in case:

- the component is installed on an aircraft under certification (or already certified aircraft) and is qualified as the Category A component in compliance with the IAC AR Advisory Letter N11-95;
- the component is designated for the installation onboard of a particular type of an aircraft;
- the component is in a full-scale production as a Category A component.

All Western-made components (except the standard ones) shall be approved by IAC AR

Worthiness certificate is issued for the Western-made component:

- if it is designated for the sales on CIS-market;
- if it complies with the requirements of the Qualification Basis approved by IAC AR.

Letter of approval is issued for the Western-made component:

- if it is designated for the installation on a particular aircraft type and complies with the General Designer Requirements.

Lately the design bureaus in compliance with the operators requirements install the Western-made components, especially electronics, onboard an aircraft. So the Traffic Collision Avoidance System TCAS - II produced by various USA companies is installed onboard IL-86, IL-96, IL-62, TU-154, and many other aircraft and helicopters are equipped with the Satellite Navigation System GPS, etc.

IAC Aviation Register pays much attention to the strict adherence to the installation procedure of the Western-made components onboard an aircraft in order to prevent the installation on an aircraft of the equipment that does not comply with the Airworthiness Requirements. The procedure of obtaining the IAC AR Approval for the Western-made components is specified in the Advisory Letter N10-94. The main stages of this process are stated in the table below:

Stage	Activities	Responsible
1. Specification of the Component Standard Configuration (Design)	a) Specify the Drawing Number and Drawing Revision Level (to obtain the Letter of Approval) b) Discuss with the Component Developer the decision to obtain Component Worthiness Certificate	
2. The procedure to obtain Worthiness Certificate	a) Send to IAC AR the Application accompanied by: - Components Requirements Specification (DDP) - the approval of the Aviation Authorities of the country of origin - Manuals	Component Developer
	b) Qualification Basis development c) Information/Notification of the Application Acceptance d) Additional test performance (under the supervision of the Developer Aviation Authorities or IAC AR) e) Drawing Number and Revision Level Specification f) Presentation to IAC AR of the Add. Test Report and Declaration of Design and Performance	IAC AR (Expert Group)
	j) Worthiness Certificate generation	IAC AR
3. Procedure to obtain the Letter of Approval	a) To send the Application to IAC AR accompanied by: - DDP; - Developer Aviation Authorities approval; - Manuals; - Requirements, the compliance with which shall be approved by IAC AR; - information necessary to enquire of the Airworthiness Approval Tag by IAC AR;	General Designer
	b) To send the enquiry to the Component Developer Aviation Authorities; c) To inform the General Designer of the possible date to start the certification tests of	IAC AR

	an aircraft with the component; d) To perform certification tests of an aircraft with the component; e) To present the Certification Test Report and the Declaration of Design and Performance to IAC AR; f) To issue a Letter of Approval	General Designer IAC AR

I would like to draw special attention to the stage of making a decision of the installation of the imported component on an aircraft. To our regret some design bureaus and operators purchase the equipment to be installed on Russian aircraft only on the basis of the Advertisements of the Western firms without considering the peculiarities of the Western equipment production procedures. In some cases it leads to the fact, that the equipment installed operates ineffectively onboard an aircraft or there appear unplanned limitations of its operation.

When the General Designer makes a decision to install onboard the developed aircraft a West-made component, the most important task becomes the accurate specification of the component standard design.

To provide this it is necessary:

1. To analyze a component performance, their suitability for the certified aircraft and the component performance with the Aircraft Certification Basis Requirements;

2. To analyze the environmental effect levels for which the component has been tested, their sufficiency for the aircraft and compliance with the Certification Basis (NLGS-3 Suppl. 8);

3. To clarify if the Supplier Country Aviation Authorities Approval exists and the type of this Approval;

4. To specify the additional qualification test activities necessary to obtain the Worthiness Certificate or the Letter of Approval on the basis of the activities stated in par. 1, 2, 3;

5. To decide on the type of the IAC AR Approval necessary:

- Worthiness Certificate;
- Letter of Approval (If the decision to obtain a Worthiness Certificate is absent, coordinate this item with the Component Developer);

6. To obtain the Worthiness certificate the component developer sends the application to IAC AR and performs other procedures specified in Advisory Letter N10-94;

7. To obtain the IAC AR Letter of Approval, if no additional qualification tests are required, it is necessary:

7.1 To specify the drawing number for the component installed on an aircraft;

7.2 To specify the component drawing revision level (If the additional qualification tests are necessary, the decisions on par. 7.1;

7.3 To execute the procedure specified in the IAC AR Advisory Letter N10-94.

The home-made component approval procedure is specified in Section 9 of AR-21. The main stages of this procedure are shown in the table.

Stage	Activities	Responsible
1. The component Classification	To divide all the components into 3 Categories: - Category A Component - Category B Component (Classification Criteria are stated in IAC AR Letter N11-95) - Foreign Components	General Designer Mock-up commission
2. Application Preparation	a) To develop the components Specification, Qualification Basis Project plan; b) To make a decision on the necessary type of IAC AR Approval; c) To send an application to IAC AR; d) Certification Center Appointment; e) Notification of the Application Acceptance	Component Developer Independent Experts IAC AR
3. Component mock-up	a) Qualification Basis Development and Coordination; b) Qualification Plan Development and Coordination; c) Development of the Qualification Basis Table of Compliance; d) Qualification Basis and Qualification Plan Approval	Component Developer, Certification Center IAC AR
4. Qualification Tests	a) Qualification Test Plan development and Approval; b) The Qualification tests performance in accordance with the procedures specified by the current standards; c) To develop and send to IAC AR: - Qualification Test Report; - Qualification Basis Requirements Table of Compliance; - component operation	Component Developer Certification Center IAC AR Component Developer Certification Center Independent Experts

	manuals; - Declaration of Design and Performance; - Notification	
5. Generation of Worthiness Certificate (Letter of Approval)	a) Declaration of Design and Performance Approval; b) Qualification Test Report Approval; c) Worthiness Certificate generation	IAC AR

In conclusion several words should be said about the category B component Approval. The approval of the Category B component by the General Designer Independent Expert Commission means, that the components have passed the necessary tests and this fact is certified by the independent Expert Commission at the component manufacturer enterprise and the component operates on an aircraft in compliance with the restrictions specified in design documentation of the Developer. The approval for all Category B Components shall be presented by the General Designer to IAC AR prior to the certification tests of the aircraft.

CERTIFICATION PRACTICE FOR PASSENGER AIRCRAFT AVIONICS IN RUSSIA

(NIIAO experience in the certification of flight management and navigation complexes)

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Certification of integrated digital avionics complexes is a difficult and time and labor consuming problem. This is due to both the complexity of up-to-date avionics and the wide range of requirements to avionics fail-safety developed on the basis of rich domestic and foreign experience of civil transport aircraft operation.

At present in Russia the process of harmonizing Russian and Western airworthiness requirements is taking place and Aviation Rules , AR-21, AR-25 (for transport aircraft), have been carried into effect. These rules take into consideration both Russian and Western practice of airworthiness acceptance for aircraft and their components.

Along with the harmonization of airworthiness requirements the active penetration of advanced Western technologies and components into equipment installed on new and upgraded aircraft should be noted. Upgraded aircraft IL-96 and TU-204 and a new aircraft BE-200 may be considered as an example.

Now the process of certification of integrated avionics complexes will be discussed taking into account the processes mentioned above and the experience gained by NIIAO in carrying out similar work for the integrated complexes of standard flight management and navigation equipment for the IL-96-300 and TU-204 aircraft.

These aircraft were certified for compliance with applicable Russian Airworthiness Requirements (NLGS-3) and other regulatory documents.

The compliance of a complex with certification airworthiness requirements is shown as a result of the following activities:

- qualification tests of components before installation on aircraft;
- airworthiness acceptance for components in aircraft.

Component qualification work relies on the List of Components certified/qualified before installation on aircraft. The List is developed by the General Designer of an aircraft. This document contains all aircraft components to be approved by the Aviation Register before installation on A/C. In addition, the List contains information on test types and environmental categories.

Component certification work relies on the List of Documentation to be submitted to the Aviation Register to obtain a Type Certificate. The List is also prepared by the General Designer of an aircraft. This List contains a variety of documents (estimates, circuit analyses, mathematical simulation results, etc.) and rig and flight test results. In addition, avionics fail-safety estimation is also very important.

A program is developed for the purpose of coordinating avionics certification work. The main stages of the program are as follows:

- Approve certification requirements (basis) by the IAC AR;
- Prepare the List of Components;
- Analyse and agree the List of Components;
- Develop Qualification Test Program;

- Perform qualification tests;
- Prepare and provide the developer with the List of Documentation for certification of avionics in A/C;
- Mathematical simulation of the basic functions of an integrated avionics complex;
- Evaluate avionics complex fail-safety;
- Evaluate avionics operation at the rig;
- Prepare analytical documentation for avionics;
- Support flight tests of avionics in A/C;
- Revise documentation and support the General Designer to obtain a type certificate.

Airworthiness requirements for an aircraft and its components (Certification Basis) and Means of Compliance are specified at the early certification stage. The avionics developer, using the certification requirements, develops and implements the Components Qualification Program which includes the following tests:

- Environmental tests
- Explosion proofness tests
- Electromagnetic compatibility tests
- Power supply tests
- Software tests for components that use computers.

If formal deviations from certification requirements specified for components are detected, but in fact components satisfy these requirements, equivalent compliance documents are prepared.

Airworthiness acceptance for an integrated complex on an aircraft is supported by:

- mathematical simulation of different flight modes and stages;
- analysing circuits, channels for altitude and speed parameters as well as parameters that define aircraft attitude;
- validating equipment configuration;
- determining fail-safety of an avionics complex;
- rig and flight tests.

Complex operation is tested during rig tests to show its compliance with separate sections of airworthiness requirements relevant to avionics. Fail-safety of an avionics complex is evaluated by several independent crews as experts using a simulation rig and this evaluation requires a great amount of effort.

Flight tests of equipment on aircraft are an important part of the certification process. Positive results of flight tests demonstrate safety of avionics complex operation most conclusively.

Nevertheless, powerful computational resources required for mathematical simulation, and software verification, subscale simulation rigs, EMC rigs, mechanical and climatic test rigs were used to provide the successful certification of the flight management and navigation complex.

The certification process results in developing a large package of documents to be submitted to the Aviation Register. For certification before installation on aircraft, the AR Letters of Approval are executed for more than thirty flight management, navigation and communication systems on the basis of a considerable body of test reports.

The equipment developer proves compliance with requirements specified in over 150 paragraphs of NLGS/AR to support certification of equipment on aircraft. In addition, the developer independently prepares supporting documentation for over 90 paragraphs and develops over 40 analytical documents and 20 rig test reports. Moreover, the developer takes part in preparing flight test reports for over 50 paragraphs of airworthiness requirements.

The certification procedures considered for avionics of home-produced aircraft are aimed to provide the conclusive and valid compliance with airworthiness requirements developed based on long-time practice of transport aircraft operation. It should be noted that the time has yet come to develop and apply the methods of objective quantitative estimation of a level of safety along with traditional methods. The foundations are laid in both Russian, acceptable means of compliance tried in certifying the IL-96 and TU-204 aircraft and Western regulatory documents such as FAA AC. To start this work it is necessary to clearly define and present the term "Flight Safety", or "Level flight Safety", as a state vector. The dynamics of the state vector is to be described by a corresponding differential equation. The solution of the vector equation preset for determining a level of safety can give an objective quantitative estimate of flight safety.

Such an approach may be used at all stages of the aircraft life cycle, both in design and tests and operation that enables objective comparison and prediction of safety levels to be made for aircraft of the same or different classes. In additional, safety may be estimated and predicted for different flight stages.

The above process of avionics certification is generally similar to that implemented by the foreign companies.

In conclusion, some features of present-day avionics certification may by noted:

- new Aviation Rules come into effect and any practice of their use is unavailable, particularly as to requirements based on FAR-25;
- foreign-produced components are certified during purchase and installation on aircraft;
- some regulatory documents such as certification procedures for jointly developed and produced components and equipment of cabinet/module, design, are unavailable;
- wide utilization of test rigs is required to support the certification process that is caused by increasing avionics complexity;
- it is necessary to develop and use new methods of estimating flight safety levels together with traditional methods.

Civil Aircraft Airborne Systems Software Certification

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Airborne Systems Software Certification presents a rather new aspect of airborne equipment certification. In Russia and countries of CIS a standardizing document "Requirements to the Development, Test Procedures and Documenting of the Software for Airborne Systems Using Onboard Digital Computer" Appendix 8.1.6 being a Supplement to Airworthiness Requirements NLGS-3 is used since 1991. The document was compiled by the Aviation Register Working Group WG-23 and designated first of all for IL-96, TU-204 and IL-114 aircraft.

The document was based on the requirements of DO-178A, domestic United System of Design Documentation and NLGS-3 standards.

At present foreign and domestic software certification uses an approach based on performing specified development, test and documenting procedures depending on the software declared level. Fig. 1 presents the description of failure conditions, criticality categories and levels of the software in compliance with Airworthiness Requirements (NLGS-3, FAR-25, Supplement 25) and Supplement 8.1.6, DO-178A and DO-178B Standards.

The comparing analysis of the documents: Supplement 8.1.6 and DO-178A shows, that the level 1 specification of Suppl. 8.1.6. has been defined with "a reserve", what is due to the absence of the domestic experience in software certification. The price we pay for such an approach is the increase of software test procedures for the systems which failure may lead to an emergency situation. The IL-96 and TU-204 aircraft flight and navigation equipment and aircraft systems assignment to an appropriate software level is presented on Fig. 2. A rather large number of the systems having 1 level software is explained by the a/m definition of this level.

During IL-96 and TU-204 avionics software certification a group of independent Experts evaluated the following aspects:

- software development procedures;
- Software Design/Requirements Specification Compliance;
- Program/Software Design Compliance;
- quality and sufficiency of the testing of modules and their cooperation;
- Software/Hardware Cooperation Test Procedures and Results;
- Integrity provision procedures (Configuration management) and Software quality assurance procedures.

The first experience on software certification allowed to make the following conclusions:

- the Suppl. 8.1.6 requirements helped to increase the software development and test levels, and first of all this concerns the critical systems;
- to use the resources more efficiently it is necessary to revise the aspects of software development management, paying more attention to verification and quality

assurance tests conducted by the programmers and experts independent of the developer;

- the existing methodology and instruments require systematization and their perfection to the recommended application level.

At present the improvement of the standardization and methodological basis in the sphere of avionics software certification is directed to the coordination of the domestic and western Airworthiness Requirements. In the nearest future it is planed to issue a standardizing document for the new generation of civil aircraft and helicopters, which will fully comply with the DO-178A, and using the experience in the certification of ARIA-200 avionics complex to prepare the foundation for introducing the DO-178B standard.

Let us consider the software development process in compliance with the requirements and recommendations of DO-178B. Fig. 3 presents a full software life-cycle including its main stages:

- planning;
- S/W Requirements Specification (R);
- S/W Design Development (D);
- Coding (C);
- Integration (I);
- Final verification stage;
- installation on an aircraft;
- S/W operation supervision;

and the process/activities performed:

- Software Development Management;
- S/W development;
- S/W verification;
- S/W Configuration Management;
- S/W Quality Assurance;
- Certification Authority Activities coordination.

It should be noted that a unified understanding of the S/W life-cycle, especially at its initial development stage, makes up the basis of successful cooperation of the Customer, developer and Certification Authority as far as planning and finances are concerned.

The modern tendency of the onboard systems software development procedure demands the increase of the level of the requirements to the activities that accompany the S/W development itself, i.e.- to S/W verification, configuration management and quality assurance. While evaluating a number of S/W Development Projects it was found out that the accompanying activities costs constitute approximately half the total S/W Design costs.

The next tendency is the transition to the CASE S/W Development Technology based on the usage of the integrated tools, supporting the development and associated activities. So the development of S/W design for ARIA-200 avionics complex in cooperation with AlliedSignal (USA) and Sextant (France) is supported by such object-oriented programming facilities/tools, as STP, STOOD, VAPS, FPD, and as far as the configuration management is concerned: SCCS(SUN), DCS(PCIBM) and PALAS-X.

The application of the a/m tools and facilities requires to perform new kinds of activities during S/W Certification, i.e.- qualification (certification) of auxiliary programming tools. And the requirements here are such, that the supporting tools S/W level shall be not lower, than the level of the avionics software. The avionics software certification may be replaced by the supporting tools qualification only if those supporting tools costs are cheap, what in its turn becomes possible only if they are widely used, for e.g. for several projects.

Airworthiness Requirements Russian NLGS-3 abnormal situations	critical	emergency	complicated (major)	degraded flight conditions (minor)
FAR-25 $25.1309 event probability	extremely improbably	improbable	improbable	probable
AP-25 abnormal situations-event probability	critical situation-extremely improbable	emergency situation-improbable	complicated conditions-improbable	complicated conditions-improbable
Suppl. 8.1.6 System Criticality Category S/W Level	critical 1	critical 1	essential 2	non-essential 3
DO-178A System Criticality Category S/W Level	critical 1	essential 2	essential 2	non-essential 3
DO-178B Failure Conditions Category S/W Level	catastrophic A	hazardous / severe B	major C	minor D
Event probability 1/flighthour	10^{-9}	10^{-7}	10^{-5}	

Fig. 1. Failure Conditions, S/W Categories and Levels

IL-96			TU-204			
System						
ADS-85			Digital Thrust Control System-85			
FMS-85			ADS-85			
FCC-85			FMS-85			
I-42-1C	GPWS-85		FCC-85	GPWS-85		
FFIS-85	Contrast Control Panel-85		I-42-1C	Flight Envelope Protection System-85		
ILS-85	LRRNS-85	FLIS-85	EFIS-85	WXR-85		
RA-85	WXR-85	VOR-85	ILS-85	LRNS-85	FLIS-85	
EICAS-85	A-331	Electronic Chronometer -85	RA-85	A-331	VOR-85	
Automatic Stability and Controllability System-96	DME-85	Magnetic Parameter Recording System A-02	EFIS/EICAS -85	DME-85	Electronic Chronometer-85	
Ground Proximity Warning System-8	ADF-85		Automatic Stick Control System-204	ADF-85	Magnetic Parameter Recording System-85	
Engine Electronic Control-90	Engine Control α Monitoring System-90	Centralized Engine Control System-2M	GPWS6/GP α Collision Avoidance Systems	Engine Control α Monitoring System-90	Engine Control System	
			Engine Electronic Control-90			
1	2	3	1	2	3	Software Levels

Fig. 2: IL-96 and TU-204 System Software Level

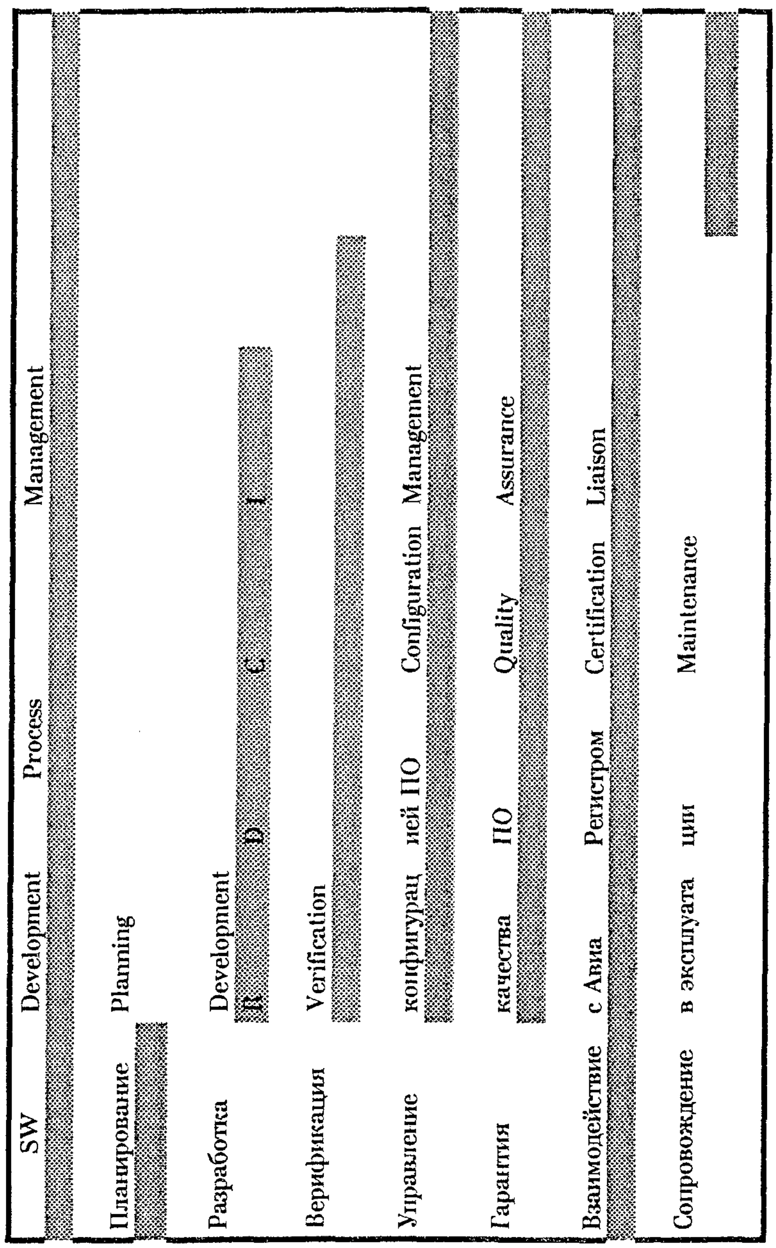

Fig.3: S/W Development Life-Cycle

AIRWORTHINESS TEST PROCEDURES AND FACILITIES FOR POWER SYSTEMS

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The aircraft and helicopter onboard equipment under its minimum cost, weight and volume should provide its specified output performance under rated duty. That is one of the main foreign and domestic airworthiness requirements.

The present paper deals with the problem of compatibility between avionics and aircraft power systems (PS) over equipment power supply circuits.

By avionics and PS compatibility we should understand the avionics capability to operate with the specified output performance under the changes of the power supply parameters, such as voltage and frequency, phase imbalance and voltage curve form modulation within the limits specified by Airworthiness Requirements for normal and emergency operation of PS.

If power supply parameters comply with the conditions of abnormal (uncontrollable) operation of the PS, avionics:

- should comply with the requirements of the equipment performance standards, if such are available;
- can seize operation for the period of abnormal PS operation;
- should automatically recover its output characteristics after the resumption of normal or emergency mode of PS operation;
- should not be a source of emergency situations or hazardous operating modes of aircraft units.

PS is described as having the lowest and avionics as having the greatest cost, weight and volume under respectively wide range of power supply parameter modulations, that is why the power supply parameters which are described as close to optimal are standardized in domestic and foreign equipment performance standards [See Ref. 3,5,6,7].

PS and avionics compatibility over power supply circuits will be provided, if:

- nominal PS and avionics voltage and frequency values are the same, and the allowable power parameters variations or their distortion values at the avionics terminals do not exceed the relative allowable modulations or distortions accepted for PS.

To certify the avionics compliance with the Airworthiness Requirements concerning the input power supply it is necessary to perform the below described tests prior the installation of the avionics on an aircraft.

PS Normal and Emergency Operating Mode

Avionics should pass the tests under the following voltages:

• for the equipment having 115 V, 400 Hz, 108 and 119 V, 380 and 420 Hz, if this equipment is classified as the 2nd class receivers, and 102 and 124 V, 360 and 440 Hz, if it is classified as the 1st class receivers;

• for the equipment consuming 27 (28) VDC 24 and 29, 4 V, if it is described as the 2nd class receivers, and 18 and 31 V, if it is described as the 1st class receivers.

There exist some differences between the requirements of Russian Suppl.1.5 and DO-160C [3,6] standards concerning the a/m test types.

To comply with the foreign equipment performance standards the home-produced equipment having passed the tests for steady-state voltage and frequency effects in compliance with the Russian Suppl.8.1.5 Requirements should undergo additional tests:

• for 2nd class equipment at 104 and 122 V, 380 and 420 Hz over 115 V circuits, 400 Hz and 228 V over 27 (28) VDC circuits.

The foreign-made avionics tested for steady voltage effects in compliance with the RTCA/DO-160C or ISO 7137 Requirements Standards will fully suit the Suppl. 8.1.5 Requirements, if it is tested additionally:

• at 102 and 124 V, 360 and 440 Hz over 115 V circuits;
• 400 Hz and 31 V over 27 (28) VDC circuits.

The three-phase equipment shall undergo tests for phase voltage imbalance:

• 3 V for 2nd class receivers;
• 4 V for 1st class receivers under average phase voltage level of 113...117 V and at 390...410 Hz.

The diagram of Fig. 1 allows to simulate the amplitude modulation and a.c. voltage sinusoidal modulation, and the d.c. voltage fluctuations.

To simulate a.c. or d.c. voltage curve for the particular frequency component one hardware set including standard signal generator, power amplifier and transformer (see fig.) is sufficient. Generally the supply voltage curve may have several frequency components of its form modulation, that is why in principle the corresponding number of hardware sets may be used, each of which will contain three signal generation components, power amplifier and transformer.

However in most cases it is sufficient to simulate one sinusoidal modulation frequency component of the a.c. voltage curve or d.c. voltage fluctuation.

Then modifying the frequency by signal generator and amplitude by power amplifier it is possible to test the influence of any specified frequency component of the voltage curve form modulation on the tested avionics.

As the requirements for amplitude modulation and voltage imbalance at hardware terminals stated in foreign and domestic airworthiness requirements are the same, the test procedures to confirm the home-produced and foreign equipment compliance with these requirements are approximately similar.

As far as the test for the effects of the frequency components of the voltage curve form sinusoidal modulation is concerned, it shall have some specifics, depicting the difference between the foreign and home requirements, stated in Table 1.

Table 1

Standardization Documents	Total higher harmonic quantity	Each harmonic	Amplitude
Domestic: Supplement 8.1.5	< 8%	< 5%	1,41 ± 0,15
Foreign DO-160C, ISO 7137	< 5%	< 4%	1,41 ± 0,10

The tolerable d.c. voltage fluctuation amplitude level (of the alternate component) is the same in domestic and foreign equipment performance standards and should not exceed + 2 V.

The tolerable d.c. voltage frequency component amplitudes, specified in Suppl. 8.1.5 and DO-160C (ISO 7137) for the frequency range of 0,2 to 150 kHz, are practically the same.

The tolerable voltage variation frequency components within the frequency range of 0,01 to 0,2 kHz constitute:

0.9...0,32 V in compliance with Suppl. 8.1.5 and

0,2 V in compliance with DO-160C (ISO 7137).

Taking into account that the voltage variation frequency components within the frequency range of 0,1...0,2 kHz in secondary PS are defined by the voltage amplitude modulation at the input of the rectifiers, and assuming the maximum tolerable voltage modulation amplitude equal to 1,24 V [6,7], we receive:

$$U_{actual} = 1{,}24 \times \frac{28}{115} = 0{,}302 \text{ V}$$

Considering the obtained result it is necessary to conduct tests for voltage variation frequency component effects in compliance with Fig. 16-5 of the DO-160 Standard changing the voltage modulation frequency component values within the frequency range of 0,01...0,2 kHz from 0,2 V to 0,32 V.

Voltage curve is also described by the presence of voltage pulses. Voltage pulses or switching overvoltages, as they are referred to, are occasional. if voltage frequency components are periodical, then voltage pulses occur at switching off the load including inductive circuits. Foreign and domestic equipment requirements standards specify practically the same requirements for voltage pulse generation. In compliance with them voltage pulses shall not exceed ± 60 V, and internal impedance of the pulse source shall be not less then 50 Ohm. This specifies practically the same test procedures to test the equipment susceptible to the voltage pulses presented in the domestic and foreign equipment performance standards [Ref. 3,5,6].

Additionally to the above the domestic equipment performance standards specify tests for:

• effects of ± 70 V voltage pulses for equipment consuming 115 V, 400 Hz,

and

• effects of ± 50 V voltage pulses for equipment consuming 27 (28) VDC.

Foreign category B equipment standards specify the tests for effects of intermittent transient voltages ± 50 V (+78 V and - 22 V) with the frequency of 5...10kHz for equipment consuming 28(27) VDC and iterate transient voltages 400 V (+ 400 and - 160 V) with the frequency of 50 kHz for equipment consuming 115 V, 400 kHz [6,7].

Taking into account that:

• domestic equipment performance standards do not specify the internal + 50 and - 70 V voltage pulse sources impedance values [3,5], but foreign standards require the tests of the category B equipment designated for the insertion into the circuits where the reduced voltage pulse protection level is accepted, it becomes clear, that these tests are of secondary importance in the certification test list over power supply. Such a conclusion is backed by the facts, that:

• ± 50 or ± 70 V voltage pulse sources cannot have the internal impedance value approximating zero (see above), but it is the result of the tested avionics impedance, pulse generator internal impedance and switching facility terminals impedance;

• it is not necessary to test the category B equipment [6,7] for the effects of intermittent or iterative transients as it is not susceptible to them (see description of Category B equipment [6,7]).

Besides the insulator resistance of the home-produced and foreign equipment should be tested under 500 and 1000 V during 1 minute for the equipment designed for nominal voltages of 27 (28) and 115 V [3,7].

In addition to the above the PS normal operation is characterized also by such power parameters as power interrupt and transient voltage variations.

Power interrupt is described by the drops of the voltage at the avionics terminals beyond the accepted normal and emergency operating mode limits.

The power interrupt at the avionics terminals under normal operation is limited to:

80 ms - by domestic standards [3,5]
200 ms - by foreign standards [6,7].

In compliance with the domestic and foreign equipment performance standards the power interrupt is simulated 3...5 times and after that the conclusion of the tested avionics capability to operate under power interrupts is made.

The attention should be paid to the more comprehensive tests of avionics having digital systems and/or memory devices specified in the foreign standards [6,7]. In compliance with these documents voltage is changed at different rates during tests - 15000, 5750, 2300 and 2000 V/s to reduce a.c. voltage and 115000, 2300 and 5750 V/s to increase the 115 VAC voltage linearly [6,7]. For 27 (28) VDC voltage the rates are correspondingly 115/27 (28) times lower. In accordance with the procedures accepted in this country voltage rise and fall time during power interrupt simulation does not exceed 1 ms. Under PS normal operating mode short-time (less than 0,1 ms) voltage departures beyond the limits are accepted for steady values, that is why domestic and foreign performance standards specify the tests for transient 160 and 60 VAC and 40 and 12 VDC voltage effects during 100 ms under Suppl. 8.1.5 and 30 ms under DO-160C. There exist some peculiarities of domestic and foreign requirements, but they are not principle.

Abnormal Operating Mode

The equipment requirements under abnormal PS operation are presented above. To make the conclusion of a certain sample compliance with the Airworthiness Requirements under abnormal PS operation, it should pass the tests for effects of:

- steady voltages of 97 and 134 V at the frequencies of 370 and 430 Hz a.c. respectively or 21 and 33 VDC;
- 180 VAC transient voltages and/or 45 V - for equipment powered by secondary d.c. PS;
- power supply interrupt of 7...10 s during which the voltage may fall in compliance with any pattern;
- reduced voltages equal (0,35 - 0,65) U_{nom} during a minute.

To perform all the tests described above it is necessary to have the following laboratory equipment:

1. Power sources capable to change a.c. phase voltage from 40,0 to 500 Hz, and d.c. voltage from 8 to 100 V;

2. Hardware capable to reproduce the assigned voltage curve modulation mentioned before;

3. Voltage pulse generation devices;

4. Devices reproducing the power supply interrupts of the specified duration;

5. Devices monitoring voltage, frequency, voltage modulation frequency components, including pulses, power supply interrupt duration, pulse and test modes.

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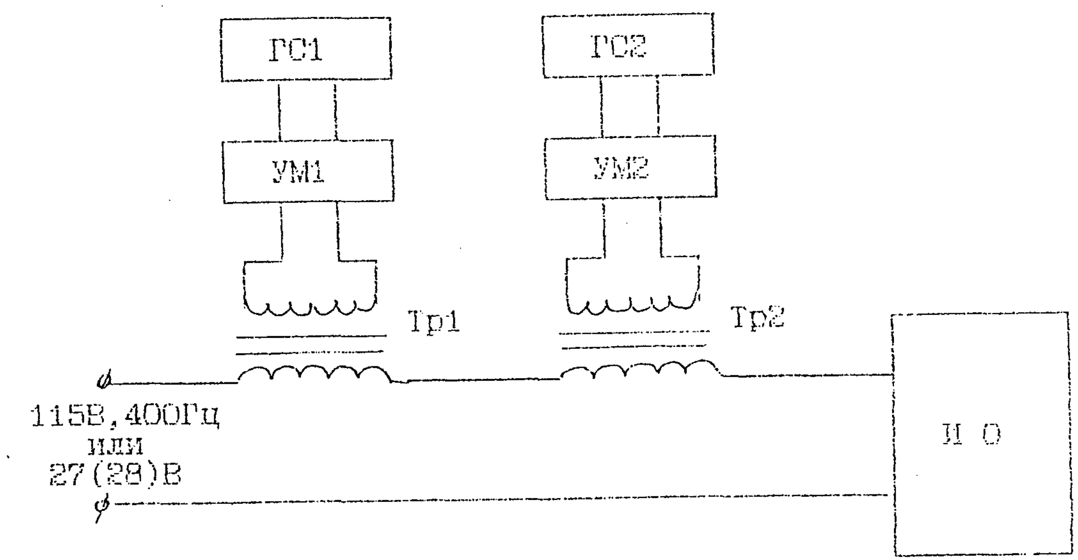

Fig. 1: Hardware set layout to test the effects of voltage curve modulations / distortions on the tested avionics:

1- signal generator 1; 2- signal generator 2; 3- power amplifier 1; 4- power amplifier 2; 5- transformer 1; 6- transformer 2; 7- 115V, 400Hz or 27(28)V; 8- tested equipment.

AVIONICS EMC TEST PROCEDURES and FACILITIES

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The problem of avionics electromagnetic compatibility (EMC) provision presents an important part of engineering, testing and operation of modern aircraft. EMC engineers cooperate with aircraft and avionics developers and designers in solving a complicated technical problem of avionics design and location within the limited space of the aircraft and in providing their simultaneous and interference-free operation.

Under the progress of aviation technologies and aircraft equipment development the EMC problem solution requires more and more efforts from the aircraft and avionics developers. Lately along with the tendency of onboard functional systems (FS) number growth, radio transmitters capability and radio facilities sensitivity increase, the new aspects that make this problem more complicated appear:

- the analogue hardware is replaced by the digital electronic circuits, the equipment susceptibility range increase from ones to hundreds MHz;
- the digital hardware is widely introduced into the systems that influence directly the flight safety. This fact significantly increases the influence and effects of electromagnetic incompatibility;
- EM field intensities on routes have significantly grown due to the increase in number and transmitting power of the ground and mobile radio transmitters of various purposes;
- the application of composite materials in manufacturing aircraft structures leads to the reduction of the fuselage external EM-fields shielding capability and lowers the hardware protection parameters;
- the integration of the onboard equipment stated the new tasks in EMC provision between the modules in the cabinet and modules protection from the external electromagnetic fields.

The present paper discusses the standardization procedures to be performed to test the modern aircraft avionics EMC during certification.

The aircraft avionics EMC depends upon the two main factors:

1) electromagnetic situation at the places of the location of the hardware packages, antenna and aerial/feeder lines, digit/analogue data exchange lines between the units of aircraft FSs, power supply circuits, etc...;

2) the aircraft FSs interference immunity, their susceptibility to EM effects.

In its turn the EM situation depends upon the inadvertent noises created by the aircraft equipment and external noise sources during operation and upon the attenuation provided by the propagation medium between the noise sources and the recipients (the hardware experiencing the influence of the inadvertent noises). Thus three categories of objects influencing the EMC onboard the aircraft may be distinguished:

- inadvertent noise sources;
- noise propagation medium;
- noise recipients.

Each inadvertent noise source may generate noises and each recipient accept the EM influence along three main channels:

- through hardware cases/packages;
- through interunit communication line wires, power supply and grounding circuit wires;
- through antennas (for radio facilities).

Taking into account the cross couplings appearing during the inadvertent noise propagation the main nine paths of the noise penetration into the recipient from the source may be distinguished:

- antenna-to-antenna, antenna-to wires, antenna-to-housing (case)/package;
- wire-to-antenna, wire-to-wire, wire-to-case;
- case-to-antenna, case-to-wire, case-to-case.

The structural diagram of the noise penetration paths onboard the aircraft from the source into the recipient is presented in Fig. 1.

The probability of the avionics operation quality reduction due to the noise penetration along the a/m paths lies in a vast range. For example, such paths as "antenna-to-antenna", "antenna-to-wire" and "wire-to-wire" produce up to 90% of the total incompatible situations onboard the aircraft.

Large-scale introduction of the digital hardware into radioelectronic, electronic and electric equipment leads to the growth of incompatible situations as far as the avionics is concerned.

The results of the noise effects on the digital aircraft equipment may be much more heavier than the effects on the radio facilities as it is used in the systems providing the safety of flight. The failure of one of these systems may have critical results. The noise effects on digital systems may be made worse due to the fact that the digital microcircuits do not return to their initial operating state themselves when the EM influence siezes (as is the case with the analogue systems).

Thus main attention while considering the aircraft EMC problem shall be given to the noise protection of the electronic systems performing "the critical" functions directly influencing the safety of flight. Under the FAA 25.1309-TA circular these systems comprise:

- power distribution systems;
- power supply systems;
- engine control electronic systems;
- IFR flight instrument navigation systems;
- emergency annunciation and warning systems;

• external interwiring to supply power, initiate signals and provide control functions;

• displays;

• control panels and communication facilities to provide connection with other systems, etc.

There exists the notion of the **necessary** and **sufficient** conditions for solving the EMC problem onboard the aircraft. The necessary condition for the solution of the problem is the avionics compliance with the EMC requirements concerning the noise generation and noise susceptibility. However, the implementation of these requirements does not ensure the EMC problem solution. The unsuccessful arrangement of the equipment, aerial/feeder lines, avionics communication lines, their power supply circuits and aircraft bonding may lead to (and often results in) the appearance of the situations of the incompatibility even of the high quality equipment. However, the compliance of the equipment with the EMC requirements forms a **necessary basis** for the successful EMC problem solution onboard the aircraft. The ***sufficient*** conditions for solving the EMC problem onboard the aircraft comprise:

• equipment and airplane compliance with the EMC parameters requirements and rational (as far as the EMC is concerned) arrangement of the equipment onboard the aircraft.

Avionics EMC requirements comprise two main aspects: susceptibility and noise generation. Each aspect consists of the two parameter categories. One category describes the antenna input parameter of the radio facilities, and the other - EMC parameters of radio, electronic and other devices besides antenna. They are defined in compliance with the anticipated electromagnetic situation on the aircraft. This situation depends upon the avionics configuration, its location onboard the aircraft and external HIRF on the flight route. These requirements form the equipment EMC qualification basis. It comprises the list of the documents specifying the following avionics parameter categories:

•noise generation and noise susceptibility (not including antennas);

• radio receivers frequency selectivity and the level of secondary radiation of the radio transmitters at the antenna input;

• structural components bonding and shielding.

The effective and timely solution of the EMC problems influences the aircraft and avionics performance levels, number of additional structural modifications during tests. For example: modifications of the aircraft structure to provide the necessary EMC at the stage of the primary design or later are much more expensive than at the initial project design stages. In case of sufficient and timely-made scientific researches all the costs to provide the aircraft EMC do not exceed 2...5% of the development costs.

The home-produce avionics EMC parameters are specified in the Russian Airworthiness Requirements (HLGS-3), supplement 8.1.4.1 [1] and Aviation Rules AR-25 (see Fig 2). The Supplement 8.1.4.1 Requirements comply with the DO-160B standard requirements (sections 18, 19, 20, 21), they specify the EMC parameters within the frequency range of 10 kHz to 1215 MHz.

Regarding the noise generation the avionics is tested along 3 EMC parameters:

1. Power supply circuits noise current and voltage;
2. Communication line noise current;
3. Noise field electrical component strength.

Regarding the avionics noise susceptibility the equipment is tested along 9 EMC parameters:

1. The avionics susceptibility to the influence of the magnetic field on the aircraft airframe;
2. The avionics cable connectors susceptibility to the sound frequency magnetic field interference;
3. The avionics cable connectors susceptibility to the electrical field interference;
4. The avionics cable connectors susceptibility to transients fields interference;
5. The avionics susceptibility to power supply wires radio frequency noises;
6. The avionics susceptibility to the power supply wires sound frequency noises;
7. The avionics susceptibility to the radio frequency noises within the communication line wiring;
8. The susceptibility to the radio frequency magnetic field influencing the avionics case and wiring;
9. The susceptibility to the radio frequency electromagnetic field influencing the avionics case and wiring.

Our industry, NIIAO has built-up the EMC test rig complex which allows to conduct avionics qualification tests in compliance with the requirements of Supplement 8.1.4.1 of the Russian Airworthiness Requirements (NLGS-3) and DO-160B Standard.

The rigs to monitor the avionics EMC parameters compliance comprise two groups:

1. The rigs to monitor the parameters of the noise generated in power supply circuits, communication lines and the electromagnetic field passing trough the avionics case and wiring (2 rigs);
2. The rigs to monitor the parameters of susceptibility to the noises in power supply circuits, communication lines and to electromagnetic field passing through the avionics case and wiring (6 rigs).

The Institute test rig complex is a shielded room of 7×7×9 meters where the a/m rigs are accommodated. They are supported by the executive computer, measuring sensors, generators and antennas produced by ROHDE and SCHWARZ. The qualification tests of the avionics complexes for IL-96, TU-204, IL-114 aircraft were performed on the basis of the a/m rigs.

The avionics integration stated new problems in the field of setting forth the requirements to the integrated systems EMC and in the field of their monitoring as well. When the hardware was developed on the basis of the units, it was located onboard the aircraft within special shelves or racks. These load-carrying structures did not fulfill any protective functions of shielding or noise filtering. All these functions were wholly implemented by the hardware cases.

Quite different situation arised under the cabinet/modular design concept applied during the development of the integrated avionics. The cabinet unlike the racks for the hardware units carries out a series of functions concerning the common functional systems modules protection from inadvertent noises:

- provision of the module external electromagnetic field shielding;
- filtering of the noises incoming from the external communication lines;
- filtering of the noises incoming from the power supply source wires.

Simultaneously the cabinet performs to protective functions regarding the reduction of the levels of the noises generated by its modules through the case and power supply and communication line wiring. Besides a some integrated equipment modules solve the problems simultaneously for several cabinet systems (for e.g.: I/O modules, data concentrator modules, power supply modules).

The existing avionics EMC parameter standards (NLGS-3, DO-160C) have been developed for the non-integrated hardware, there is no distribution of the protective functions between the unit configuration separate components. Under this condition the integrated hardware EMC requirements and their testing procedures generation can be carried out in several ways, for example:

1. by specifying separate requirements and test procedures for each module and cabinet;
2. by specifying the requirements and test procedures for each functional system within the cabinet.

The first mentioned way of standardization has a number of advantages:

- it allows to qualify modules and the cabinet as independent products, which is specially important when they are produced by different manufactures;
- it allows most effectively to conduct FS modules rig tests using such modern noise generation and susceptibility level monitoring facilities as TEM-chambers and horn-chamber.

The drawback of this standardization principle is the necessity to revise the existing standards and to differentiate them between the modules and the cabinet.

Today when no statistic database concerning the real level of protection of the modules and cabinets against noises exists the second way of the standardization is more preferable. There is no need to change EMC standards, but arise the difficulties in their testing. The fact is that any integrated FS may be tested on the rig only within the configuration of the expensive and bulky cabinet. This makes rig tests more comprehensive as the capabilities to use such effective electromagnetic noise generation facilities as TEM-chamber and horn-chamber are very limited.

An increasing sophistication of the external electromagnetic situation on the flight routes and the reduction of the aircraft structure composite materials components shielding parameters lead to the necessity of the avionics EMC parameters standards revision.

This kind of activity is carried out both abroad and in this country. Today the **DO-160C** standard fully coinciding with the DO-160B standard in sections 18, 19 and 21 and considerably changed in section 20 has been accepted abroad. The monitored standard frequency range has been broadened (from 1,215 GHz to 8 GHz), the hardware radiation field susceptibility standards have been raised from 2 V/m to 200 V/m versus the environmental category.

The special HIRF susceptibility requirements have been defined for the electronic systems that perform "critical functions" influencing directly the safety of flight. These requirements have been specified in the Advisory Circular RTCA AE4R-91-XX dated 15 July 1991 [3]. The standard testing procedures are also defined there.

Three ways to test the standard compliance are stated:

- direct aircraft exposure to HIRF;
- design-basis/experimental concept based on the aircraft exposure to LIRF and measuring the noise currents induced by those fields in the connecting wires of the "critical" systems with the further data recalculation for the expected HIRF levels;
- rig tests of the "critical" systems within the fields of 100 V/m in the range of 10 kHz to 18 GHz.

The first concept gives the most detailed picture of HIRF influence on the aircraft equipment, but has two significant drawbacks:

- the "critical" system susceptibility test results may be obtained only at, the last stage of the aircraft development, when any hardware or aircraft structure modifications are most expensive and labor-consuming;
- the "direct exposure" test procedure is highly expensive and labor-consuming.

The second way of testing is less expensive and labor-consuming than the first one, but has three significant drawbacks:

- the "critical" system susceptibility verification result becomes known only at the last aircraft development stage;
- the electromagnetic field penetration to the sensitive electronic circuits through the hardware case is not considered;
- the inserts in the communication lines and current sensors provide the significant measurement errors for the frequencies higher than 300 MHz.

The experience obtained while testing the hardware in compliance with the NLGS-3 requirements showed that in many case there was a significant susceptibility reserve (15-20DB) against the noises penetrating through wiring, but there was no reserve susceptibility for the fields penetrating through the hardware packages.

The third way of HIRF susceptibility verification has no mentioned drawbacks of the first two techniques, but it does not consider the real hardware location environmental conditions onboard the specific aircraft (i.e. circuit impendances, fuselage resonance parameters, etc.). It is designed for the statistic estimated operation condition model.

The essence of HIRF tests using rig test technique lies in the fact, that 100 V/m susceptibility standard presents the maximum possible limit (the worst case), which may occur during hardware operation. That is why, when the FS complies with this requirements, no full-scale HIRF tests onboard the aircraft are required. This concept forms the basis of the International Standard **CRI-SE-10** [4] concerning the certification of IL-96, TU-204 equipment under HIRF.

Taking into account real rig complex capabilities the specialists suggest the following sequence of hardware certification tests under HIRF:

- all "critical" systems are tested at NIIAO (prior to its installation onboard the aircraft) on the rig in accordance with the procedures specified in CRI-SE-10 for the frequencies 10 kHz to 1200 MHz;
- the systems installed onboard the aircraft are tested in Flight Research Institute (LII) using the first technique for the frequencies exceeding 1200 MHz;
- for the systems that had not passed the rig tests the aircraft bay HIRF shielding level is measured for the frequencies of the system maximum susceptibility;
- the system compliance is verified by recalculating the rig test results considering the fuselage shielding level.

Presently our industry conducts activities directed to the implementation of the HIRF equipment susceptibility certification methods. Particularly, NIIAO implements the "critical" system rig test technique. The institute commenced the rig test center development to test the standard compliance. The TEM-chamber for the frequency range 10 kHz - 200 MHz and horn-chamber for the frequency range of 600 MHz - 12 GHz have been produced; the signal sources to create the fields of the necessary strength are purchased and the technological documents are being developed.

Presently the RTCA commission is developing the document DO-160D, where the 20th Section is significantly changed concerning the avionics electromagnetic field susceptibility. The document unifies common requirements concerning avionics electromagnetic field susceptibility and "critical" systems HIRF susceptibility requirements. The draft of this document dated 16 March 1995 contains 17 categories concerning the avionics environmental conditions, one of the categories (R) is designated specially to standardize the "critical" systems susceptibility to HIRF fields during rig tests. The "critical" systems susceptibility under this category is evaluated in electromagnetic fields within the frequency range of 100 MHz to 8 GHz. The following requirements for electromagnetic field strength are specified:

- for unmodulated carrier and **20-28 V/m** AM-signals;
- for **150 V/m** pulse signals (within the frequency range of 400 MHz to 8 GHz).

Within the frequency range of 10 kHz to 400 MHz the systems are additionally tested for the susceptibility to the induction fields from the communication line wiring. The susceptibility standard is specified within the range of 0,6 to 30 ma of the noise injection current.

It is planned to revise the AP 8.4.1 of the Russian Airworthiness Requirements NLGS-3 in compliance with the RTCA DO-160D document.

Thus presently the problems of the integrated equipment standardization and qualification are the most important as far as the EMC is concerned, also the aircraft functional systems certification tests performance for the systems fulfilling "critical" functions under HIRF conditions are very important.

The existing standards, such as NLGS-3, DO-160C, DO-160D, CRI-SE-10 constitute the basis for the a/m problems solution. The most complicated problem here is additional tooling of the industry rig test center to test the compliance with the standards.

Reference

1. AP 8.1.4.1 Technical Requirements to the performance of the aircraft FS components specifying the EMC. Russian Airworthiness Requirement NLGS-3 Appendix.
2. RTCA DO-160B, DO-160C, DO-160D standards.
3. RTCA AE4R-91-XX Circular dated 15 July 1991.
4. CRI-SE-10 document.

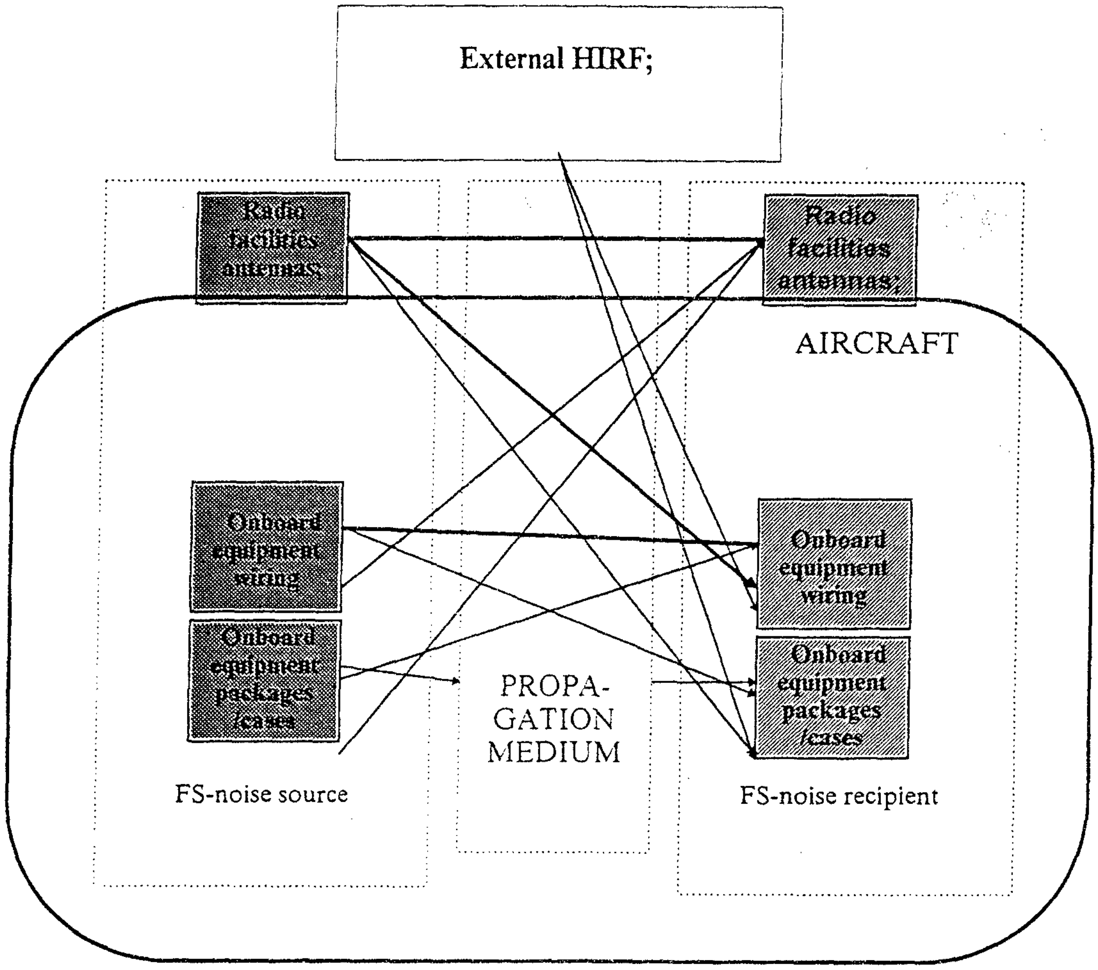

Fig. 1 The noises penetration paths from the source to the recipient.

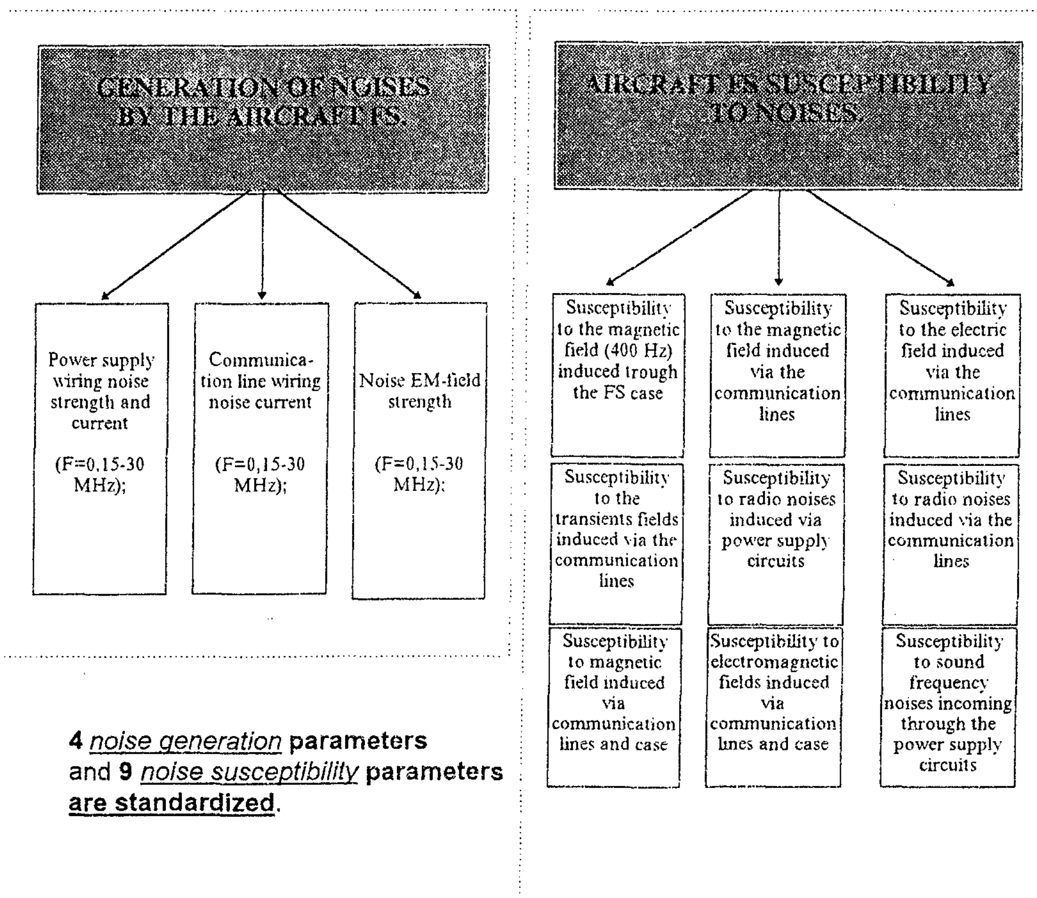

Fig. 2 Standardized aircraft FS EMC parameters (NLGS-3, DO-160B)

MECHANICAL, THERMAL AND CLIMATIC CERTIFICATION RIG TESTS OF AIRCRAFT EQUIPMENT PRIOR TO ITS INSTALLATION ON AIRCRAFT

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It is well known that the aircraft airworthiness is defined as its capability to perform a safe flight under all specified anticipated operating conditions provided that all other air transportation system components operate normally.

One of the main anticipated operating conditions component is characterized by the state values and environmental effects on the aircraft equipment, including barometric pressure, air temperature and humidity, different mechanical effects, above all vibration and shock, solar radiation, sand and dust, etc.

These effects can result in different equipment failures - from a temporary loss of operational capability to a full breakdown. So in accordance with the field data about half of the failures of the avionics occurs due to environmental effects of various physical nature. And out of them 90% of the failures are the results of thermal effects, vibration and humidity. That is why to provide the civil aircraft airworthiness the performance of certification environmental tests of avionics "before installation on aircraft" is specified.

In Russia Environmental categories and parameters, test procedures are specified in the current Airworthiness Requirements, Supplement 8.1.2. In the West RTCA/DO-160 is used as an international standard.

The current Airworthiness Requirements NLGS-3 Supplement 8.1.2 draft differs from the DO-160C standard in some aspects. This created certain difficulties in mutual acceptance of foreign and domestic avionics test results.

At present a new draft of the domestic environmental test standard is developed. It complies with the DO-160C standard as far as the test standards and procedures and environmental categories are concerned. The developed standard will allow the test results to be taken into account on a mutual basis.

The experimental rig test center of NIIAO provides the performance of 24 test types out of 26 avionics test types considering environmental climatic, thermal and mechanical effects, specified in Airworthiness Requirements Supplement 8.1.2, excluding linear acceleration, dust and sand effects. The experimental rig test center comprises 11 types of rig installations, including complex environmental rigs.

The set of 18 thermoaltitude chambers and dedicated climatic test chambers with the volume of 0,063 m^3 to 60 m^3 provides testing of avionics of different purposes with the length up to 7 m and diameter up to 2,5 m reproducing all possible climatic and thermal operating conditions specified by domestic and foreign equipment performance standards.

To provide all mechanical tests the NIIAO test center has an acoustic chamber with test box volume of 2,8 m^3 which allows to test avionics for acoustic noise effects with the sound pressure of 110...165 dB within the range of 50... 10000Hz, vibration test rig providing vibromotive force up to 5000 G, and 3 shock test rigs which allow to test avionics with the weight up to 500 kg for sinusoidal and wideband vibration effects within the frequency range up to 5000 Hz and for single and multiple shock-effects.

Complex environmental rig with useful chamber volume of 1 m^3 allows to test avionics with the weight up to 80 kg under simultaneous reproduction of necessary temperature, pressure and vibration load parameters within the chamber.

At present NIIAO test center is certified by Russian State Standard Commission and IAC Aviation Register for the required authority in accordance with the requirements of the GOST R Certification System and AT and OGA Systems in the field of environment tests of aircraft equipment.

It is necessary to notice that persistent improvements of the equipment performance standards as far as the avionics environmental tests are concerned and the requirements applied to the experimental rig test center during certification under GOST R Certification System and AT and OGA Systems Requirements imply high engineering level and systematic modernization of test facilities, measuring, test results processing and documenting systems.

However, as the deliveries of new test equipment from abroad and in Russia are significantly reduced, the existing facilities become obsolescent and outdated. The domestic manufactures of onboard equipment for aviation purposes use more than 50% of their test equipment for more than 15 years. The repairs and maintenance of such an equipment for it to be in a good operating condition gets more difficult with each year.

To provide the industry with the modern test facilities to test newly developed and produced products it is planned to develop using a building-block (modular) approach the integrated system (complex) of standard environmental test facilities using which it will be possible to develop test installations necessary to solve particular tasks with minimal costs.

The test complex is developed on the basis of designing separate compatible functional modules, that provide the reproduction and control of the environmental effects: increased or reduced temperatures, increased humidity, reduced atmospheric pressure changing slowly or rapidly (depressurization), solar radiation and vibration loads.

Standard test chambers of two types: thermal and altitude form the basic module of such complex. Each chamber type may have two form-factor: 1 m^3 and 4 m^3 , the units consisting of several similar form-factor chambers may be built, i.e. on the basis of 1 m^3 chamber module the units with the volume of 1, 2, 3,... m^3 may be built, and on the basis of 4 m^3 chamber - units with 8,12,16,... m^3 volumes.

Basic modules upon the customer's request may be completed with the functional modules providing the reproduction of the specified effect within the basic module. When a test installation (facility) consists of a unit comprising several modules, each module should be completed with functional modules. If the test facility, is to have a higher capability than one functional module can provide, two or more modules of the same type operating simultaneously can be used.

In some cases it is possible to incorporate the second functional module which operation is based on another principle.

For example: to achieve quicker cooling in the chamber an air cooling or nitrogen cooling installation may be connected additionally to a freon-type refrigerator.

The environmental test complex based on modular approach comprises:

- heat-producing module creating in the operating chamber basic module the temperature up to 125 ^{0}C with temperature values uniformly distributed within the chamber volume, temperature change rate constituting up to 30 ^{0}C per minute;
- freon refrigerator creating temperature up to -60 ^{0}C with cooling rate of 2 ^{0}C per minute;
- air cooling or nitrogen cooling installation providing the temperature reduction within the chamber up to -65 ^{0}C with cooling rate of 10... 15 ^{0}C per minute;
- humidity generator providing humidity level up to 100% under temperatures up to 60 ^{0}C;
- vacuum installation to provide the atmospheric pressure drop within the altitude chamber up to 1 mm Hg with the pressure falling rate of up to 6 mm Hg per second;
- solar radiation source, creating at the 500×300 mm workstation solar radiant emittance up to 1600 W/m^2.

Additionally the test chamber basic module can be equipped with the vibrobench located within the chamber volume.

All functional modules are produced as independent units and components to be interfaced with the basic chamber module. They have independent control and external interface output to be connected to the integrated control system. Test center may also be equipped with the independent data acquisition and processing system for the data incoming from the product under test. Besides functional modules can be used independently within the basic equipment used in production line.

DIGITAL AVIONICS PACKAGES FAILSAFETY ASSESSMENT

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The goal of the avionics packages failsafety assessment is the following:

- detection of all package's failure states and isolation of failures in special situations (such as a catastrophic, emergency, complex situation and complication of flight conditions);
- assessment of their probabilities with their check for airworthiness compliance;
- analysis of possible failure causes resulting in special situations with the goal of developing the ways of their removal.

The achievement of the above goal is made through generation of packages failure states with all types of functional and incompatible failures.

For valid and complete packages failsafety analysis and definition of special situation categories arising out of these failures it is necessary to define the following according to the MOS NLGS requirements:

- Functional failures:
  a) nature;
  b) possible causes;
  c) occurrence probabilities
- Incompatible failures:
  a) nature;
  b)possible causes;
  c) failure information for the crew;
  d) crew actions to correct failure;
  e) failure effect on flight completion;
  f) occurrence probabilities;
  g) special situation category.

Failsafety assessment and analysis are made in two stages. In the first stage the preliminary analysis is made the goal of which is to synthesize the descriptions of functional failure types and their models, then package failures are generated and analysed. Fig. 1 shows the block-diagram of a preliminary analysis. the essence of the preliminary analysis lies in the fact that at first it is made from top to bottom, i.e. the package specification and from bottom to top, i.e. the specification implementation through package hardware and software, and then types of functional failures and their models are synthesized through the generation of associative connections between the specification and its implementation.

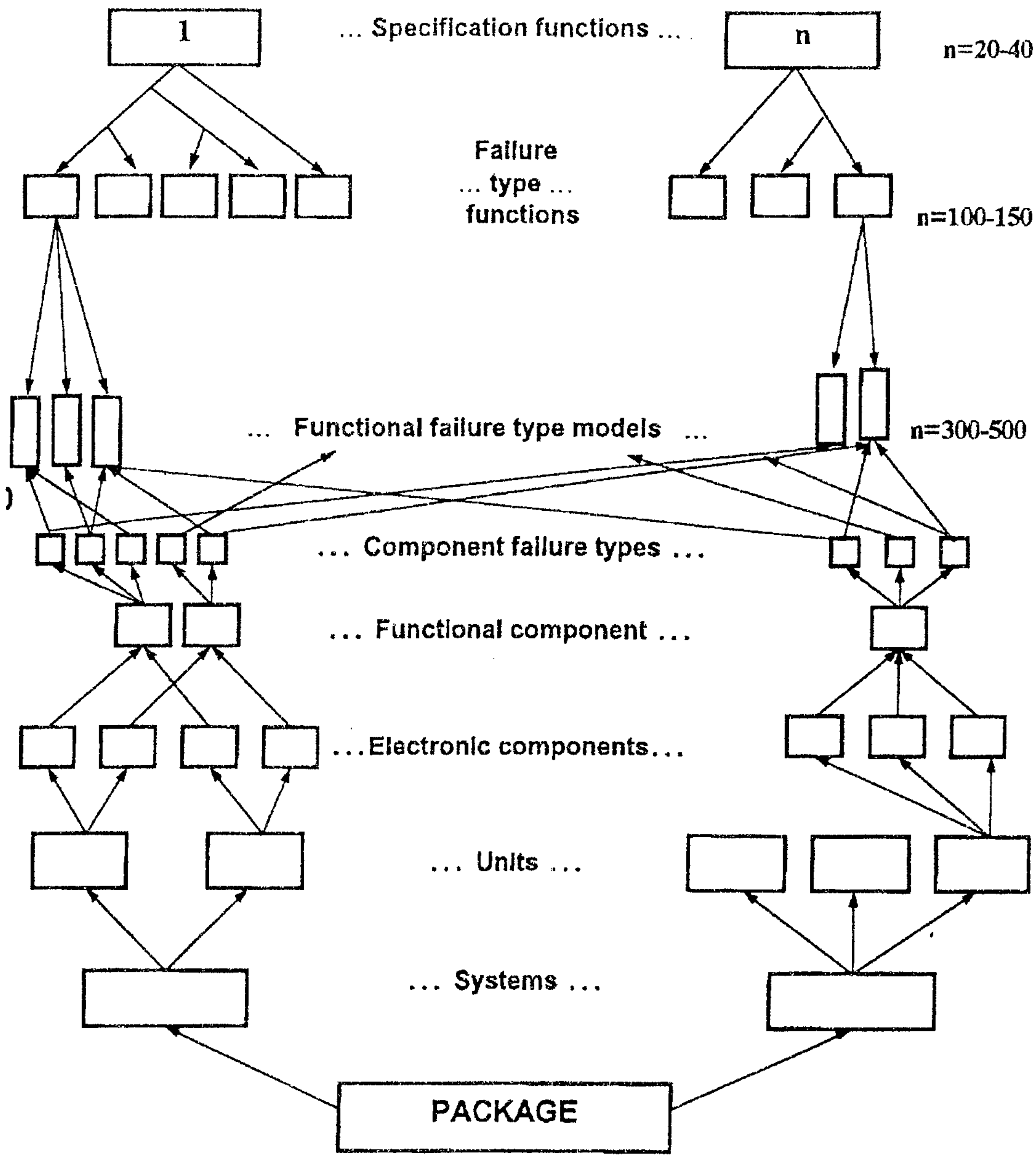

Fig. 1 Block-diagram of the preliminary analysis made to define avionics package failsafety

Let us consider these analysis:

a) Top-to-bottom analysis:

The structure of the functions selected in the package specification is generated that results in the lower structure level of the functions included in functional failure types. This part of the analysis is not formalized and is closely related to the task of generating the functions that the package must perform with the level of a detail down to a specific operation. This structure arises out of a correct selection of functions in the package specification. If there are no correctly structured functions in the package specification, this work must be done during the top-to-bottom failsafety analysis. A correct structure must be in the form of a list including package general functions each of which must be structured down to a lower structure level including the functions of specific operations that provide all details for a given general function.

The sets of such functions with the level of an operation detail and their failure types (functional failure types) characterize all package failure states.

b) Bottom-to-top analysis:

The functions implemented on package components are analysed. The failures of these components will result in real types of package functional failures. Consequently, in the bottom-to-top analysis all types of package component failures must be analysed and then types of functional component failures must be defined. According to MOS NLGS terminology this part of the analysis is called "The engineering analysis of package components".

The isolation of part of a system to a hardware failsafety analysis component (a functional component) is made according to the following principle:

Part of a system is isolated to a failsafety analysis component if it implements an individual function (part of a function). In this case:

- its failsafety index considerably effects the failsafety index of a complete system;
- if this part of a system is not isolated to an individual component, this results in a considerable inadequacy of the external manifestations of simulated functional failure types to their real manifestations.

The functions are defined for every functional component that are implemented on it. If this component is a processor, the function is implemented as a program. Thus every software component (module) must be related to the functions included in functional failure types.

Besides, since there is a real probability of a software error which also results in functional failure types, the software components must be isolated and the probabilities of their software errors must be defined in the bottom-to-top analysis.

c) Combined Analysis

In the combined analysis the connections are defined between the functions included in functional failure types, and the characteristics of functional components resulted from the bottom-to-top analysis. Such associators connecting the analysis results from top and from bottom are:

- connections between the functions included in functional failure types and the functional components;
- connections between the nature of package general failure types and functional component failure types;
- connections between function parameters and functional component parameters;
- connections between functional components and pilot channels;
- connections between the general nature of failure functions and the operations in functions;

• topology of information connections between package components (through the E3 and E4 IWDs);

• connections between the operations in functions and the direction and values of deviations in package general failure types;

• reconfiguration of the package information connection structure during component failures.

The functional failure modes are generated from the data of these associators. The models are descriptions of functional failure types and Boolean functions that are identifiers of functional component failure types and are interconnected by logical "AND", "OR", "NOT" operations.

This completes the preliminary package failsafety analysis.

The source data for the preliminary analysis are:

• Specifications for the package, subsystem and software design;
• Engineering analysis of subsystem reliability characteristics.

Let us consider the technological stages of the work done for the assessment of avionics package failsafety where along with preliminary analysis stages we will consider final analysis stages.

1. Arrays of hardware and software failures are generated from source data.

In practice there are around 100 such components in a package with their 300-400 failure types.

The structure of functions is generated around the package specification, its top level comprises general functions and its bottom level comprises detailed functions being specific operations performed with a combination of hardware and software components. Such functions may include for example, functions of control panel operations (mode turn on and off, RMU control, etc.), functions of outputting control signals to controls, display functions, etc.

2. For every lower level function the components are defined on which it is implemented. The main avionics package components are computers to which programs are loaded that implement avionics functions. Besides computers, the avionics functional components are flight data sensors, flight data displays and flight controls.

Thus, the avionics package is a multiprocessor network with a ramified terminal system.

Since main avionics functions are programmed, the functional failures may occur both due to hardware failures resulting in incorrect program implementation and to software errors. The probability of a software error depending on a function criticality may be selected and confirmed trough program testing.

The types of package functional failures are generated depending on hardware and software failure types.

3. The models of the package functional failure types including all healthy and failure package states relative to a given function are generated automatically based on the following information:

• structure of information connections between the components on which the given function is implemented;

• rules to reconfigure the information connection structure during failures of part of components.

Thus the completion of the above technological procedure is the end of the preliminary analysis, and the technology is ready for the final analysis, i.e. generation of all package failure states.

Let us consider the technology of package failure generation and analysis.

4. All possible package states are searched automatically, that include various combinations of one, two and three hardware components failure types.

If the number of hardware failure types equals 400, the number of such states will equal:

$$N=\sum_{i=1}^{3} C_{400}^{i}=\frac{400*399}{2}+\frac{400*399*398}{6}\cong 10^{7} \qquad (1)$$

5. Models of functional failure types are analysed automatically for every hardware failure state to define healthy and failure package states. For example, the analysis of the TsPNK-114 package failsafety in cruise flight showed 2,700,000 hardware failure states and 119 package failure states (incompatible failure types) whose probability was greater, than E-09. Thus, the percentage of the package failure states relative to the total number of component failure states was equal to 0.001 which is the index of high TsPNK-114 failsafety.

6. All types of functional and incompatible failures with all characteristics given above are generated automatically for all package failure states. In this case the technology provides for the automatic compilation of the texts describing the nature of individual failure types during their compatible occurrence for a specified combination of component failures as well as other texts pertinent to individual component failures.

7. The calculation of every incompatible state is done according to the formula:

$$P_{njj}=\frac{\prod_{i=1}^{n} e^{-\lambda_i t} * \prod_{i=1}^{no} (1-e^{-\lambda_{oi} t})}{\prod_{i=1}^{no} e^{-\lambda_{oi} t}} \qquad (2)$$

where:

n - is the number of unique component failure types in the models of compatibly occurring functional failure types (at the given j-th incompatible state),

no - is the number of component failure types occurred in the j-th state,

λi - is the intensity of the i-th component failures out of the set n;

λoi - is the intensity of the i-th component failure out of the set n0,

t - is the period of probability computation.

The probability of incompatible failure for at least one cause is calculated according to the formula:

$$p_N=\sum_{j=1}^{k} p_{nj} \qquad (3)$$

where:

k - is the number of states with similar descriptions of failure types,

This technological sequence was automated as the computer AFOBEPOL technology the efficiency of which was confirmed by the failsafety analysis during the certification of the IL-96, TU-204, IL-114 avionics packages.

Let us consider the mathematical description of the avionics package failsafety assessment and analysis.

a) The sets describing all avionics package states (healthy and failure) form the Boolean algebra U.

b) The sets of all algebra U elements At are indexed, i.e. {At}t is understood as a mapping that assigns the corresponding element At to every t being an identifier (The T-set of all identifiers).

c) The algebra U consists of two subalgebras:

- the subalgebra U1 that includes all sets pertinent to an avionics package and its subsystems selected in the specifications for an avionics package and its subsystems;

- the subalgebra U2 that includes all sets implemented on functional (hardware and software) components of an avionics package and its subsystems.

d) If a package is properly designed, isomorphism must exist between the subalgebras U1 and U2.

Let us consider algebra U1 sets:

1. The set Afs of all avionics package and its subsystems functions.

The set Afs consists of the following subsets:

1.1 The set Aftz of functions selected in the specification for an avionics package and its subsystems;

1.2 The set Afsn of functions, the nonfulfilment of which characterizes concrete failure states of an avionics package and its subsystems.

1.3 The set Aftzi of functions being part of the i-th level of the hierarchical structure showing connections between the sets Afstz and Afsn.

To compress data base information and create source data arrays for avionics package classes the method of generating zero level classifiers was applied. The essence of the method lied in learning information structure for each of the above sets and generating for every element the structure of a word combination subset allowing to compile any set element within the selected avionics package class.

The zero level subset structure for the description of functions Afs included the following subsets:

a) Operations in function descriptions - A0;

b) Modes and other characteristics in function descriptions - Ar;

c) Parameters in function descriptions - Av.

2. The set Avfo including the descriptions of avionics package and its subsystems functional failure types that is synthesized from the sets Afsn and Avos, describing avionics package failure types.

The structure of zero level subsets for the description of the set Avos (avionics package and its subsystems failure types) included the following subsets:

a) Nature of failure types - Avoch;
b) Failure in a pilot channel - Avopi;
c) directions and relative values of deviations - Avoot.

3. The set of functional failure type models - Amvfo.
Let us consider the sets of the algebra U2.

1. The set As of avionics package subsystems.
2. The set Ae of subsystem components.
3. The set Aems including the elements of the matrix of information connections between package functional components.
4. The set Afe of functions (programs) implemented on the package components.
5. The set Ato of maintenance types and periods of recovery.
6. The set Fpe of parameters processed on package components.
7. The set Avoche of component failure type nature.
8. The set Ank of test types.
9. The set Avee Akos of special situation categories.

As was shown above, as a result of the top-to-bottom analysis the set Afsn is generated, consisting of the functions the nonfulfilment of which characterizes specific failure states of an avionics package and its subsystems. The result of the bottom-to-top analysis is the generation of all algebra U2 sets.

Let us consider the mathematical description of the combined analysis resulting in the descriptions of all types of functional failures (the set A_{vfos}) and their models (the set A_{mvfos}). As was shown above, the combined analysis defines the connections between the subsets resulting from the top-to-bottom and bottom-to-top analyses.

Let us describe these connections mathematically.

a) The connection between the functions of package failure types and the components on which it was implemented. This connection has the form of the sets A_{fsn} and A_e, intersection, i.e.

$$A_{efs} = A_e \cap A_{fsn} \qquad (4)$$

Since it is better to do the package design work, when competent specialists work at units responsible for scientific and technical support of the subsystems with their specified components, the formula (4) maps the information about the functions implemented on an individual component. However, the inverse function is necessary for the generation of a functional failure type model, i.e. the associator of the connection between the set A_{fsn} of functions and the set of components on which it is implemented, i.e.

$$A_{fse} = A_{efs}^{-1} \qquad (5)$$

b) The connection between the nature of package general failure types and the nature of component failure types. The associator of this connection will have the form:

$$A_{chvose} = A_{chvos} \cap A_{chvoe} \qquad (6)$$

c) The connection between package general failure type nature and operations in the set A_{fsn} of package functions. The associator of this connection will have the form:

$$A_{chvosofsn} = A_{chvos} \cap A_{ofsn} \quad (7)$$

d) The connection between directions and relative values of deviations in the nature of package general failure types and operations in the set A_{fsn} of package functions. The associator of this connection will have the form:

$$A_{nvosofsn} = A_{nvos} \cap A_{ofsn} \qquad (8)$$

e) The connection between the set A_e of components and the set A_k of pilot channels. The associator for the pilot's channel components will have the form:

$$A_{ek} = A_e \cap A_k \qquad (9)$$

f) The connection between the components and the matrix of component information connections. The associator of this connection will have the form:

$$A_{mse} = A_e \cap A_{ms} \qquad (10)$$

The Boolean formulas mapping the processes of generating the sets Avfo of package functional failure descriptions and their models Amvfo will have the following form:

$$A_{vfo} = A_{fse} \cap \quad A_{chvose} \cap A_{chvosofsn} \cap A_{nvosofsn} \cap A_{ek} \qquad (11)$$

$$A_{mvfo} = A_{fse} \cap A_{chvose} \cap A_{chvosofsh} \cap A_{ek} \cap A_{mse} \qquad (12)$$

Formally the formulas (11) and (12) may be expressed in the elements of their corresponding associators and the expression may be received for the description of functional failure types and their models which would mean the elimination of the combined analysis. However, we must consider in formalizing the combined analysis process, that competent specialists for specified package subsystems and components can generate validly and accurately only paired associators given in the expressions (4) - (10).

The AFOBEPOL technology using the knowledge base of competent experts who provide scientific and technical support of individual subsystems and components in the form of the associators (4) - (10) for every individual component, automatically combines these associators and implementing the algorithm of the expressions (11) and (12), generates arrays describing functional failure types and their components.

The algorithms to simulate functional and incompatible failure types and to automatically generate the reports on the package failsafety analysis results that are implemented in the AFOBEPOL technology are rather complicated and require considerable routine operations for their descriptions. For this reason they are not presented in this report.

The sequence of the AFOBEPOL basic operations was shown on a PC during the report.

Lightning protection of aircraft on-board equipment

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- It is not necessary to speak about the importance of the flight safety problem, especially for modern aircraft carrying a large number of passengers. Flight safety and lack of catastrophic situations during passenger transportation are the main goal of all world airlines.

At the same time, airlines trend towards all-weather air transportation complicates the above goal. As is known, all complications of weather conditions are mainly related to the increase in cloudiness followed by a thunderstorm situation and presence of atmospheric electric discharges.

So along with many other problems influencing air transportation safety, one of the important problems is healthy operation of on-board equipment in pulse electromagnetic interference fields caused by lightning discharges (or on-board equipment lightning protection).

- World practice possesses considerably complete investigation of the matters related to the nature of lightning generation and phenomenon and its electrophysical characteristics. The processes related to the physics of lightning impact on an aircraft and its equipment are also investigated.

- Based on this knowledge, standards have been developed regulating impact rates that must be considered during the design and manufacture of both on-board equipment and complete aircraft structure. The certificate for an aircraft and consequently its equipment prior to the beginning of an aircraft operation by airlines must be received as the guarantee of flight safety. This certificate can be received only on the basis of corresponding tests including special rig simulation installations. The installation parameters and methods of lightning protection tests are regulated in the corresponding paragraphs of DO - 160C, (DO - 160D), NLGS and OST 101160 - 88. These documents also regulate the rates of impacting pulse voltages caused by lightning discharges, in the form of specific rigidity categories depending on the electromagnetic situation on board an air vehicle where on-board equipment is placed. Based on the statistical processing of the investigation results for pulse voltages arising from equipment inputs upon lightning impact on an aircraft, these pulse voltages were rated relative to their form as a "short wave" a "long wave" and an "oscillationg wave" with their specified parameters.

The rated voltage levels and rigidity categories are given in Table 1. Figures 1, 2 and 3 give the pulse and voltage forms of a "short, long and oscillatory wave".

- A "long wave" is the voltage potential difference, that may arise between equipment grounding points when lightning current flows across an airframe.

A "short wave" is the voltage in an open circuit induced by a magnetic field that has a "long wave" form and is generated by the lightning current flowing across an aircraft.

An "oscillatory wave" is a voltage spike excited by electric resonances induced in aircraft wires by the pulse of lightning current flowing across an aircraft.

The tests of the on-board equipment for all three types of a rated pulse voltage form provided positive test results allow to state with a considerably high degree of confidence that the equipment will be operational for all probable cases of lightning impact on an aircraft.

- The certification of on-board equipment for lightning protection is carried out mainly in two stages:

a) equipment certification through its direct rig tests, and;

b) equipment certification as part of an aircraft through the flow of the pulse current simulating the lightning current across an airframe.

The rig on-board equipment tests are mainly restricted to the simulation of the inductions in the on-board electric lines expected from the lightning current flowing across an airframe and from the pulse voltage and current occurring at the elements of input and output devices of interconnected on-board equipment units.

Depending on the amplitude of an impacting interference signal and the tolerance limit level of a corresponding circuit element, the input (output) device may turn out resistant to overvoltage caused by the lightning current or circuit or functional failure of equipment operation may occur. A special program is required to conduct such tests which determines the circuit points to record and measure necessary parameters, gives assessment of the efficiency of the means used for the protection against pulse overvoltage, selects test and measurement equipment, gives assessment of measurement errors, etc. (Fig. 1 and 2).

Fig. 1:

1 and 3 - elements (units) of the on-board equipment under test;
2 - interconnect harness and (or) interconnect cables;

"Long wave" test installation block-diagram

4 and 7 - insulation stand;
5 - metal plate;
6 - electronic oscilloscope;

Device for selected pulse generation

AT - automatic transformer;
T- transformer;
R - rectifier;
V - voltmeter;
R - limiting resistor;
R2 and R3 - shaping resistors;
C - capacitor;
L - inductance coil;
S - pulse turn-on switch.

- We think it expedient prior to tests to make an estimated assessment of the expected induced pulse voltage and current values at various points of on-board input and output device circuits. The estimated assessment results will allow to optimize the test program, to define the bottlenecks of input devices and equipment units, to provide for the possible protection of the circuit input elements from pulse overvoltages. This methodology of the estimated assessment has been developed at our institute, tried in several aircraft certification tests (the Il-96-300, Tu-204, Be-200) and has given positive results (test accuracy, test volume, protection means efficiency). The methodology is efficient also due to the fact that it is "tied" directly to the rig equipment test environment. The essence of the methodology lies in the following:

The main source data necessary to make the estimated assessment are:

- the rigidity category for compliance to conduct the on-board equipment lightning protection rig tests;
- the schematic circuit diagrams of equipment units, output and input devices interconnected via an electric line (the equivalent resistance values of the input and output line loads are determined);
- the specification and rated values of the elements being part of input and output devices;
- the tolerance limits of operating electric rates for every element being part of input and output devices;
- the length of the on-board line interconnecting equipment units;
- the interconnected line type:
    a single wire (shielded or unshielded),
    two shielded wires with differential input;
- the line conductor section and the conductor material resistivity;
- the line shielding braiding diameter and the braiding material resistivity;
- the thickness and resistivity of the rig metal plate material;
- the input and output device circuits are considered (Drawing P1);
- the equivalent resistances of input and output devices are determined (Drawing P2);
- the resistances and inductance of interconnect lines are calculated;
- the equivalent circuit of the specific chains is generated considering the interconnect line type (a single wire, two shielded wires, the interconnect line with differential input). This circuit includes as an electromotive force a pulse voltage generator with the specific (selected) wave form (Drawing P3);
- the transient process is calculated that arises in the circuit, and on its basis the pulse voltage values are determined that arise from the equivalent input and output device resistors;
- the calculated pulse voltage values are compared against their permissible values established for circuit elements and the possibility of failure in the input and output devices and the complete units is assessed.

The validity of the estimated assessment results turned out quite satisfactory. This assessment allows to find out "weak", less protected input and output device elements and at the same time to provide for specific methods and means of their protection from the pulse voltage impact.

The following elements may now be used for the protection of various types of aircraft equipment from pulse overvoltages caused by a lightning discharge impact: semiconductor limiting diodes, variable-ratio transformers, semiconductor stabilizer diodes, pulse diodes and Schottky diodes, lightning arresters, etc.

The semiconductor protection elements due to their specific voltage and current characteristics practically don't consume any power when there is no overvoltage pulse since their resistances are rather high (0.1-100 MOhms) and, consequently, practically don't dissipate any extra power, i.e. don't effect the operating mode of the protected equipment.

Upon the occurrence of pulse overvoltage with the amplitude above the protection element threshold the resistance of these elements decreases abruptly to the value of 0.01-1.0 Ohms. In this case the current flows mainly across the protection element bypassing the load and reaches the values of hundreds of amperes in some cases. As a result the pulse voltage of the protected chain is limited to a selected level, and the greater portion of the voltage drop occurs within a ballast resistance. The protection elements having high pulse dissipated power are of small dimensions and weight, which is rather important to aircraft equipment.

Besides, they have rather high speed, which is important to circuit protection at high overvoltage wavefront steepness.

The calculations are a usually made on Pcs (the simplified calculation is possible on a microcalculator).

The estimated assessment is also expedient due to the fact that it allows to get the necessary data on the circuit resistance in the early design stage, to provide for the application of resistant elements in input and output devices, to apply the specific means of input circuit protection.

The following basic provisions must be observed for the general on-board equipment lightning protection:

- the aircraft fuselage as an electromagnetic shield must have as few slots and holes as possible;
- the on-board equipment is desirable to be located in the most shielded aircraft compartments;
- the unit interconnect cable lines must have shielding braiding;
- both on-board units and their interconnect lines must have reliable grounding to an airframe;
- the input and output on-board devices must have protection from pulse overvoltages that inevitably arise from induction in the interconnect wiring due to the effect of electromagnetic fields, caused by lightning currents that flow across the airframe.

The efficiency assessment of the measures taken to provide for lightning protection, is made in three stages:

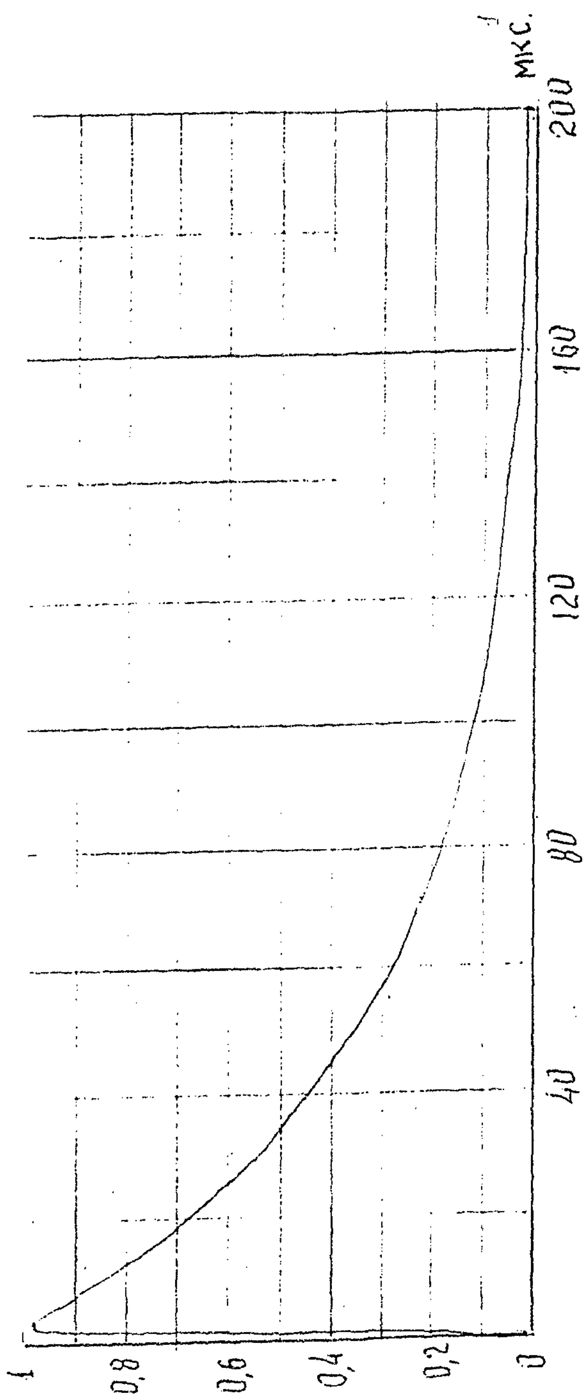
Stage 1 - the estimated assessment;

Stage 2 - rig equipment tests;

Stage 3 - the tests of equipment being part of an aircraft.

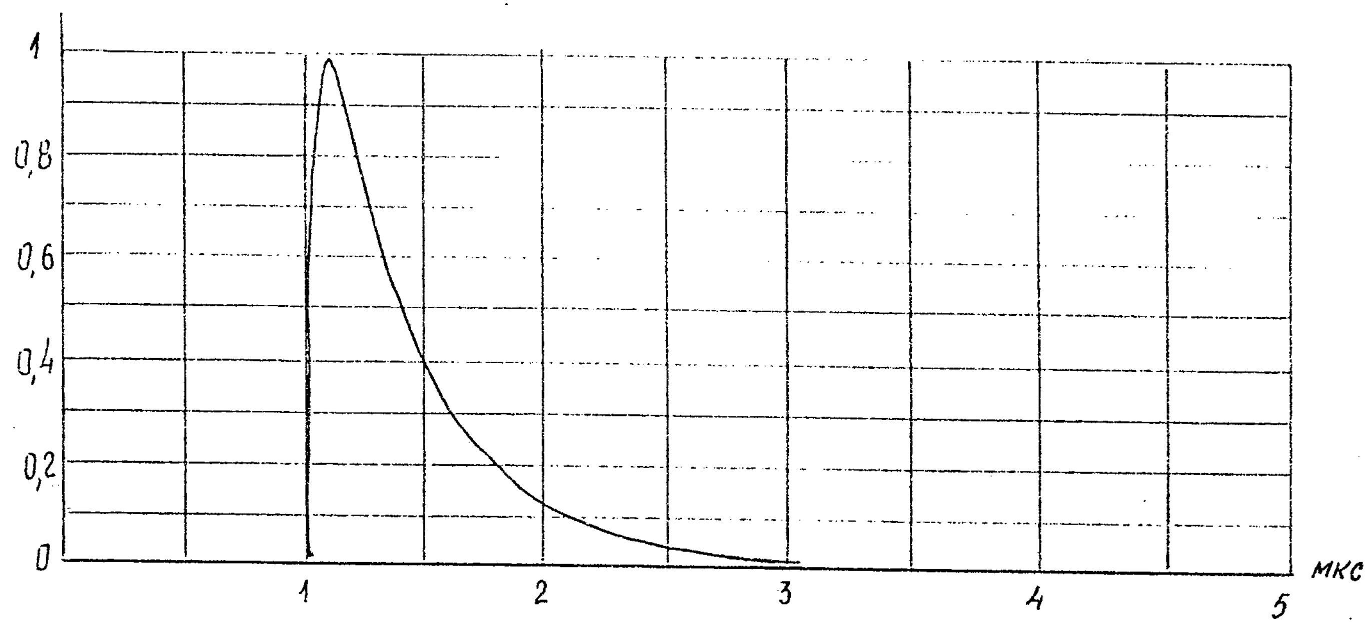
Table 1

NO	Rigidity category	Short wave (v)	Long wave (v)	Oscillatory wave (v)
1	I. Good shielding (J)	125	125	250
2	II. Partially damaged shielding (K)	300	300	600
3	III. Poor shielding (L)	750	750	1500
4	IV. No shielding (M)	1600	1600	3200



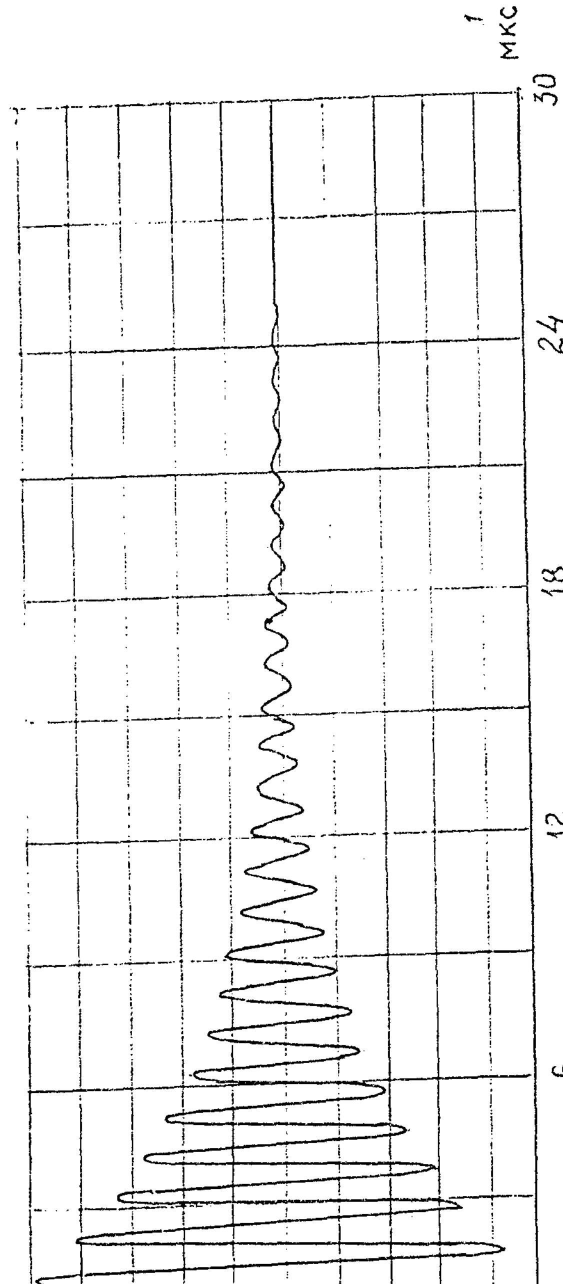
Drawing 1. The rated lightning voltage pulse of the 'long wave' type:

1 - µs



Drawing 2. The rated lightning voltage pulse of the 'short wave' type:

1 - µs



Drawing 3. The rated lightning voltage pulse of the ' oscillatory wave' type:

1 - µs.

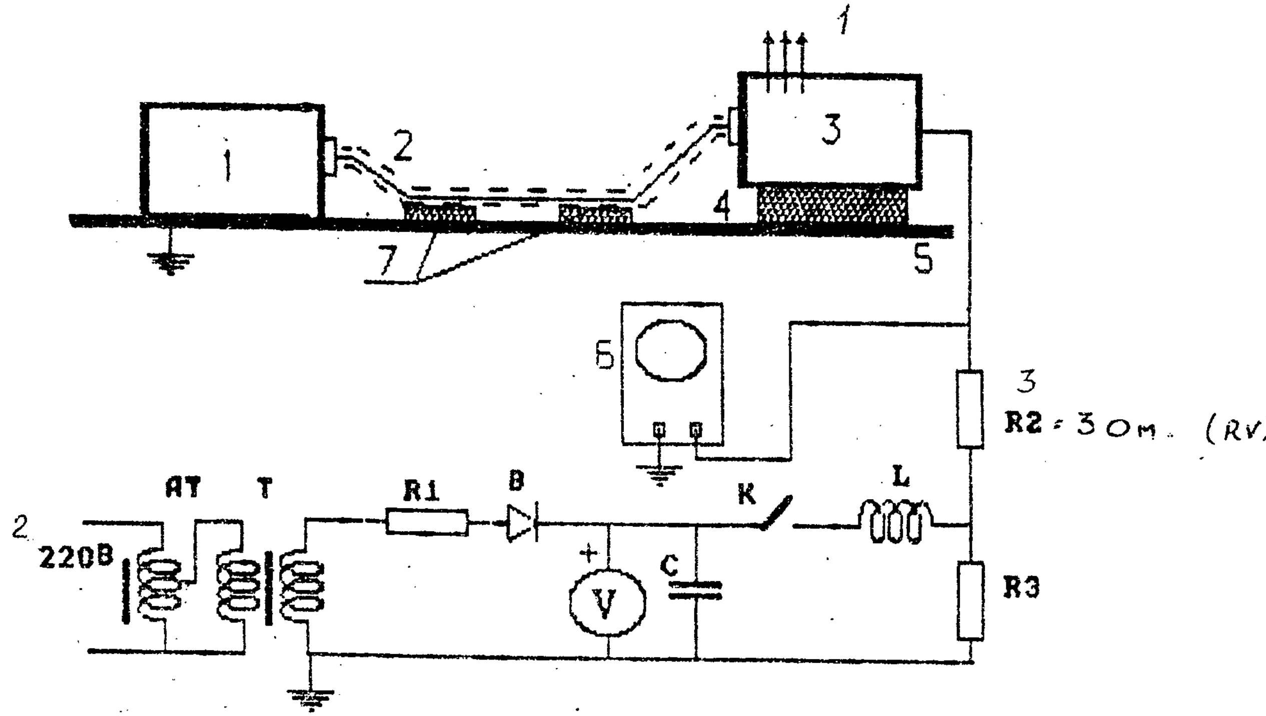


Fig.1. The block-diagram of the rig installation to test on-board equipment for lightning protection from a 'long wave' impact:

1 - to test and recording equipment;
2 - 220 V;
3 - R2=3 Ohms.

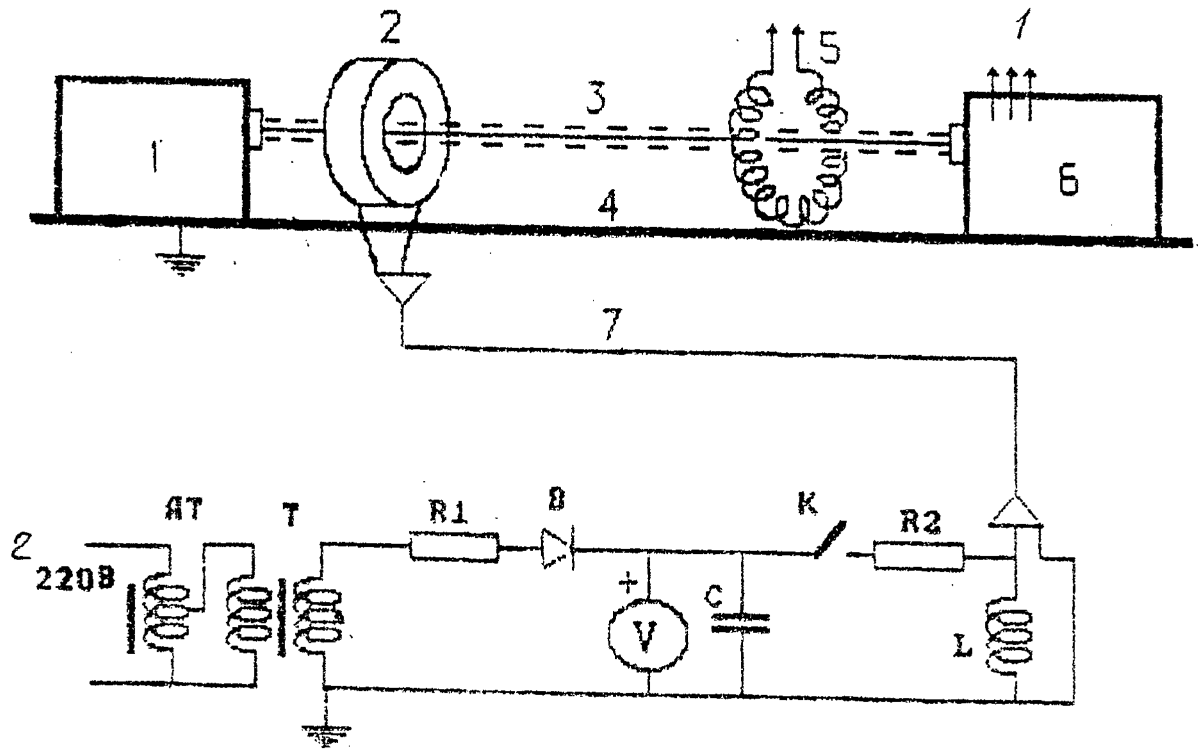
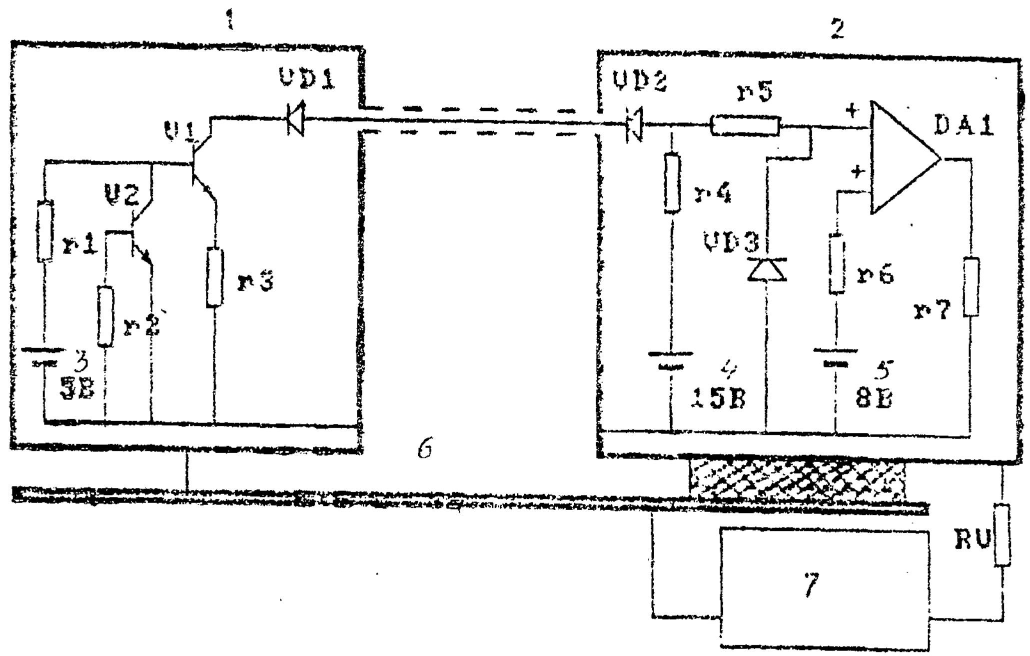
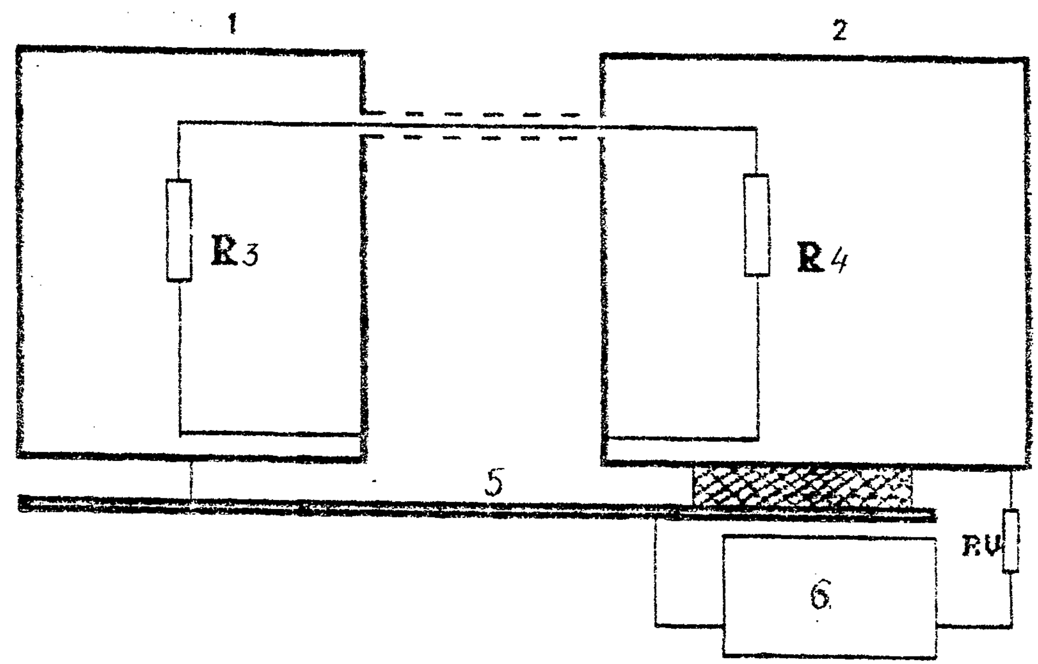


Fig. 2. The block-diagram of the rig installation to test on-board equipment for lightning protection from a 'short wave' and 'oscillatory wave' impact:
1 - to test and recording equipment;
2 - 220 V.



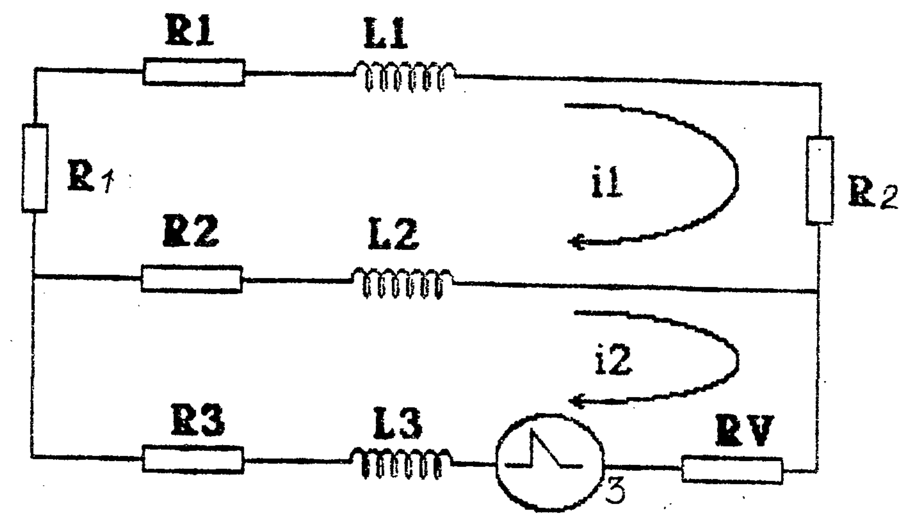
Drawing P1: The schematic circuit diagram of the equipment located on rig installations:

1 - unit 1;
2 - unit 2;
3 - 5 V;
4 - 15 V;
5 - 8 V;
6 - metal plate;
7 - pulse voltage generator.



Drawing P2: The equivalent diagram of the equipment located on rig installations:

1 - unit 1;
2 - unit 2;
3 - R_{output};
4 - R_{input};
5 - metal plate;
6 - pulse voltage generator.



Drawing P3: The equivalent circuit to make calculations for a single-wire shielded interconnect line:

1 - R_{output};
2 - R_{input};
3 - pulse voltage generator.

R1 - conductor resistance
L1 - conductor inductance
R2 - shielding braiding resistance
L2 - shielding braiding inductance
R3 - installation plate resistance
L3 - installation plate inductance
RV - pulse generator internal resistance

FEATURES OF AIRCRAFT CERTIFICATION RELATED TO VERTICAL SEPARATION IN RUSSIA AIRSPACE

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In the former USSR till the 60-s the system of flight levels had been valid with higher vertical separations than that in other highly developed countries. These separations were at an altitude of up to 6,000m and 1,000m at high altitudes.

In the sixties to eighties a number of activities were carried out to retrofit altimeter control equipment, and statistical investigations were conducted for the performance of measuring and altitude-hold equipment which enabled vertical separations to be reduced to 300m at an altitude of up to 8,100m and to 500m at an altitude of up to 12,100m. It is worthwhile to give attention to the fact that the separations are more narrow than specified in ICAO rules within the altitude range from 9,100 to 12,100m.

These activities were performed under the conditions which essentially differ from that for the work in other countries. On the one hand, the centralized economy allowed the work to be managed using plans which were mandatory for pertinent government departments. On the other hand, the deficiency of financial investments did not enable some technical advances to be applied. It is primarily a matter of constant-altitude control in scheduled flights using special radars which provide measuring absolute altitude with sufficient accuracy.

These conditions resulted in developing the methods of estimating a level of flight safety other than the methods accepted by ICAO.

The present paper deals with the description of this methodical approach and the methods of aircraft certification based on this approach for vertical separation.

With its basic points, the approach agrees with that generally accepted. The difference from the approach accepted by ICAO lies in the methods of obtaining the error distribution laws for altitude measurement and hold. With no available statistical data on total errors collected with a radar in the process of aircraft operations, this approach is based on considering error components for:

- static-pressure pickups;
- altimeters;
- altitude-hold equipment

Since altitude measurement control is provided in flight by comparing the readings of the first and second pilots' indicators, an error of a static pressure pickup is considered to consist of two components:

- a component common for all static pressure systems on aircraft;
- a component individual for each system.

The first component results from fuselage deformations at longer distances than the distances between static orifices, and also from the errors in determining pressure errors and the inadequate consideration of static pressure error corrections in the ADS computer. This component can not be reduced by using redundancy and in-flight instrument monitoring. The second component results from surface deformations close to static orifices and can be detected and removed by the c ·w in flight, if it reaches a high value.

In the mathematical description the density function for each compone.ıt is specified as a symmetric exponential distribution in the interval plus/minus the triple root-mean-square error. The size of statistical data required to prove the selection of the error distribution law is in excess of the available material obtained for field-proven static-pressure pickups, therefore the distribution for the central part of the density function is selected from the following considerations:

- the selected law should have the slower density damping for large errors than the normal one in order to avoid the possible underestimate of a hazard;
- the symmetric exponential law satisfies the first condition and is often used by ICAO in estimating safety of lateral separation.

Even less statistical data was available for selecting the error distribution for errors which exceed the triple root-mean-square deviation. A rather pessimistic version was selected where the probability of large errors comparable with the separation would not be underestimated, i.e. the uniform distribution to errors equal to the separation.

The approach was similar as applied to an altimeter. The distribution law is considered normal within the range of admissible errors. The selection of normal distribution was based on analysing the results of altimeter tests in the operation. The uniform distribution is taken over the range from the maximum admissible error to the separation. The square of end areas is equal to the probability of an inflight altimeter failure.

The total errors for altitude measurement are determined as the composition of two components:

$$f_{meas} = f_{meas\ sa} \times f_{meas\ ma}$$

The first component equals the error component described above for a static-pressure pickup as common for all static pressure systems on aircraft.

The second component is generated with special mathematical methods based upon an independent error component for a static-pressure pickup and an altimeter with due account of the procedure of inflight altitude measurement control. The analytical expressions of resulting error density are obtained for the simplest versions of a control procedure and numerical methods are used for more complex procedures.

The error density function for altitude stabilization is similar in the mathematical representation to that for a static-pressure pickup and different from it since it takes into consideration the inflight control of stabilization errors provided by the altitude warning unit. The calculations utilize a relation which permits distribution tails to be considered with allowance for the adjustment threshold of a warning unit and its errors.

The resulting total error of altitude measurement and hold is obtained as a composition of the above components. It should be noted that the model is given in the paper in a simplified form so that technical details should not obscure the principal idea.

Standard requirements which provide flight safety at separations of 300m are developed with the described model for an aircraft and aircraft equipment, taking into account the frequency of aircraft rendezvous in airways, and possible lateral navigation errors. These requirements corresponding to the overlap probability of 3×10^{-2} are given in the Table below compared with the similar requirements of ICAO regulatory documents related to reduced vertical separations. The requirements are generally in agreement. Some differences in the requirements structure result from the differences in the approach described above. The higher level of requirements to equipment redundancy and reliability specified in Russian standards attracts the attention. This is due to two reasons:

- not to specify excessive requirements to altitude measurement accuracy for older aircraft;
- to reduce the load of an air traffic control service (an aircraft requires particular attention of an air traffic controller only after the second failure of altimeters).

The most expensive and labour-consuming procedure in the aircraft certification process is estimating the performance of static-pressure pickups. This estimation is performed in flight in a pair with a so called pacer aircraft, i.e. an aircraft where static pressure error corrections are determined with high accuracy with several independent techniques (to eliminate measurement-method errors). At least three aircraft of a given type are tested with a pacer aircraft. The altimeter performance are estimated based upon ground tests conducted in the process of flight tests of an aircraft. Results of operational checks of similar altimeters on operating aircraft are used, if required. The total error of altitude measurement is checked in a single flight as a minimum (after the algorithm of calculating static pressure error corrections has been implemented in an altimeter).

The performance of altitude stabilization in the automatic and wheel control modes are estimated in extended en-route flights. The duration of a test in the wheel control mode is no longer than the duration of such a mode approved in the flight manual so that the possible effect of crew fatigue can be assessed.

If all the performance determined during tests comply with that given in the Table, an aircraft is considered to satisfy the requirements. If a part of performance exceeds the required performance and another part is lower, it is allowed to substantiate the non-excess of a specified probability of vertical overlap based upon the calculations using the above model.

Russian and ICAO Requirements for Flight Operations
with 300m Vertical Separation

Russian Requirements	ICAO Requirements
1. Resident Error of a Static-Pressure Pickup for an Aircraft Type	
25m	25m
2. Random Errors of a Static-pressure Pickup	
w/o altimeter errors considered a) rms value for one system: 13m b) rms value for an aircraft: 13m with altimeter errors considered maximum value of the total error for an example: 75m	with altimeter errors considered a)residual error plus triple rms value for an aircraft type: 60m (75m in operation) b) maximum value for an example: 60m
3. Altimeter Errors	
maximum: 40m rms : 18m	included in the requirements per the previous item
4. Altitude Hold	
a) rms value: 15m, including operation	a) rms value in operation: 13m b) maximum: 15m for young aircraft 40m for old aircraft (before 1990)
5. Altitude Deviation Warning	
threshold: 60m tolerance: ±20m	threshold: 60m tolerance will be negligible
6. Total Errors of Altitude Measurement and Hold in Operation	
not specified	25m - 0,016z^2 (rms), where z = mean value of the specified error for an aircraft type in meter
Equipment Reliability	
failure probability for a flight hour static pressure system: $5*10^{-5}$ pressure altimeter : $5*10^{-4}$ altitude warning unit : $1*10^{-3}$	only the probability of undetected failures in altitude measurement for a flight hour is assumed to be specified: 10^{-5}
Number of Independent Altitude Measurement Channels	
three	two
Automatic Altitude Measurement Control	
provided	not provided

Alternate Current Generation System Certification

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Aircraft Power System Department of NIIAO is capable to perform Power System certification tests, including the test of power generation systems within the Power Systems (PC).

Certification test are conducted at the stage of the factory and state (interdepartment) laboratory tests. The full-scale generation system mock-up (usually within the Power System) is tested for its compliance with the Aircraft Airworthiness Requirements, Aviation Rules, Requirements Specifications and other customer-defined equipment performance standards.

The main requirements to a.c. power generation system and PS component mock-up providing the system operation and specific power receivers power supply comprise the following:

1. The mock-up channel number, generator actuator types, generators, adjustment, protection, monitoring and control hardware of the power generation and distribution system shall comply with the need of the particular aircraft or helicopter. In some cases it is allowed to conduct PS mock-up tests under insufficient main source channels. The controls and display systems of power generation channels states indication (in case the display system doesn't constitute part of the standard utility system) shall be similar to those the aircraft (helicopter).

The prototype of each unit and hardware type used in PS and employed in the mock-up shall undergo tests for compliance with the requirements of the standards, regulations and requirements specifications applied to it.

2. Some generation channel power supply circuits (cables) shall be made similar to aircraft harness as far as the length and wire standard are concerned. This applies, for example, to the voltage controller and driver field excitation coil, voltage controller measuring circuits and channel hardware protection circuits up to the point of control in the power generation system mock-up.

A.c. preliminary distribution network shall be equipped with the protection, switching and stand-by hardware which parameters comply with the aircraft/helicopter network parameters. The same is applied to the computer and high dynamic load power supply lines. It is necessary to achieve the minimal possible difference between mock-up and aircraft source feeder reactance and resistance by increasing the mock-up feeder wires for each phase and zero-current wire.

3. System mock-up is provided with the load-producing devices which are capable to form total loads similar to those of the aircraft, that depend on the primary a.c. source in compliance with the aircraft/helicopter operating load schedule for different stages of flight under normal or partial PS operation. The a/m load devices may be connected directly to the a.c. main distribution bus of each channel. Power Generation System mock-up loads are usually considered to be identical when the following conditions are met:

- power and power factor values selected during the tests and measurements of the identical static load values are registered with the error of not more than 5%;

• non-linear static three-phase and one-phase loads are simulated by full-wave three-phase and one-phase rectifier loads accordingly;

• actuating currents and time-to-reach nominal speed values of the dynamic load simulators (asynchronous aircraft engines) differ for not more than 20% from the corresponding values of the actual loads;

• pulse generation frequency and pulse duration/pulse generation period ratio of the simulator differ for not more than 10% from the corresponding values of the actual periodic pulse load with the a/m time ratio value approximating 0,5.

4. The mock-up shall be provided with the reactive loads which can be switched on when operating in the modes similar to generation channel idle mode. Such devices may include:

• capacitance noise filters installed near the ground-power connector not far from the point of onboard a.c. source control;

• input transformers (on their simulators) of separate receivers which can be connected/switched into the onboard network under idle operation mode.

5. The power of each rig motor actuating the rotation of the driver or the generator should not less than twice exceed the generator nominal power, and its response to the generation load change shall not result in the difference of the generator rotation speed transient variables (in value and time) in comparison with the transient variations of these parameters under actual device actuation constituting not more than 10%.

The generation system tests are conducted in compliance with the Procedures N1305-89-VII under the changes of the parameters defining it operation mode in the ranges specified by equipment performance standards and under the estimated operating conditions onboard the aircraft(helicopter):

• source load within the range from 0 up to 150% nominal load under power factor of 0,8-1,0 and in compliance with the actual load operation schedule;

• source load creating current constant component equal to 0,25A;

• irregular phase loads of 0-15% source phase capacity, and phase loads corresponding to the load operating schedule;

• non-linear source load constituting 25% of its nominal power and the loads corresponding to the load operation schedule;

• periodic pulse load constituting 7% of the channel capacity under pulse generation frequency 10-15Hz, and the similar object-assigned load;

• load switching from 10 to 160% of the system (channel) capacity and other limits in compliance with the load operation schedule;

• high dynamic load of the tested object and its switching;

• drivers/generators rotation speeds in accordance with the operating ranges;

• drivers/generators rotation speed variation rates within the operating ranges;

• drivers oil temperatures within the specified limits (if necessary).

During the rig tests of the generation system mock-up the tests and validation of the compliance with the following requirements are performed:

1. The requirements to the generated a.c. power quality at control points (central/main distribution bus) under normally or partially operating system, such as:

• provision of the accuracy of average and absolute voltage and frequency values under steady-state operation modes;
• voltage imbalance under phase irregularity of loads;
• phase voltage shift angle;
• voltage amplitude modulation factor and frequency components and frequency modulation curve under steady-state operation;
• voltage curve sinusoid distortion factor, effective value of the highest harmonics component of the frequency up to 10kHz;
• voltage amplitude value factor;
• constant voltage component;
• transient voltages and current frequency.

2. Each channel shall operate independently of the another and the d.c. power system.
3. The proper operation of the switching-on-and-off source circuits under controls operation and the proper operation of the annunciators under trouble-free operation of the generation system and ground source when the d.c. power sources are either switched on/or switched off are tested.
4. The main distributor bus power supply interrupt durations are validated performing switchings (by the controls) of power sources:
• from one onboard source to another;
• from ground power source to onboard source(s) and backwards.

5. The absence of false operation of the ground source protection, control and voltage monitoring hardware under the simulation of standard steady-state and transient voltage is tested.
6. The compliance with the requirements for the limits of the steady-state and transient voltage and frequency values at control points (main distributor bus), for the limits of the duration of the failed channel switching-off period under abnormal generation system operation is checked.
7. The compliance with the requirements for the non-transition of the failures and abnormal operation of one channel to other electrical circuits and generation system channels is validated.
8. The requirements for generation system safe operation provision in abnormal operation mode due to the below described failures are tested:
• open-circuit in generator feeding wiring to the contactor and open-circuit in the generator zero wiring;
• 115V, 400Hz power supply open-circuit and protection and control unit zero wiring open-circuit;
• sub-driver/control and protection unit wiring open-circuit;
• sub-driver/control and protection unit wiring short-circuit;
• driver actuating coil open-circuits and generator contactor control coil open-circuits;
• switching on of the trip or torque limiter of the generator actuator.

The abnormal generation system operation may occur under the failures of the individual components of the generation system itself and distributing system as well. The tests shall result in testing all the generation system protectors specified by the equipment specifications and in testing the GS operation capabilities i the situations beyond the protectors operation range. During tests the GS abnormal operation is simulated by open-circuits in all power and interunit control drivers, their short-circuiting on the mass and on each other in a separate cable or connector. During each test only one wire open-circuit or short-circuit is simulated. The control wiring short-circuit tests shall be preceded by the engineering analysis of the possibility to perform such a simulation. If the short-circuit simulation results in channel hardware (units and devices) failure, the test is not performed, but the failure results are analyzed.

The damages may exist, which are not displayed, i.e. are not accompanied by the power quality parameters fall outside the tolerable limits estimated for normal operation mode and are not detected by BITE. In this case the experimental tests and analysis of the secondary failure occurance are performed, as the secondary failure in combination with the primary failure may result in hazardous situation occurance.

The Procedures N1305-89-VII specify the test order, techniques and the necessary accuracy of measuring the power quality parameters and the measuring facilities list.

The section "Metrological Certification" contains the validation of the tolerable errors in measuring the power quality parameters in compliance with the requirements applied.

The full-scale power quality analysis in compliance with the Airworthiness Requirements carried out during rig tests of aircraft/helicopter power systems is labour-consuming. When the loop oscillograph is used for registration the manual oscillogram processing is necessary and the necessity in using a wide sensor, pointer and electronic measuring instruments range arises. The documenting of test results is also labour-consuming, besides a number of power quality parameters such as transient voltages modulation frequency pulsation values cannot be practically obtained.

To computerize the power system rig tests our department is developing a measuring-and-control computing complex, which in accordance with the Procedures N1305-89-VII provides the implementation of all test modes during normal PS operation.

For the a/m purposes it provides for standard two-channel power system:

- the capability to control the loads, generator rotation actuators (through Leonard-type actuator control panel), i.e. to program the primary power source operating modes;
- multichannel acquisition and recording into PC RAM and then into ROM (hard disk) of the electrical signals from PS and their further processing (including statistical) to obtain all power quality parameters, specified in GOST 19705-89 standard and the Procedures N1305-89-VII accordingly (including the voltage switching pulses).

Measuring-and-Control Computer Complex Main characteristics

The complex provides input of up to 80 analogue signals, including:

- 115V AC, 400Hz three-phase voltage (from 12 a.c. distributors);

- three-phase alternate current (from 4 feeders of current transformers);
- 35V, 800Hz main generators subactuators three-phase voltage (from two generating channels);
- 28V DC voltage (from 12 distributors);
- source load and distributors direct current (from 12 feeders of current bridge (shunt));
- under voltage level and a.c. frequency and d.c. voltage level ranges in compliance with GOST 19705-89 standard.

The measuring-and-control computer complex provides simultaneous operation (dynamic switching) of up to 16 analogue input channels with the highest iterrogation rate to 160/N kHz, where N-programmed number of analogue input channels (N=2,4,8,16).

The complex provides input of up to 8 discrete input signals with nominal 28V voltage from PS (if input voltage range complies with GOST 19705-89 standard) and output of up to 32 discrete 28V voltage relay and contactor control signals, including signals along 16 channels with the highest load capacity to 3A. Discrete input/output channels have resistance decouplers from the PS circuits.

The complex provides output of the analogue signal along one channel simultaneously with the operation of the analogue input at the rate of up to 300K values per second and is capable of transmitting the signal to analogue input channels for their calibration.

Taking into account that the preliminary calibration of the analogue input channels is performed, the complex provides the static measurement error (and/or calculation error) of the analogue values not exceeding:

- 0,2% - while measuring the a.c. (active current value) and d.c. voltages;
- 0,5% - while measuring a.c. and d.c. values;
- 5% - while measuring the highest level harmonics of a.c. voltage curve, d.c. voltage pulse frequency components and frequency modulation frequency components;
- 10% - while measuring the frequency components of the voltage modulation curve envelope.

To test the PS protection hardware in abnormal operating conditions the complex simulates a series of failures in the system (open-circuits and short-circuits) in compliance with the Procedures/Methodology N 1305-89-VII using contactors and relays controlled by PC.

The Complex Hardware Configuration

The Complex comprises:

- the executive IBM PC/AT-standard complete set with ISA system bus and not less than 4 spare areas for expansion;
- PS communication devices part of which is incorporated in the IBM PC/AT configuration at the system board expansion points and part is mounted in the remote cabinet.

Within the IBM PC/AT configuration the following components are incorporated:

• ADC 12 analogue input/output principal board (product purchased from "Instrument System" Enterprise) providing input/output of analogue signals from/to the PC;

• tunable operating filter board serving to limit the range of the analogue signals incoming to the analogue input/output board;

• buffer amplifiers board serving to connect the remote cabinet modules with the modules incorporated in the PC;

• discrete input/output board providing the input of the discrete signals from the PS and control from the PC via executive electromechanical devices during the PS rig tests.

The remote cabinet comprises:

• two analogue signals conditioning boards providing the conditioning of the levels of the analogue signals from the PS to the standard level of analogue input/output (-7 ± 0.10B) and resistance coupling of the board output circuits from PS circuits;

• analogue signals switching board designated to match the quantity of the incoming analogue signals from the PS (up to 80) and number of ADC 12 (16) analogue input/output board input channels;

• analogue and discrete signals amplifier board providing the PC discrete signal power increase to the level sufficient to control the power contactors and the loading capacity of the ADC 12 board analogue output;

• centralized analogue power supply unit providing the +15V power supply for all functional board analogue circuits mounted in the remote cabinet.

The structural component of 002 type in accordance with the CAMAC-standard produced by Vilnius plant of electrical measuring instruments completed with 043A type power supply unit (output voltage +6V, total output power up to 400Vm) is used as a remote cabinet.

The structural component type has been selected taking into account its sufficient size to comprise the analogue signals conditioning boards Nos 1,2 and analogue signals switching board with a large number of components and due to its low costs.

The CAMAC cabinet bus is at present used only for power supply, but there is a basic possibility to use its total capabilities.

The application of the measuring and control computing complex allows to computerize the rig test performance procedure, test results acquisition, processing and documenting process and to reduce the labour and time consumed 5...10 times.

The development stage: the maintenance and design documentation has been developed and the USO boards are being debugged, the complex Software is developed and debugged.

WAYS AND PROBLEMS OF USING THE CHARACTERISTICS OF FLYING ACTIVITY AND CREW MEMBERS STATUS IN THE CERTIFICATION OF INTEGRATED AVIONICS SYSTEMS

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The problem of determining the objective characteristics of flying activity and crew member state arised in a post-war period, but especially actual it became in the middle of the 60-ies when the need to create transport aircraft with the reduced number of crew members appeared. The reduced crew complement was to solve the navigation and separation tasks in compliance with more rigid requirements and perform landing under weather conditions being worse than the existing weather minima (100×1000). The interest to crew member state parameters evaluation became the highest in the fields of space engineering, submarine engineering, etc. Some research institutes and design bureaus of the industry, Academies of Science and Armed Forces during several years carried out the researches which resulted in obtaining the fullest possible parameters and coefficients of flying activity and crew state, their effect on the safety of flights and crew operation effectiveness. The techniques to obtain such data during experimental researches and flight operations have been developed (See Table 1).

In the 80-ies the parameter evaluation hardware was improved, the activities to specify the nominal parameter values, their acceptable limits depending on flight conditions and the tasks solved, their influence on the safety of flights were carried out.

By the beginning of the 90-ies the techniques for the integrated evaluation of the flying activity results and crew state parameters have been developed.

As far as the administrative and management aspect is concerned, the Guidelines of human engineering of civil and military aviation equipment, tests and operation have been developed. This document specified the procedures and methodology of ergonomic parameter application.

However, due to some circumstances the requirements for evaluating the aviation equipment components compliance with the ergonomic criteria have not been introduced into the Airworthiness Requirements and Reduced Weather Minima Specification Standard in spite of the fact, that actually the aircraft development and certification was accompanied by the estimates of the flying activity and crew member state beginning from the development of TU-154, Yak-42, IL-86 aircraft. The methodology of using ergonomic parameters along with the subjective flight evaluation parameters specified in Airworthiness Requirements has not been worked out.

As a result during the development and certification of the aircraft equipped with electronic display systems (TU-204, IL-96-300) there were some difficulties due to the subjective lead pilots judgments of the electronic display system and optimized display formats options, though their high quality was provided during the experimental tests conducted with the participation of enlarged pilot personnel and on the basis of the objectively existing activities and state parameters.

Additional activities performed in connection with the requirements of those pilots and further practically uneffective modifications led to the increase of costs and delays in certification of the data display systems.

To provide further improvement of the certification procedure and its quality we suggest to study during the discussion and refinement of Airworthiness Requirements draft items and Supplement issuing the following aspects:

- the introduction into the Airworthiness Requirements of the ergonomic parameters and their standardized values depending on the particular situation hazard categories based on equipment failure modes;
- the specification of the procedure and rules of ergonomic parameter application considering the results of electronic display system controls test on experimental and half-scale rigs with the participation of the flight crews and their introduction into "MOC"-document;
- in future to provide the installation on the first aircraft under development during tests of the hardware to monitor the flight activities and psychological and physical aspects parameters.

Table 1

Main Activities to specify human engineering criteria performed during civil aircraft development and hardware used

Aircraft	Activities and Hardware
TU-154	Verification of: load, emergency annunciation response, "Thomson" electronic display system; Optimization of landing mode under ICAO II Category, "Buran" landing elements. Cameras, oscillograph, psychological and physical parameter recorders.
Yak-42	Evaluation of attention and workload distribution, pilot state parameters. "NAC" TV-system, filming, hardware to evaluate psychological and physical parameters (type "Kuvshinka")
IL-86	Integrated evaluation of psychological and physical parameters, attention distribution during tests on the simulator using research experimental complex of special design bureau within the system of State Research Institute of Civil Aviation
TU-204 IL-96-300	Integrated evaluation of psychological and physical pilots state, visual analyzer for extended flights when simulator electronic displays are used. Hardware "Reologia" to perform integrated evaluation of psychological and physical parameters and automatic data processing of special design bureau of BGU.

Proposals to Ammend some paragraphs of
Airworthiness Requirements (Supplement 25)

1. Par. 2.1.3 - 2.1.5: to specify the values of psychological and physical strength parameter, defined by:

,

where:

- mode of electrocardiogram RR intervals statistic distribution;
- mode amplitude;
- variation swing.

In accordance with the Table:

Situation	Flight Condition Sophistication	Complicated	Emergency	Critical
IS	500 - 1500	1500 - 2500	2500 - 3000	3000 and higher

It is possible to apply standardized strength indeces in relative units to reduce variation ranges.

2. Section 2: to introduce the requirement to confirm the compliance with the ergonomic parameters of the particular category situation.

3. Airworthiness Requirements paragraphs dealing with the requirements prohibiting to exceed the situation sophistication criteria under equipment failures should be expanded to include obligatory backgrounding of the sophistication category by the evaluation of the activities and crew members state parameters/human factors.

METHODS OF PILOTS FUNCTIONAL WORKLOAD ESTIMATION PROPOSED FOR USE IN AIRWORTHINESS CERTIFICATION

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Certification of an aircraft, or a product, as specified in standards, is currently based on procedures, rules and methods described in Aviation Rules (AR), Airworthiness Standards, and Means of Compliance.

Until now the process of estimating the man-machine interface (pilot-aircraft interface) has used mainly subjective estimates from experts (evaluation pilots) who applied for this purpose different rating scales such as Tsuvarev scale or Cooper-Harpor scale to estimate DIFFICULTY of performing any operations or actions with a 8- or 10-point scale from "rather small" (0 - 0.2) through "large" (0.6 - 0.7) to "impracticable" (above 1).

At the same time the current AR contain such statements as follow (see Attachment 2):

- mental and physical efforts typical for normal, nonstandard and emergency situations (AR 25.1523 with App. D);
- excess of " crew workload (psychophysiological) above an acceptable level" (AR 25, par. 2.9 (c));
- unacceptability of the crew-workload that does not enable the crew to perform their jobs "accurately and completely" (AR-25, subpar. 2.9.1 (ii));
- complexity of crew reactions to functional failures "shall not require excessive efforts of the crew" (AR-25, par. 3.3.6 and 4.9);
- i.e. such terms as "workload", "complexity of actions" and "excessive efforts", are used.

In addition, it is said in Section "Controllability and Maneuvering" (AR-25, 143 (a)) that an aircraft shall be safely controlled and execute necessary maneuvers in (1) takeoff, (2) climb, (3) level flight, (4) descent, and (5) landing, and "pilot's extremely high qualification, fast reaction or physical force shall not be required... under all possible operational conditions, including:

(1) sudden failure of a critical engine;
(2) failure of the second critical engine; and
(3) configuration change".

But what do "workload", "complexity of actions", "excessive efforts" and "extremely high qualification" mean? What is "an acceptable level" of crew workload (psychophysiologic) equal to? Answers to these questions can be found in the Procedures of Objective Estimation of Control Process and Pilot's Functional Workload developed by NIIAO for comparing the elements of produced information and management field such as displays, annunciators, control levers ("pilot-aircraft" interface).

Experience of practical usage, development and improvement of these Procedures allows the presumption to be made that they can be used to objectively estimate summary parameters such as Control Process Quality (Q_{cp}) and Operator's Functional Workload (λ_Φ) which depend on designs, usage practice, environmental conditions, complexity of tasks to be performed, and operator's psychophysiological state.

The practical usage of these parameters, particularly for the modes [AR-25.143 (a) (1), (2), (3), (4), (5)] and also for other more complex modes such as flight director approach, inflight refueling, furnished statistical data which made it possible to suggest that if the average values of the above parameters fall within specific limits consistent with the Cooper-Hasper rating scale, they can be the estimates of a level of safety for a given aircraft or system under otherwise identical conditions.

Thus, by using the summary parameters, Q_{cp} and λ_Φ, obtained on the objective basis, it is possible to substantiate the compliance of product designs which effect the pilot state and "level of safety" via his "workload", with any requirements of Aviation Rules (Airworthiness Standards) for a number of cases specified in these documents.

Now, some brief comments to the estimation of Q_{cp} and λ_Φ are given.

1. Q_{cp} is determined as the product of the accuracy of a mode, C_A (coefficient of accuracy), by the coefficient of control actions (information yield), C_Y:

$$Q_{cp} = C_A \bullet C_Y;$$

where

C_A, the coefficient of accuracy, is calculated as the sum of ratios of the root-mean-square deviations from specified values of parameters to be maintained to their rated values for the rating "3" at a given flight stage. C_A is determined as a consequence (yield) of system operation which depends on:

- aircraft stability and control characteristics;
- aircraft information field;
- control field (levers and controls);
- quality of an operator as a command control element;

and is calculated from the formula:

$$C_A = \sqrt{1 - \left[\frac{\sigma^2 X_{1\Phi}}{X^2_{1add}} + \ldots + \frac{\sigma^2 X_{n\Phi}}{X^2_{nadd}}\right] \bullet \frac{1}{n}},$$

where

X_i is an appropriate parameter, V_{IAS}, V_Y, Y as a minimum, since H=f(Vy), =f(Y), etc., and $X_i = E_C$ and E_G (course and glideslope deviations) for landing;

$\sigma X_{i\Phi}$ is the actual value of the rms deviation of a i-th parameter in the process of flight mode execution;

X_{add} is the acceptable deviation of an i-th parameter corresponding to the rating "3" of flying procedure standards.

C_Y , the coefficient of yield, is determined as a reaction (yield) of a system command control element (pilot) to current conditions of flight (mode) which takes the form of control actions to remove detected deviations of flight parameters from specified values, i.e. information yield in the control process.

This yield is proportional to the same factors as C_A and additionally depends on environmental conditions and the complexity of a mode.

$$C_Y = 1 - \left(\frac{PCAX_{s\Phi}}{PCAX_{sMAX}} + \frac{PCAX_{a\Phi}}{PCAX_{aMAX}} + \frac{PCAX_{p\Phi}}{PCAX_{pMAX}} + \frac{PCAX_{t\Phi}}{PCAX_{tMAX}} \right) \bullet \frac{1}{n}$$

,

where

$PCA_{i\Phi} = \frac{1}{T}\int_0^T |x_{i\Phi}| dt$, the actual power of operator's control actions (integral) over the aircraft control stick deflections in absolute value centered with respect to an expected value;

PCA_{iMax}= the value of PCA for the maximum deflection of a given lever in a given control channel;

X_s - stick pitch deflections;

X_a - aileron deflections;

X_p - pedal deflections;

X_t - throttle deflections.

The normalization of quantities involved in the calculations is used to make the results of processes of various physical nature universal and abstract and to provide the invariance of calculation procedures for operator's workload, i.e. to make it possible "to mix unmixable", and thus to provide the correct calculations. The normalization and scaling of C_A and C_Y is provided by the following:

$$C_A = \frac{X_{add} - \sigma_\Phi}{X_{add}} = 1 - \frac{\sigma_\Phi}{X_{add}} \quad ;$$

$$C_Y = \frac{PCA_{iMAX} - PCA_{i\Phi}}{PCAB_{iMAX}} = 1 - \frac{PCA_{i\Phi}}{PCA_{iMAX}}$$

The derived coefficients, C_A and C_Y, allow the accuracy and yield criteria to be expressed in DIMENSIONLESS units within the interval $0 < X_i < 1$.

2. OPERATOR'S FUNCTIONAL WORKLOAD λ_Φ

λ_Φ is estimated in terms of the relation of the index of stress after R.M. Bayevsky (IS):

$$IS = \frac{AMo}{MO\Delta x},$$

where
AMo is the amplitude of RR intervals,
MO is the mode of RR intervals,
Δx is the variation of RR interval length.

and the Control Process Quality (Q_{cp}) by using the values of the Coefficient of Load (C_L) and its correction factor (F_c):

$$\lambda\,\phi = C_L - F_c; \qquad C_L = \frac{ISm}{Qcp}; \qquad Fc = \frac{ISm - Qcp}{ISm + Qcp} \bullet C_L,$$

Since the values of IS after Bayevsky are essentially individual, the SCALING of this index is applicable to reduce different numerical values of the index to a common scale.

With the maximum and minimum of IS_{av} in a given day, the value of IS can be scaled as follows:

$$ISmi = \frac{ISavi - IS\phi}{Cm}, \quad \text{where} \quad Cm = ISmm - IS_\Phi.$$

Now, with the values of ISmi for all modes, the estimate of operator's WORKLOAD for a given process can be expressed in RELATIVE UNITS.

The OPERATOR'S FUNCTIONAL WORKLOAD will be the higher, the greater the index of stress and the lower, the better the control process quality for a given index of stress.

On this basis, the coefficient of load is determined:

$$C_L = \frac{ISm}{Qcp};$$

C_L does not take into account the behavior of the Q_{CP} and IS_m functions at the ends of their ranges. To consider this factor, a kind of weight coefficient is introduced as a correction factor for C_L (F_c) which corrects its values at the edges of the ranges of Q_{CP} and IS_m:

$$Fc = \frac{ISm - Qcp}{ISm + Qcp} \bullet C_L,$$

whence

$$\lambda\,\phi = C_L - Fc.$$

By using this formula and the methods of calculation for its components, the great number of experiments (over 2,000) was examined for over 300 flight hours. In addition, the net time of measurement modes was 30 hours in which the following average values of pilots' functional workload were shown for some most applicable standard flight modes:

	Q_{CP}	λ_{Φ}
- climb	0,64	0,33
- level turn	0,79	0,25
- straight level flight	0,82	0,23
- descent for point approach	0,73	0,25
- Cat II manual flight director approach	0,57	0,50

Let us consider AR-25, par. 143 (a), subpar. (2), (3), (4), (5)! If we compare the values of λ_{Φ} from this table obtained on the certified TU-154 with the Cooper-Harper rating scale, we shall see that the table values comply with the suitable criteria of the Cooper-Harper scale.

And this is well natural for these modes on the TU-154 aircraft.

The variations of Q_{CP} and λ_{Φ} were estimated in the process of a 6-hour flight with the use of electronic and electromechanical displays and also flight refueling.

This data allows the presumption to be made that Q_{CP} and λ_{Φ} can be used for the objective estimation of produced aircraft equipment from the ergonomics viewpoint in the certification process. The goal is to obtain not only the ratings of evaluation pilots, but also the objective estimates of the equipment compliance with Aviation Rules (Airworthiness Requirements) at much lower expenses.

In the conclusion it is necessary to cite AR-25, par. 2.5.2: "Certification is conducted in accordance with the current means of compliance or procedures developed by the certification organizations and authorities. The procedures shall be approved by the Aviation Register and included in the Certification Program".

Based on the contents of this subparagraph, the use of the methods of objective estimation for crew workload contradicts by no means the requirements of Aviation Rules, but can enable the certification to be carried out in a more specific way and at less expenses.

Attachment 1

1. Table of expert estimates (emotional) (RIAT-80, section 4, p. 40)

"A" "ACCEPTABLE"	1. I like it a. no comments	- - I (5.3-5.5) fine II (4.8-5.2) excellent
	b. not clear, if there are comments or not	(4.3-4.7)
	c. comments	- - III (3.8-4.2) good, I like it, but there are comments
	2. Not clear, if I like it or no	(3.3-3.7)
	3. I don't like it	- - IV (2.8-3.2) fair, acceptable, but I don't like it
"B" - not clear, if acceptable or not		- - - (2.3-2.7)
"C" "UNACCEPTABLE"	4. It could be worse	- - - - V (1.8-2.2) bad, unacceptable, but it could be worse
	5. Not clear, if it could be worse	(1.3-1.7)
	6. There is nothing worse	- - VI (0.8-1.2) very bad, there is nothing worse VII (0.5-0.7) disgraceful, it is hard to imagine the worse one

2. Cooper-Harper Rating Scale

		Difficulty	Practicability
1.	0 - 0.2	Rather small	
2.	0.2 - 0.4	Small	Fair
3.	0.4 - 0.6	Moderate	
4.	0.6 - 0.7	Large	
5.	0.7 - 0.8	Rather large	Acceptable
6.	0.8 - 0.9	High	Questionable
7.	0.9 - 1.0	Rather high	Unreal
8.	> 1.0	Impracticable	

Attachment 2

AR-25.1523 with App. D, Minimum Crew Complement

	Basic crew functions to be certified	
1.	Flight-path control	demonstration
2.	Collision avoidance	demonstration
3.	Navigation	demonstration
4.	Communication	demonstration
5.	Engine and system control, monitoring	demonstration
6.	Decision-making	demonstration

	Workload factors to be certified	
1.	Simplicity of handling of flight-path, engine and equipment controls	simulator
2.	Accessibility and visibility of instruments, available failure warning devices	simulator
3.	Number, categories and complexity of operation procedures	simulator
4.	Mental and physical efforts typical for normal, nonstandard and emergency situations	simulator demonstration
5.	Character of system monitoring in en-route flight	demonstration
6.	Duties that require the absence of a crew member at the workstation	demonstration
7.	Level of aircraft system automation	demonstration
8.	Load due to communication and navigation features	demonstration
9.	Possible sharp growth of workload in an emergency situation resulting in other emergency situations	demonstration
10.	Loss of crew member's performance capability	demonstration

SECTION 3

Engines and Powerplants

CFD CONTRIBUTION
IN POWERPLANT DESIGN AND INTEGRATION

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1. INTRODUCTION

Powerplant design (nacelle, nozzle,...) and integration become a more and more important problem for modern aircrafts.

Increasing bypass ratio for commercial engines results in large nacelle diameter which integration to wing may significantly affect the aircraft drag. Associated to low pressure ratio fans, the air intakes and nozzles performances play also a more important role in overall performances. Equivalent challenges occur in designing advanced military nozzles and integrating them into fighter aircraft, keeping simultaneously high internal performances and low afterbody drag.

Progress in CFD methods the last fifteen years allows to propose today a new approach using a large set of codes and providing performant designs requiring only final validation tests. Compared to the former approach that used extensive test campaigns, significant cost and time savings as well as quality and performance improvements can be obtained. Moreover, off-design behaviour, such as internal or external flow separation in nacelles, can be rather well estimated, providing an acceptable prediction of the safety limits and making someone more confident with certification success at first run.

The paper illustrates how numerical codes solving aerodynamic flowfield can be used to get better air intakes, nacelles, mixers, thrust reversers and nozzles designs. Some examples of isolated nacelles aerodynamic analysis are shown at low and high speed conditions. Others deal with the wing/nacelle/pylon aerodynamic interactions, including computed results validation on such configuration. Finally, some examples show that viscous codes including modern turbulence models can predict the aerodynamic behaviour of nozzles and afterbody devoted to future fighter aircrafts.

2. TOOLS FOR NUMERICAL APPROACH

Propulsion systems design and integration were previously carried out mainly through experimental approach using both isolated nacelle tests (scale ~ 1/8) and aircraft tests with turbine powered simulators (scale ~1/17). Numerical approach benefit is to procure more detailed flow informations at design and off-design conditions for a large number of parameters, within shorter time than that required by conventional approach, and finaly to provide solutions for improving or eliminating unsuitable configurations. That assumes numerical analysis is able to describe physical phenomena with sufficient accuracy.

Following examples will show that 3D Euler code completed by 3D boundary layer evaluation in non separated cases, or 3D Navier-Stokes code including adequate turbulence model in complex configurations, are well validated and available for numerical approach of powerplant analysis.

SNECMA uses 3D Euler and Navier-Stokes codes based on ONERA works [1],[2]. These codes use 4 steps Runge - Kutta scheme completed by residual smoothing implicit step. Application to powerplant complex geometry needs absolutely an efficient meshing technique. Choice has been made to select structured multi-block mesh topology. In some cases block number become quite important and an efficient interactive and graphic tool is needed for blocks splitting and meshing tasks.

Concerning 3D boundary layer we use an ONERA code [3] based on 3D equations solutions by a characteristics method. The code provides laminar, transitional and full turbulent attached boundary layer main characteristics, using Michel 's or "k - " models for the turbulent case. Only the separation inception point is computed.

3. PROPULSION SYSTEMS FOR COMMERCIAL AIRCRAFT

3.1. Air Intakes and Nacelles

Nacelles and air intakes should be designed at the same time in consideration of a certain number of constraints related to the engine and the aircraft, such as geometrical hard points, radii and slopes at clamps stations, engine accessories, axial position, ground clearance,...

A methodology has been developed [4],[5],[6],[7] using alternatively inverse and direct computations. First the inverse axisymetric potential method [8] allows to optimise the preliminary shapes of three air inlet / nacelle profiles (top,side,bottom). Then using a 3D Euler code, completed by 3D boundary layer evaluation, profiles are modified in order to optimize the cruise performances (internal and external). Another very important goal is to fullfill the off-design requirements both at high speed cruise and during severe unusual take-off conditions leading to very high local incidences (cross wind, one engine shut off,...)

Fig. 1 provides a vertical section of a computed flowfield around a nacelle at flight speed over normal cruise speed. One can see a supersonic pocket ended by a normal shock. Total drag is then evaluated by adding friction drag coming from 3D boundary layer calculation (non separated case) and wave drag coming from integration of normal shock losses based on 3D Euler Mach number distribution. Agreement between tests and theoretical predictions (Fig. 2) is sufficient to determine the nacelle drag divergence Mach number which is an important parameter regard to the aircraft quality.

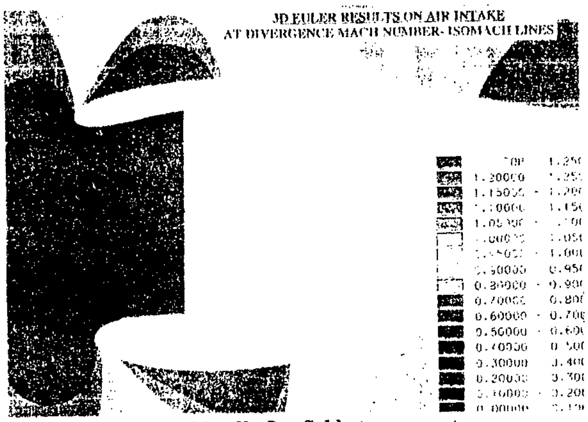


Fig. 1 Nacelle flowfield at max. upstream Mach number (M0 > 0.86)

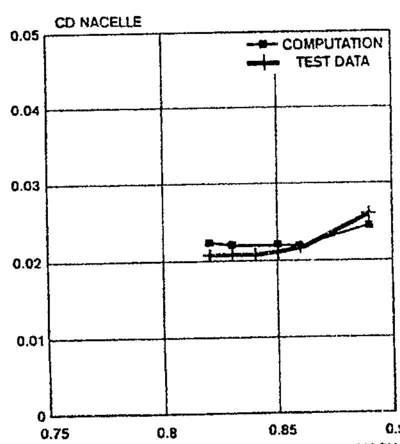


Fig. 2 Nacelle drag versus upstream Mach number (MFR = 0.69 , incidence = +4°)

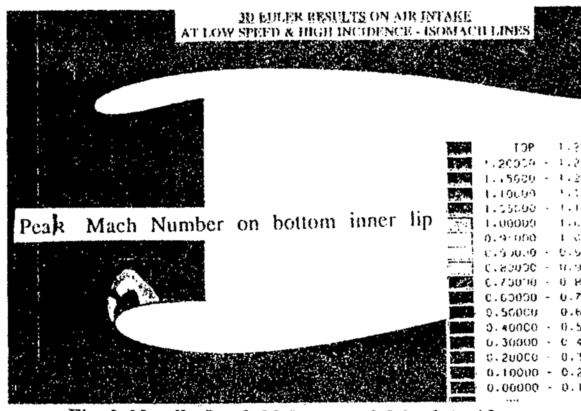


Fig. 3 Nacelle flowfield, low speed & high incidence (typically M0 = 0.25 ,α = + 30 °)

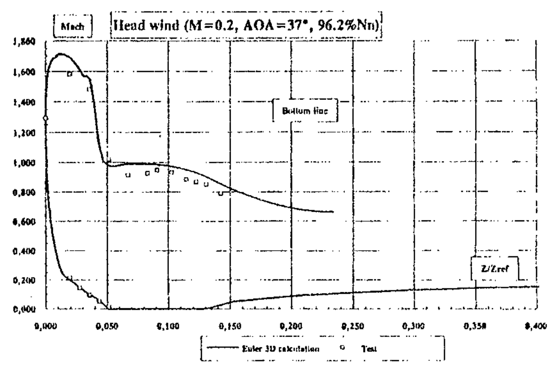


Fig. 4 Mach number distribution on air inlet at low speed & high incidence conditions

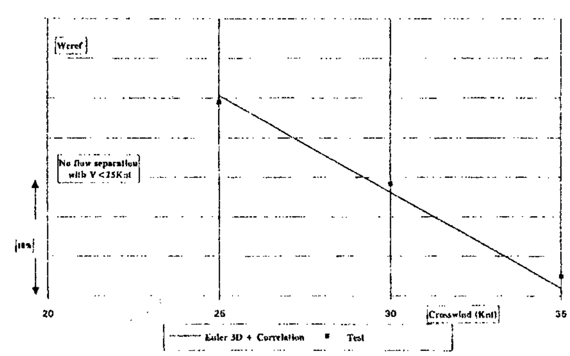


Fig. 5 Prediction of crosswind separation limits

Concerning prediction at low speed / high incidence conditions, 3D Euler code is only used to determine the supersonic peak Mach number. For instance, Fig. 3 shows computed supersonic pocket on bottom inner lip, occuring typically at extreme high incidence operating conditions. Before strong separation, agreement between calculated and experimental isentropic Mach number is very good (Fig. 4). Stall inception prediction is made through empirical correlations using inviscid peak Mach number. Fig. 5 shows that an acceptable prediction of the cross wind operation limits (fan massflow versus wind velocity) can be reached through this way and finaly the suitable rolling take-off procedure can also be determined at design step end.

3.2. Nozzles and mixers

Axisymetric Navier-Stokes computation is used for nozzle design and analysis at cruise conditions. A typical flow structure in a separated flows nozzle for a high bypass ratio engine operating at Mach 0.8 is shown on Fig. 6. In this particular case computation includes a venting flow blown on the core cowl slightly downstream fan exit throat (see zoom on Fig. 6). These results are used to modify the core cowl shape, the blowing axial location, and more generaly all the wall shapes in order to maximize overall thrust in cruise conditions, including the thrust recovery coming from the pressure distribution on vented core cowl

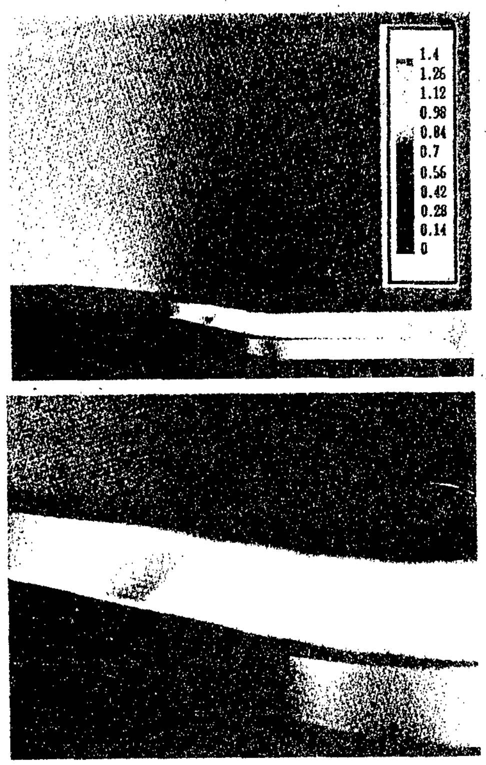


Fig. 6 Axisymetric viscous flow computation
Mach number distribution, cruise, M0 = 0.8

Concerning nozzle flow analysis another important improvement brought by the CFD use is the capability to predict the 3D effects due to the pylon partial blockage inside the nozzle. We are in particular able to predict thrust vector angle within 0.5 ° accuracy using only 3D Euler code. We can also try to reshape the pylon close out and to modify its axial location relative to the throat in order to maintain thrust vector angle within the required range

Mixer design [9] would require 3D navier-Stokes codes to optimize lobes shapes and positions (scalloping,...) in order to obtain maximum temperature decrease in the hot flow stream with minimum losses due to mixer drag. Accurate prediction of thrust and flow coefficients needs a good description of total temperature and pressure maps at exit section. Fig. 7 shows the total temperature maps, from hot area (red) to cold area (blue), at several sections from lobe exit station to nozzle exit station, close to throat section. These results have been obtained using Baldwin-Lomax algebraic turbulence model. Most important flow features are dues to axial vortices induced by spread angle between hot lobe top and cold lobe bottom. Probably, an important part of these structures could be captured by 3D Euler codes. Also, the most important part of lobes design (no separation inside the lobes, vortex distribution control,...) could be carried out by 3D Euler code. But, test/CFD comparison (Fig. 8) points out 3D Navier-Stokes code with Baldwin-Lomax model underestimates mixing intensity. However, radial distribution shape of the circumferencialy averaged total temperature seems correctly predicted.

Fig. 7 Computed total temperature maps
on 4 sections of a forced mixer and nozzle

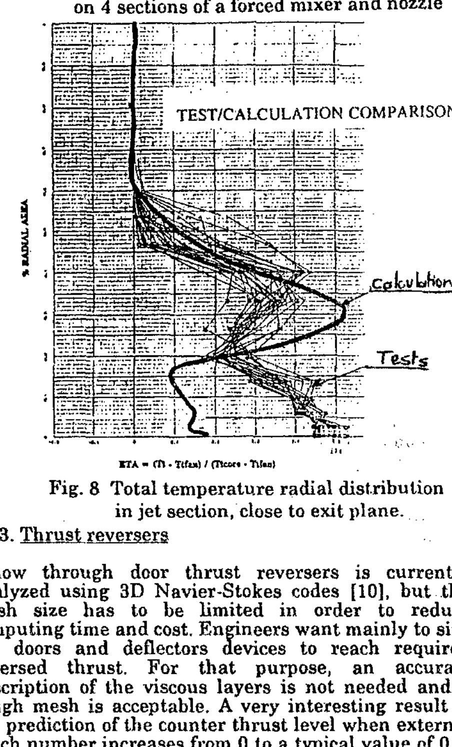


Fig. 8 Total temperature radial distribution
in jet section, close to exit plane.

3.3. Thrust reversers

Flow through door thrust reversers is currently analyzed using 3D Navier-Stokes codes [10], but the mesh size has to be limited in order to reduce computing time and cost. Engineers want mainly to size the doors and deflectors devices to reach required reversed thrust. For that purpose, an accurate description of the viscous layers is not needed and a rough mesh is acceptable. A very interesting result is the prediction of the counter thrust level when external Mach number increases from 0 to a typical value of 0.3. The thrust is obtained by dynalpy flux integration through an "upstream" surface laying on the opening and door boundaries. Then, door back pressure contribution is added.

Fig. 9 shows flow configurations for two Mach numbers during landing process (Mach 0.3 and static condition). For our simplified case (no pressure ratio variation), theoretical prediction indicates at Mach number 0.3 a pure door counter thrust (no ram and nozzle exit dynalpy variations included) 27 % higher than that obtained in static condition. These results allow the aircraft deceleration prediction and also the fan throatling variation control during door opening transient.

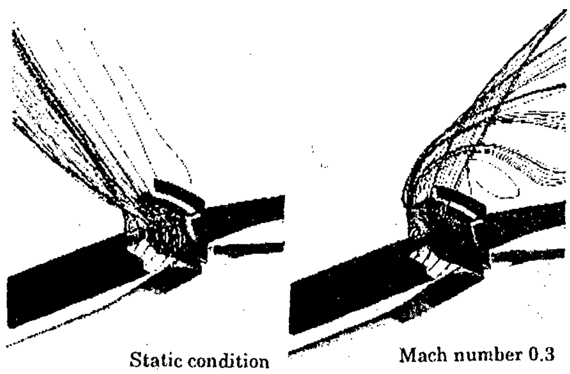


Fig. 9 Computed flow in thrust reverser half door
Streamlines and wall static pressure views

3.4. Wing/pylon/nacelle integration

An important step for the powerplant integration is the wing/nacelle junction by a pylon, especially for large engines and very high bypass ratio engines.

3D Euler codes are now engineering tools to analyze velocity and pressure fields around complete configuration such as fuselage/wing/pylon/nacelle [11], [12], [13]. 3D boundary layer calculation based on static pressure field coming from the 3D Euler code provides interesting data to the designers. For instance, one can evaluate nacelle drag, boundary layer separation lines on various walls (nacelle, pylon,..). Moreover, these tools (3D Euler + 3D boundary layer) can be used at design step to reshape nacelle or pylon, to modify nacelle location, etc.. in order to reduce shocks intensities, local separation amplitude or boundary layer thickness.

We provide hereafter an example of fuselage/wing/pylon/through flow nacelle computation. The computed geometry is a DLR/ONERA configuration named "F6" which has been used for both theoretical and experimental cooperative research works. The aerodynamic test conditions are an infinite upstream Mach number of 0.75 and an incidence of +1°.

A 19 blocks, structured, H-shaped mesh was needed, resulting in a total of 980,000 nodes. For engineering purpose, 3D Euler computation required 2000 iterations with a CFL value of 5 resulting in a CPU time consumption of about 10 hours of CRAY-YMP computer. Fig. 10 illustrates only the wall mesh in nacelle/pylon region. Fig. 11 gives a view of the wall Mach number distribution computed from wall static pressure and upstream total pressure. One can see sensible peak Mach number and associated strong diffusion on rear part of the inboard sides of both nacelle and pylon. Fig. 12 shows this calculated overspeed is in very good agreement with measured values. In addition the changes in local wing lift induced by nacelle/pylon effects is in a satisfactory agreement with the experiment.

Finally, we show on Fig. 13 the displacement thickness distribution on nacelle and pylon inboard sides. This figure indicates also wall streamlines. The wall streamlines slip and the strong boundary layer increase at the nacelle/pylon junction near nacelle trailing edge is consecutive to diffusion following the overspeed. The calculation stops when separation occurs; that is roughly where wall streamlines drawing stops.

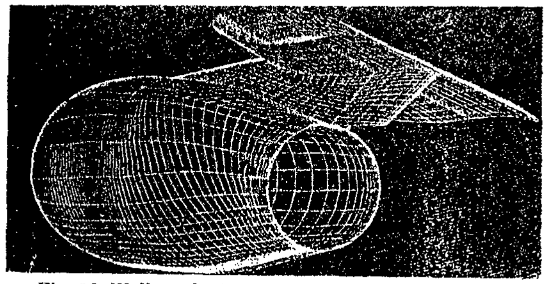


Fig. 10 Wall mesh view in nacelle / pylon zone
"F6" configuration- 19 blocks- 980,000 nodes

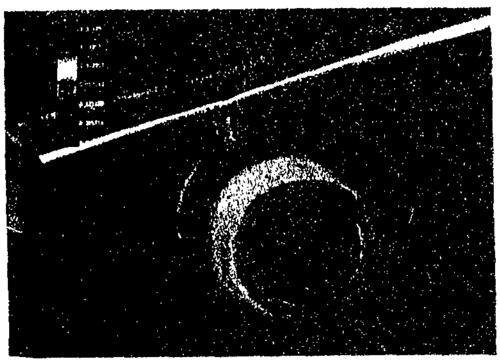


Fig. 11 3D Euler computation on nacelle/pylon/wing
Wall Mach number, M0 = 0.75, α = + 1°

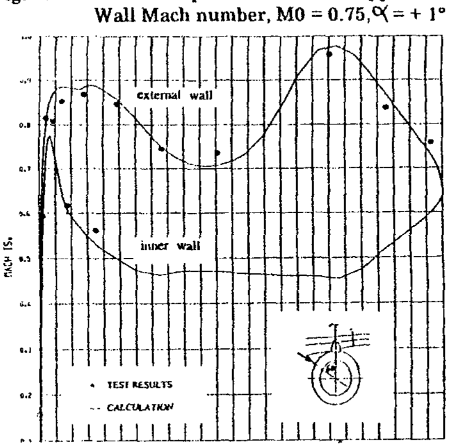


Fig. 12 Nacelle/pylon/wing flow computation
Test vs. calculation comparison

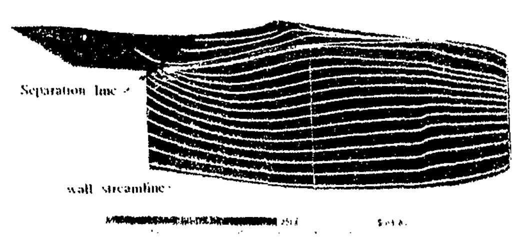
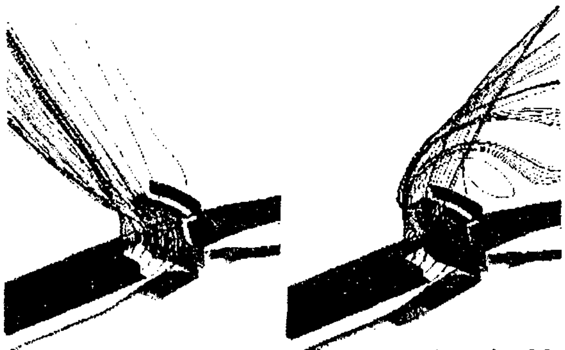


Fig. 13 3D boundary layer computation
Displacement thickness distribution
Wall streamlines drawing



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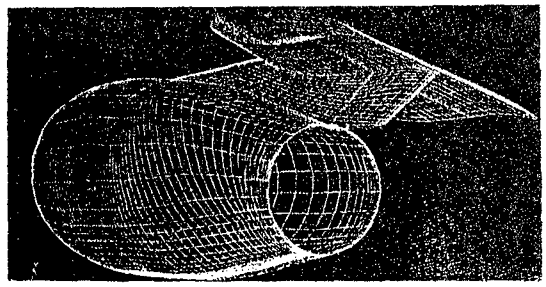
Streamlines and wall static pressure views



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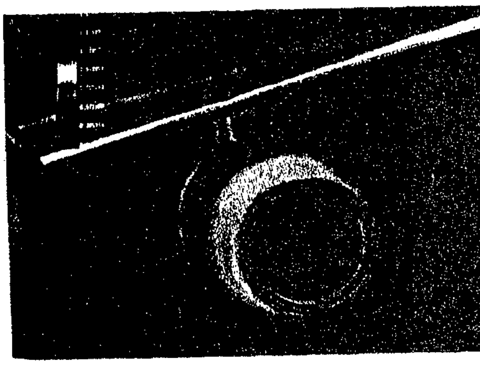

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Wall Mach number, M0 = 0.75, $\alpha = +1°$

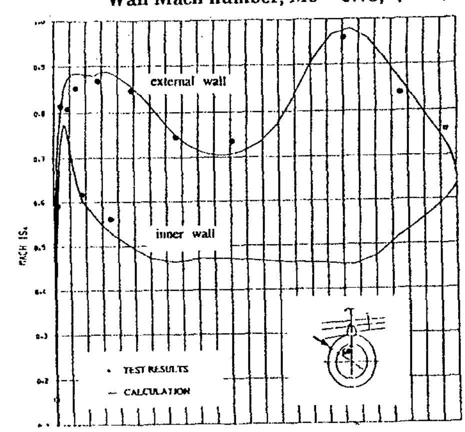

Fig. 12 Nacelle/pylon/wing flow computation
Test vs. calculation comparison

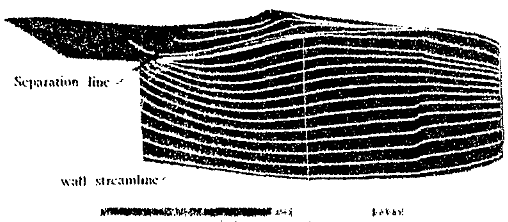

Fig. 13 3D boundary layer computation
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Wall streamlines drawing

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4. PROPULSION SYSTEMS FOR MILITARY AIRCRAFT

The final purposes is to use axisymetric or tridimensional Navier-Stokes codes during nozzle and afterbody final design step [14]. Flow complexity with possible 3D separation and the necessity to optimize overall performances (internal thrust plus afterbody drag) lead to close cooperation between airframers and engine designers

Navier-Stokes have been firstly validated on axisymetric afterbody having a simple convergent nozzle or a base blown nozzle. Tests have been carried out in the S3ch ONERA wind tunnel at an external Mach number of 0.8. Agreement between computed parameters and experimental values is satisfacory. When a two-equations model ("k -ε" model) is used to simulate turbulence effects, the Navier-Stokes code reproduces very well pressure distribution on afterbody external wall.

The axisymetric code has already been applied to improve base-ventilated nozzles. Fig. 14 and 15 provide the computed flowfield descriptions with subsonic and supersonic external flow. These data have been obtained using Balwin-Lomax turbulence model and are very useful for engineering purpose. For instance, geometrical arrangement of the cold flap trailing edge versus the hot flap one's, as well as the base massflow, have some effects on the base pressure. These effects can be investigated with sufficient accuracy by CFD tools.

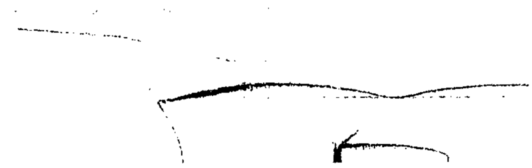

Fig. 14 Axisymetric N-S computation, base blown nozzle
Subsonic case (Mo = 0.8), Pt_{jet} / Po = 4.8
No afterburner

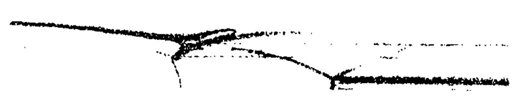

Fig. 15 Axisymetric N-S computation, base blown nozzle
Supersonic case (Mo = 1.6), Pt_{jet} / Po = 14.
Afterburner on

In order to optimize complex afterbody configuration with non axisymetric nozzle, 3D Navier-Stokes codes are needed. Validation of such a code is made using a less complex afterbody geometry such as the 2D NASA-Langley afterbody test case [15]. Fig. 16 describes the computed flowfield on the two planes of symetry (horizontal and vertical) at external Mach number of 0.94 and pressure ratio of about 4. These results, obtained using the algebraic Balwin-Lomax turbulence model, show a strong shock followed by a separation occuring on the horizontal side, while the flow remains attached on the vertical side which is almost flat

Fig. 17 presents comparison between computed and measured external static pressure on the vertical plane of symetry. Computed diffusion within the separation on the horizontal wall (downstream the high curvature area and the associated strong stock) is correctly provided by code using two-equations "k- ε" model. Using Baldwin-Lomax model slightly overestimate pressure gradient and trailing edge static pressure.

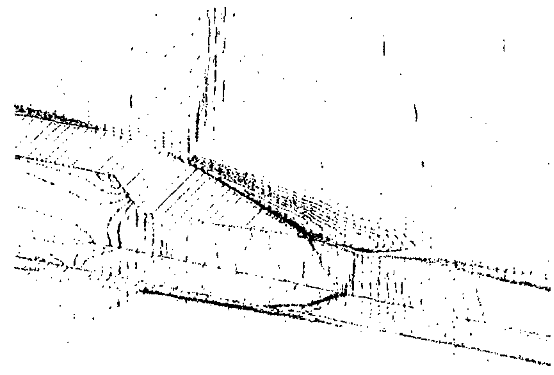

Fig. 16 3D N-S results on 2D NASA-Langley afterbody
Iso-Mach lines on vertical & horizontal sections
External Mach number = 0.94, Pressure ratio = 4.

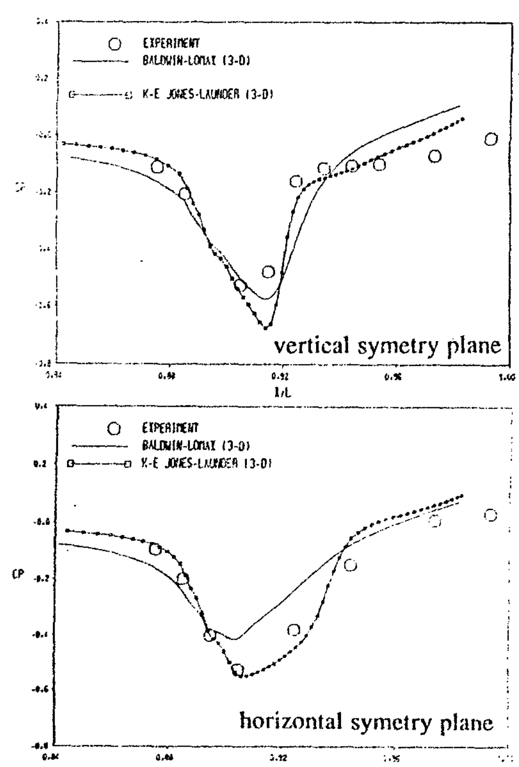

Fig. 17 3D N-S results on 2D NASA-Langley afterbody
Wall pressure on vertical & horizontal sections
External Mach number = 0.94. Pressure ratio = 4.

5. CONCLUSIONS

This paper has presented some cases of powerplant design and integration using extensively the numerical approach.

The great advantage of this approach is to optimise the configurations at design step before model manufacturing and test running in order to save time and money.

An additional advantage is to determine the most important phenomena areas and so to help the measurements location.

Generaly codes compute accurately flows close to design points. For conditions very far from design condition, empirical correlations based on 3D inviscid calculations are sometimes needed to evaluate design quality.

Further developments are needed in following areas :

- Description of large separated zones
- Computation of mixing levels
- Mesh and CPU time consumption needed to run advanced turbulence models

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COMPUTER TESTING SIMULATOR FOR AVIATION MOTORS

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INTRODUCTION

CIAM owns a unique test facilities for research of real aero-engines and its components. Aircraft engines for different applications can be tested in simulated flight conditions up to altitude $H \approx 20$ km and flight Mach number $M \leq 3$. There are special rigs for testing small turbojet and turboshaft engines under simulated flight altitude - speed and climatic conditions and for gas dynamic, heat transfer and strength testing of gas turbine engine components.

In this paper we consider the creation of the Computer Turbo-engine (Turbojet, Turbofan, Turboshaft and other) Test Technology - the CT^3 system and its performances. The CT^3 system must increase greatly the possibilities of our natural test rigs for researches of aircraft engines and its components.

We shall present the main peculiarities of developing the CT^3 system. It is based on complex 2D and 3D mathematical simulations of aerodynamics, heat transfer and stress problems. After the thermodynamic design, when we have the drafted project, the CT^3 system is included with the schematic diagram showed in fig. 1 (see on the next page). First of all CT^3 allows to carry out the detail aerodynamic design of the whole gas turbine engine system. Here it's emphasized the main difference from traditional ways of engine system design. Up to the present time the mathematical engine simulation used 1D and quasi-2D models [1-4], which demanded a long time for verification testing and development. Application of very accurate 2D and 3D models of the CT^3 system allows to get more optimal aerodynamic project of engine system without continuous real engine testing.

On the first step CT^3 applies for major steady regime simulations (the cruise and take off regimes). Here we can accurate simulate the equilibrium running points for a series of operating conditions and obtain performance curves of power output or thrust, and specific fuel consumption, when all components are linked together in an engine.

Analyzing these regimes, we can design a better variant of the working project and study off-design performances. Beginning from the off-design steady regimes (reduced and maximum powers, idling, autorotation and oth.) and having a better next variant of the working project we must analyze the very important unsteady regimes (starting, surge, burning put or out and oth.). As a result of detail aerodynamic engine analysis we are very close to the optimum working engine flow passage project.

After that, also with the help of the perspective CT^3 system we can carry out the mechanical detail analysis. On this step of design, there will be carefully solved heat transfer, stress, vibration, reliability and resource problems. Here we will deal with the finished engine project, which is used for the manufacturing of experience engine. The next step of wide using of the CT^3 system will be an accompaniment of real ground and flight engine testing and its certification. The CT^3 system must raise the engine testing to the new high quality level, where we will have essentially more information on all test regimes.

In the frame of the considering research the special software system of scientific analysis and visualization is developed. This system conditionally named Scientific Operating System (SOS) allows us to improve greatly our analysis capabilities, as well as the ability to view portions or all of the numerical data, to have a static and animated 2D and 3D pictures in color presentation.

The computer 3D engine model must accompany the whole engine life — from design to production, uprated and modified versions, and to exploitation on aircraft. Estimation of necessary computer requirements for the realization of the CT^3 shows us the first stage of CT^3 may be developed using wide spread work stations with RISC processors. Similar gas turbine simulations used earlier more power computer systems [5,6].

MATHEMATICAL MODELS OF CT^3 SYSTEM

There are considered the direct problems for whole 3D flow passage with intake, fan, bypass channel, lower and high pressure compressors, combustion chamber, lower and high pressure turbines, exhaust components and initial part of jet. Simulation takes into account the main physical effects such as viscous losses, axial and tip clearance leakages, selection and blowing out of cooling air, inertia of rotors and some mechanical losses. Angular velocities ω_j for each rotor j can be known or defined in solution from power balance conditions of compressors and turbines.

Initial data of the problem are given approximate flow parameter distributions in a whole engine passage in the time moment $t=0$. These data can be calculated either from 1D solution for simulated engine work regime or obtained very simple as at engine start simulation. The boundary conditions are: full 3D geometry of flow passage (including blades, combustion chamber, etc.), specific fuel consumption and heat of combustion, air cooling scheme, gas leakages, total parameters (pressure, temperature) and velocity direction distributions at the inlet section AB and static pressure distribution at the outlet section CD. All boundary conditions may be time functions in a common case.

Depending on the location of flow passage parts either the absolute cylindrical coordinate systems (stator regions) or relative one (rotor regions) will be used.

There are solved 3D Euler and Navier-Stokes equations in conservative forms in cylindrical coordinate system (z, r, φ).

The solution of unsteady 3D system for full gas turbine flow passage is a very expensive task. At present time for practice applications there are used some simplifications with "averaging" procedures.

3D formulation

Here is used the first simplification, when the number of cells in angular direction φ equals or less the maximum number of blades in each rows. In these case it is fulfilled averaging in each blade passage or in a few blade passages. Such formulation allows us to simulate unsteady 3D working regimes with axial, radial and angular variable parameter distributions. In our study this is the most complex formulation.

Quasi-3D formulation

Here is used the averaging in the middle of each axial gaps [7]. In this approach the gas flow within each blade row (or each engine components) is defined by full 3D equations. At the sections located approximately in the middle of each axial gap the non reflection averaging of parameters in angular direction is fulfilled [8]. It is required that mass, momentum and energy fluxes through the ring from r to $r+\delta r$ are the same for both sides of the averaging section. Such formulation allows us to consider only one blade passage in each rows. In this case some information in angular direction are loosen.

S_2-surface formulation

Here is used the averaging in angular direction everywhere. In this approach the axis symmetrical 2D problem is solved. Such formulation allows us to simulate unsteady 2D working regimes with axial and radial variable parameter distributions. In detail Euler and Navier-Stokes formulations on S_2-surface were presented in [8].

S_1-surface formulation

Here is used the averaging in radial direction everywhere. In this approach the 2D blade-to-blade problem is solved on S_1-surface obtained by rotating of the streamline. Such formulation allows us to simulate unsteady 2D working regimes with axial and angular variable parameter distributions. Also in [8] Euler and Navier-Stokes formulations on S_1-surface were presented in detail.

NUMERICAL SOLUTION AND RESULTS

Numerical solution of 2D and 3D Euler equations were obtained by implicit monotone high order accuracy methods [8,9]. These methods are used the main ideas of explicit method by S.K.Godunov [10,11]: the monotonicity and the arbitrary discontinuity breakdown. It is the conservative shock capture method, allowed to reflect very carefully the local flow structure. The piecewise parabolic parameter distributions into each computational cell are applied.

The full by-pass gas turbine engine simulations are shown in fig. 2-7. The geometry of whole flow passage is presented in fig. 2. The computational grid for this task is shown in fig. 3. The results in meridional plane for cruise steady regime are following: "mass flow rate" contours (fig. 4), static and total pressure contours (fig. 5 and 6) and total temperature contours (fig. 7).

CONCLUSION

The CT^3 system allows to simulate steady and transient working processes in detail and predict accurate turboengine performances. As a result of such detail aerodynamic flow passage analysis there are approaching to the optimum working flow passage project.

Detail verification of CT^3 allows to have a new high level quality of engine testing, when the measuring point number equal the point number of numerical grid. Here in any point all parameters are registered and whole flow passage is transparent (it can observe the research physical process in any region on computer display using impressive color graphic system).

The CT^3 system can essentially influence over its competition capacity. There is opened the new advertisement possibilities. So all important performances can be presented very impressively using the transparent computer simulator.

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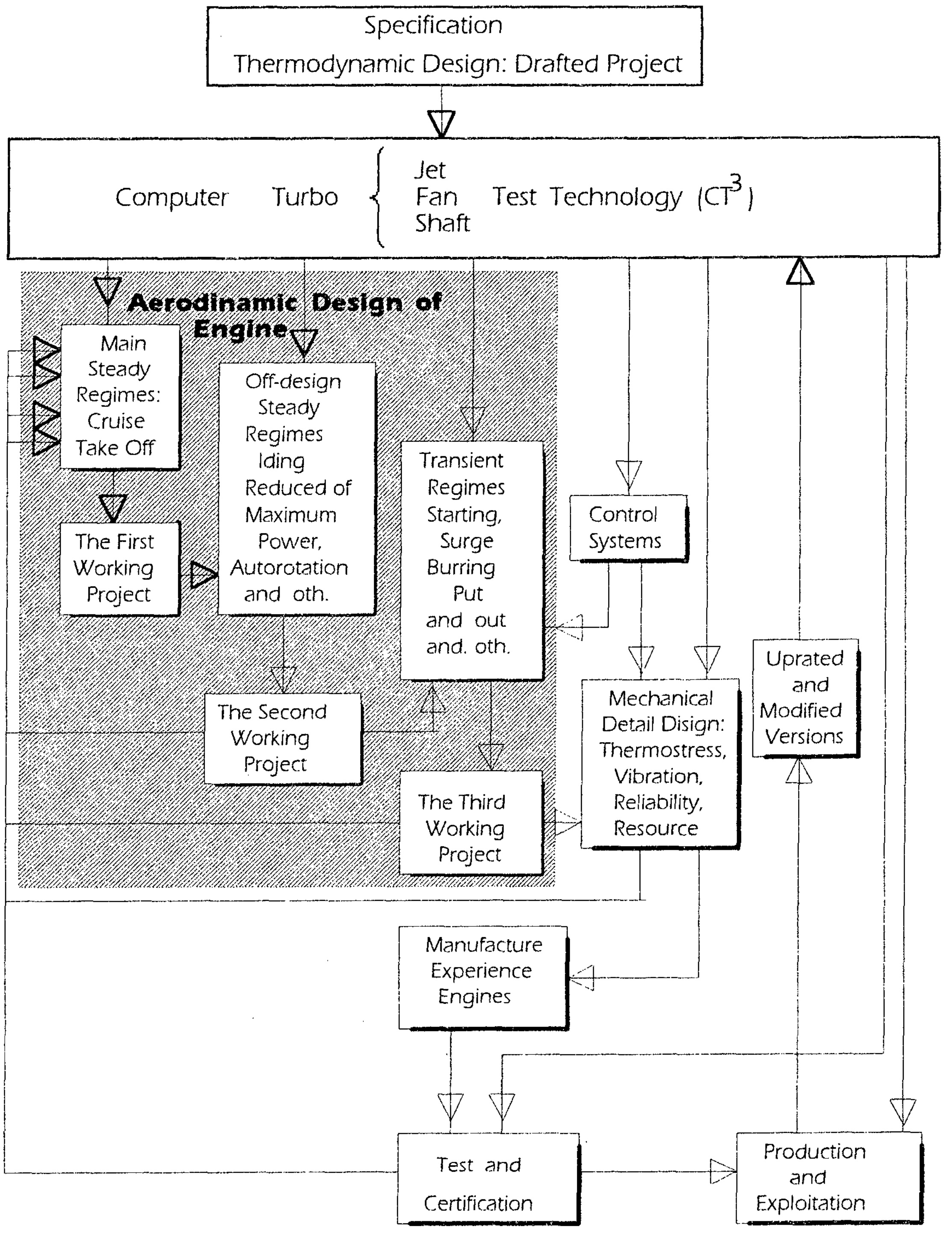

Figure 1. CT^3 Application

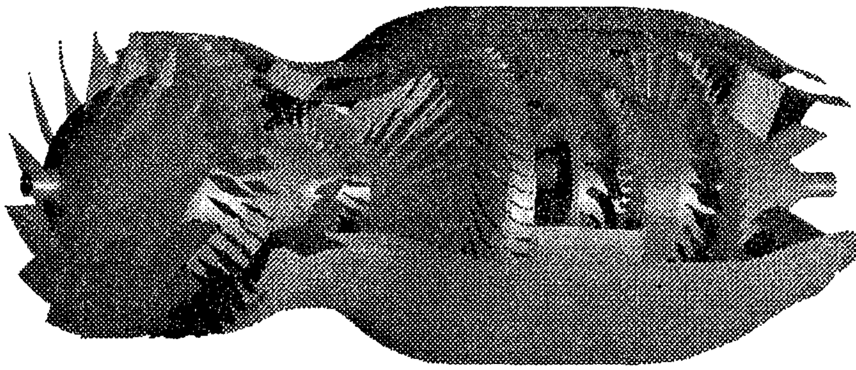

Figure 2. Looking through the whole engine

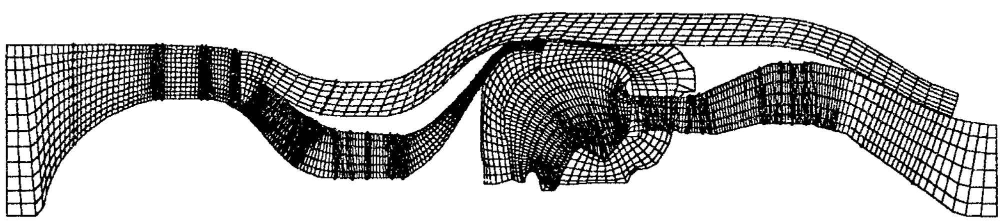

Figure 3. Computational grid for by-pass gas turbine engine

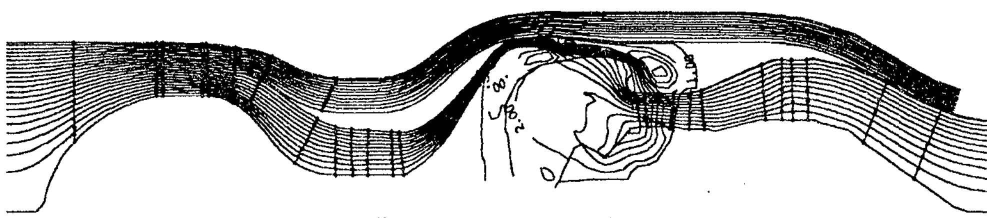

Figure 4. "Mass flow rate" contours (steamlines)

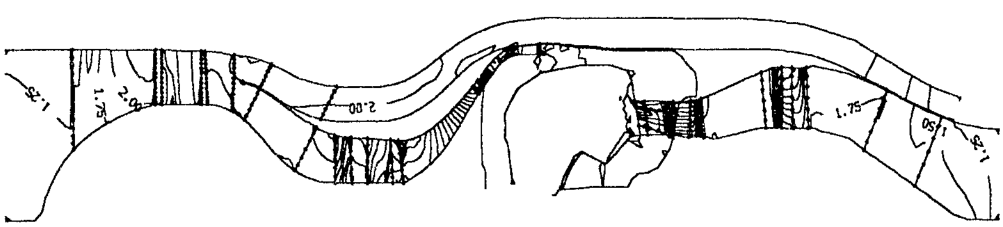

Figure 5. Static pressure contours.

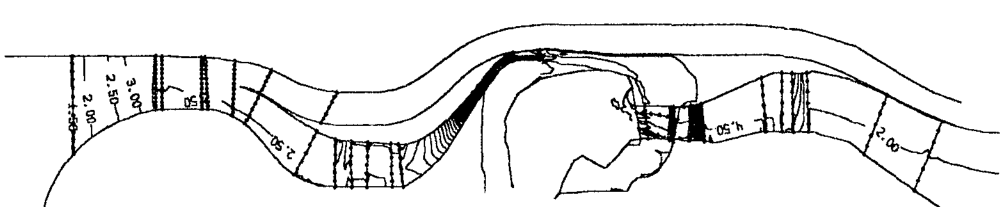

Figure 6. Absolute total pressure contours.

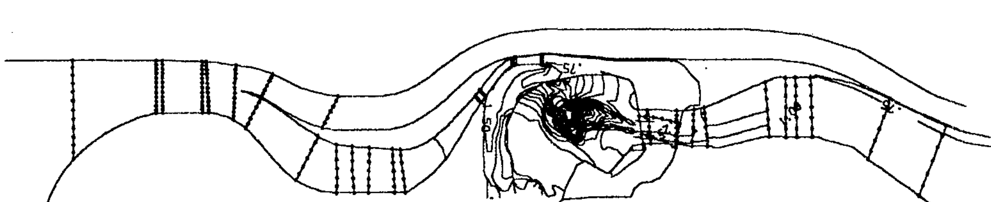

Figure 7. Absolute total temperature contours.

PROVISION AND SUPPORT OF "NK" FAMILY AIRCRAFT GAS TURBINE ENGINES AIRWORTHINESS

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NK-ENGINES, Samara, Russia

The problem of aircraft gas turbines airworthiness provision was always actual but nowadays it became on acute problem due to requirement of improvement of gas turbine parameters with provision at the same time gas turbine engine operating life improvement and better reliability.

Conceptions of reliability and service life are permanently being improved and developed in the process of technics and structural strength science evolution.

Actuality of gas turbine service life problem grows due to broad application of used aircraft engines for mechanical drive in gas pumping stations, as electric generators and as other power generation units]. Conception of long service life is a quite relative one. It seems that the term long service life should be considered as such value of operating life at which time factor is assigned to play a specific role leading to deterioration of material strength and plasticity characteristics, variation of details surface layer state, lowering of strengthening techniques effects, variation of separate engine units operating parameters and characteristics during engine operational service.

High reliability values and long service life are peculiar feature of "NK"-family engines. Rates of "NK"-family engines service life increase and achieved reliability values are the highest as compared to those of home-produced engines. High reliability and service life values are achieved owing to specially developed methods of engines design and development.

Academician N.D.Kuznetzov claimed that gas turbine engine reliability is laid at design stage, is improved during development, and is provided by production manufacturing, is supported in operating service.

Main principles of engine parts high structural strength, reliability and operating life provision in general are provided still at design stage giving the possibility of engine diagnostics and operation on condition and with acceptable defects as well as low maintenance man-hours flight hour. Provision of engine operability at design stage requires the analysis of similar prototype — structures typical defects and preventing their appearance in structure under development.

Development of engine separate parts and the whole engine specifications is the basis of serviceability provision at design stage. Specification which is based on previous engines operating service experience and on present day requirements includes the requirements for parameters, serviceability, reliability, operating life, producibility operating characteristics, maintainability, possibility of engine condition diagnostics, fail-safe etc.

Specification requirements are provided taking into consideration the provision of operating life and observing design principles. These principles are formed on the basis of the whole company experience in "NK"-family engine development as well as other gas turbine designers experience. Nowadays main elements of modern design includes the following:

- use of high level computer models, both for gas dynamic design and for parts stress-strained state analysis as well as structure elasticity;
- consideration of real operating condition together with taking into account defects summation methods at multifactor loadings;
- consideration of potential variation of engine operating parameters and engine parts condition within operating time;
- provision of parts high structural strength due to design method which provides minimum stress concentration, optimization of material selection and blanks manufacturing methods, wide application of surface strengthening methods, surface protection (especially engine flow duct details surface) from corrosion and erosion;
- provision of engine control ability owing to built-in check systems for engine and its main units operating parameters control, and suitables access to critical engine units for their control during scheduled works;
- provision of engine parts and the whole engine fail-safe operation.

Figures 1, 2, 3 (as examples) show the results of calculation of power turbine module, gas generator LP spool rotor and gas turbine load-carrying casing.

Computer calculations of high level allows to estimate more precisely stress and dynamic conditions of parts and engine as a unit; to optimize rotor supports location and engine attachment points; to ovalization minimize skewness and radial clearances variation taking into account cases.

Validity of calculations was verified by comparison of calculation results and experimental ones. Fig. 4 shows natural frequency calculated and experimental data comparison results for turbine wheel.

It is shown that calculated and experimental data are in good coincidence.

The use of high level computer models are to be based on valid initial data for parts loading in real operating conditions and materials properties.

Statistical data analysis of civil aircraft engine operating conditions shows that operating conditions for different aircraft of one and the same class are similar and it makes possible formulation of one integrated flight cycle for these planes characterizing gas turbine engine loading averaged conditions.

Gas turbine engine main parts and units operate in multifactor loading condition. These main components and parts of gas turbine engines are undergone continuous static loads and many parts at high temperatures, at steady-state operating modes, low-cycle loads due to engine start-ups and shut-downs and variations of operating modes, high frequency vibrations induced by engine operating processes.

Different kinds of loads influence lowering engine operating life. Engine parts operating life can be estimated taking into account multifactor loading with of various methods [1, 2].

NK-family engines service operation experience shows that within the frames of TBO turbine inlet temperature can grow by 50°C and this is taken into account for engine operating life evaluations.

Properties of the materials used is one of the most important factors to provide high characteristics of engine structural strength.

The term structural strength is considered as a complex of characteristics defining material strength with regard to structure influence determining the kind of stressed state and stress concentration, manufacturing technology determined by material micro and macro structure, surface layer condition, welding availability etc. as well as the influence of operating conditions which are characterized by temperature, medium and loading type.

The most important structural strength characteristics are those ones related to time loading: long duration strength, creep a stress relaxation, low cycle durability, erosion and corrosion resistance, crack propagation rate at low-cycle loading.

Structural strength is experimentally analyzed with the use of specimens and full-scale parts; in this case each characteristic is defined taking into account structural, technological and operating factors.

Figure 5 shows an example of a test bench for turbine blades test with combined thermocyclic and vibratory loading.

There are 139 test rigs and benches at our company for engine components structural strength and reliability development.

The important aspect for "NK"-family engines" safety provision is wide use of finishing strengthening technologies. More than 60%

engine parts surface plastic strain deformation are undergone and it increases parts fatigue limit 1.2—1.5 times, improves wear resistance and contact fatigue life 2—4 times, eliminates stress concentrator caused by machining, stabilizes long duration strength and low-cycle durability characteristics and therefore improves safety and engine service life.

Besides resistance to vibration is provided by wide use of damping, frequency adoption, different spacings between nozzle vanes [3]. Different spacings between nozzle vanes provide 2—3 times lowering of resonant stress levels.

Compressor blade surface protection is performed by spraying on it anti-corrosion coatings based on titanium and vanadium nitrides, turbine blade surface protection is performed by spraying on it heat-resistant coatings based on NiCrAlY. Additional heat-resistant coating (HRC) based on ZrO_2 stabilized by Y_2O_3 is sprayed on turbine first stage cooled blades. HRC-structure is shown in Fig. 6.

Fig. 6. Heat-Resistant Ceramic Coating Structure

The required level of convective cooling is the necessary condition of providing beneficial effect of HRC. Operating time of HRC-coated blades at maximum blade surface temperature up to 1000°C is higher then 5000 hours. HRC-coated blades are still widely used. HRC not only reduces blade temperature, but also insulates its surface of corrosion-erosion effect of high-speed gas flow.

Engine development stage: provides the following :

— rated values of fuel economy, no harmful effect on ecology and reliability;
— manufacturing technology providing stable characteristics of structural strength, accuracy of engine parts and the whole engine production and assembly;
— rates of acceptable part defects and levels of alternating stress without provoking possible defects in the process of service operation.

The drawbacks are eliminated basing on the supposition that every revealed drawback has its own actual and definite origin or can be explained by complex of reasons the probability of which realization must be lowered as much as possible.

Special tests providing comparison of parameters required by Specification and rated as a result of development as well as engine operability at extreme conditions.

The following is verified during special tests:

— gas turbine, its parts and details vibration and thermal state ;
— systems parameters and operability;
— characteristics of operating service.
— reliability and survivability of gas turbine engine.

If is very important to determine on the basis vibrational state date limiting level of details alternating stress providing elimination of potential production defect extension and failures service operation. The investigation of compressor and turbine blades with crack in places of dents locations induced by foreign object ingestion showed that blade crack-resistance is within the limits of 60—80 MPa. Therefore maximum permissible level of vibration stress at resonant modes for "NK"-family engine blades is not higher than 60 MPa that provides elimination of defects extension caused by damages at operation and possibility to reject defective blades during gas turbine regular inspections. Depending on the degree of blade damage the decision is made to continue operation, to clean out dents without engine dismount or to dispatch engine for repair.

Gas turbine engine thermal state evaluation consists of " hot-part", rotor bearings, fuel and lubrication system elements. Working medium temperature and pressure fields and thermal stability characteristics of fuels and oils are determined also.

During evaluation of operability and parameters of fuel and lubrication systems the following as determined:
— lubrication system characteristics;
— quality of air off takes for different especially, for aircraft needs;
— operability of starting and control systems versus ambient and fuel temperatures;
— efficiency of warning and gas turbine engine protection systems.

Operating characteristics verification includes:
— evaluation of rated fuel economy values;
— evaluation of gas turbine engine operation stability at extreme positions of control system;
— verification of gas turbine engine reliability at rotor overspeed and operating body over temperature at various oil temperature values.

Gas turbine engine survivability evaluation includes:
— engine operability verification at engine inlet foreign object ingestion (water, ice, birds, aircraft structure elements, etc.);
— verification of fire non-proliferation outside gas turbine engine at titanium blade out and gas turbine engine casing integrity at compressor/turbine blade out. Case integrity test is performed at special test rig to separate blade at the required rotor speed and in the definite point on circumference, and to register quickly-running process of impact interaction of ruptured blade with gas turbine engine case. Test rig scheme for case integrity test is shown in Fig. 7.

To provide case integrity it is very important to have blades with sufficient elasticity. The number of secondary damages of side by side blades lowers when blade elasticity is sufficient and it in its turn lowers the probability of case integrity loss with debris.

Life tests, which are performed, as a rule, according to accelerated programme, are the final stage of special tests and gas turbine development.

Accelerated equivalent tests represent time-simulated operating tests, which are equivalent to the required operating life exhaustion of main parts and units durability.

Equivalent tests are programmed on the basis of operating conditions analysis and conformity of load level and durability for every loading kind and on method of damage summation. Test time save is provided by means of less loaded operation modes reduction to more loaded ones.

This equivalent test program makes it possible to reveal general defects in units and parts with short time, to check measures aimed to eliminate them and to evaluate engine operating life. Equivalent tests efficiency was estimated by means of comparison of the defects revealed in the accelerated tests as well as those ones revealed in service operation. 25 of 26 defect types usually revealed while operating service were registered in the course of accelerated tests, which proves test programs high efficiency.

Altogether more than 50 types of special tests are performed while engine development, including check-up of its parameters, serviceability and reliability characteristics. Actual operating conditions simulation is one of the principal peculiarities of the development stage and special tests realization.

Reliability provision in serial production is based on the development of technological reliability brought about by manufacture quality of experimental engine. Technological reliability requires selection of the most critical manufacture processes, their preliminary research and selection of the most suitable manufacture technologies providing manufacture reliability.

Sufficient blade plasticity is important for provision of case impenetrability. Amount of secondary damages of near positioned blades reduces probability of case penetration by detached fragments.

Investigation and eradication of negative technological heredity provision of quality and technological processes stability are an important aspect of technological reliability provision. That is achieved by machining method optimization, use of strengthening finishing, technologies, working out of the new technological processes and manufacturer participation in serial production mastering.

The issue of starting intermediate products is important in technological reliability provision. Therefore micro-structure and strength performance checking for the most important parts is being introduced.

Mechanization and automation of technological process and individual operations and methods of machining is the necessary constituent of technological reliability provision.

Complex problem of gas turbine engine reliability and high operating hours support and provision includes several interconnected problems based on analysis of main causes resulting in engine early ground.

The main of these problems as follows:
— study of gas turbine engine potentialities;
— working out of methods for engine operating life exhaustion evaluation, engine availability provision and maintenance while operating service;
— development of engine repair technologies providing higher engine details durability with minimum costs.

Gas turbine engine potentialities are investigated on the basis of engine parts study results generalization including the experience of prototypes investigation after high time operation with residual operating life evaluation.

Working out of methods for engine operating life exhaustion evaluation and engine availability provision and support include the following:
— organizational measures providing engine maximum operating hours;
— creation of algorithms and software for individual evaluation of every separate engine operating life exhaustion;
— development of methods for engine diagnostics and local repairs in the process of engine operating service.

Main engine repair techniques include the following:
— engine hot section parts repair heat treatments;
— technique of geometry repair of engine parts with erosion wear or foreign object damage defects;
— repair technique for turbine blades heat-resistant and ceramic thermo-barrier coatings;
— technique of compressor blades coating with erosion resistant layer;
— geometry restoration technique for worn-out contact surfaces and some other techniques.

The experience of "NK"-family engines building and operation with operating life up to 24000 hours and conversion of these engines for power industry needs with operating life up to 45000 hours confirms the validity of methodological approach used for engine reliability and operating life provision.

Fig. 1. Calculation of power turbine module natural frequency.

Fig. 2. Calculation of core engine LP spool rotor natural frequency, thermal and stress-strained state.

Fig. 3. Calculation of gas turbine engine load-carrying casing deformations.

Fig. 4. Calculated and experimental data Comparison results for turbine rotor wheel natural frequency.

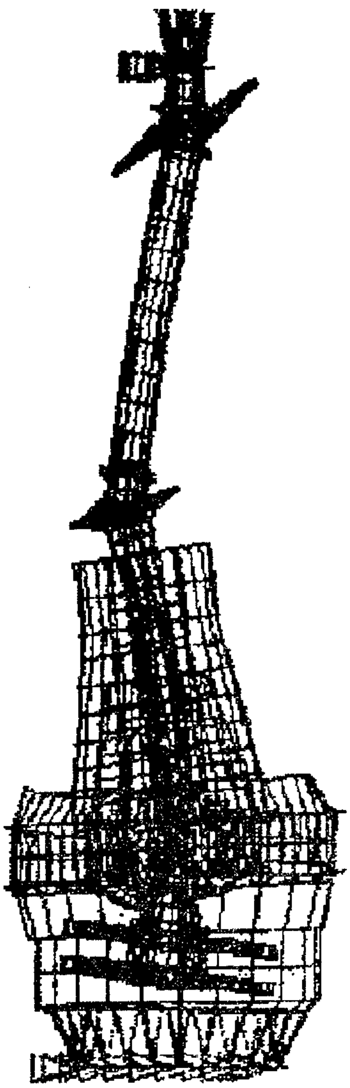

Figure 1.

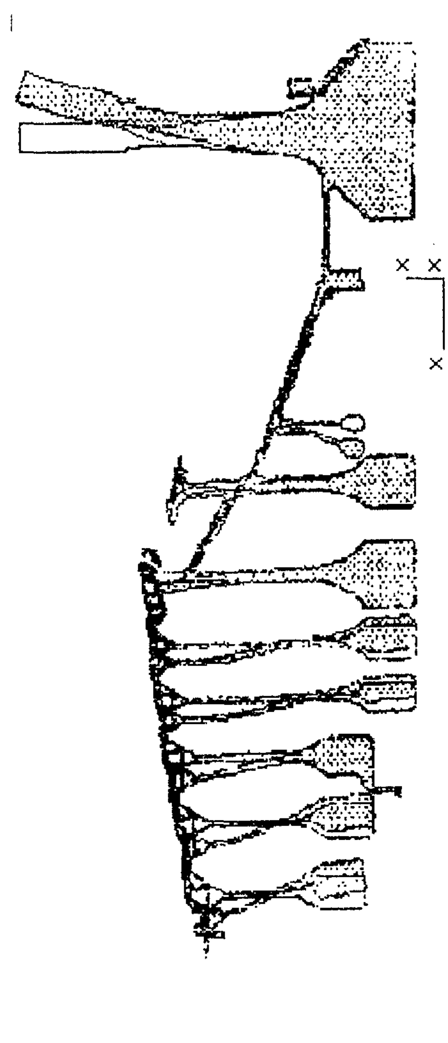

Figure 2.

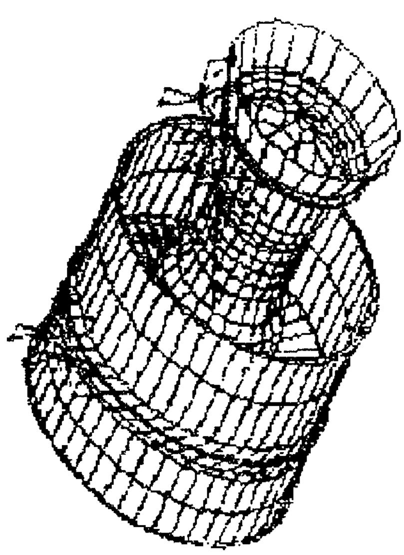

Figure 3.

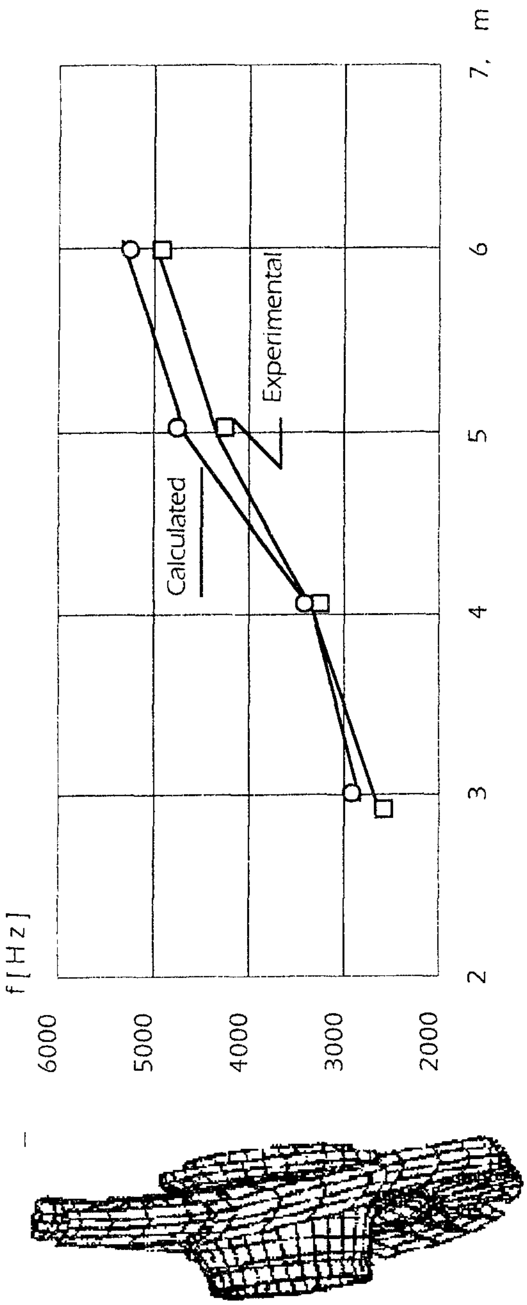
f [Hz]
6000
5000
4000
3000
2000
2
3
4
5
6
7, m
Calculated
Experimental

Figure 4.

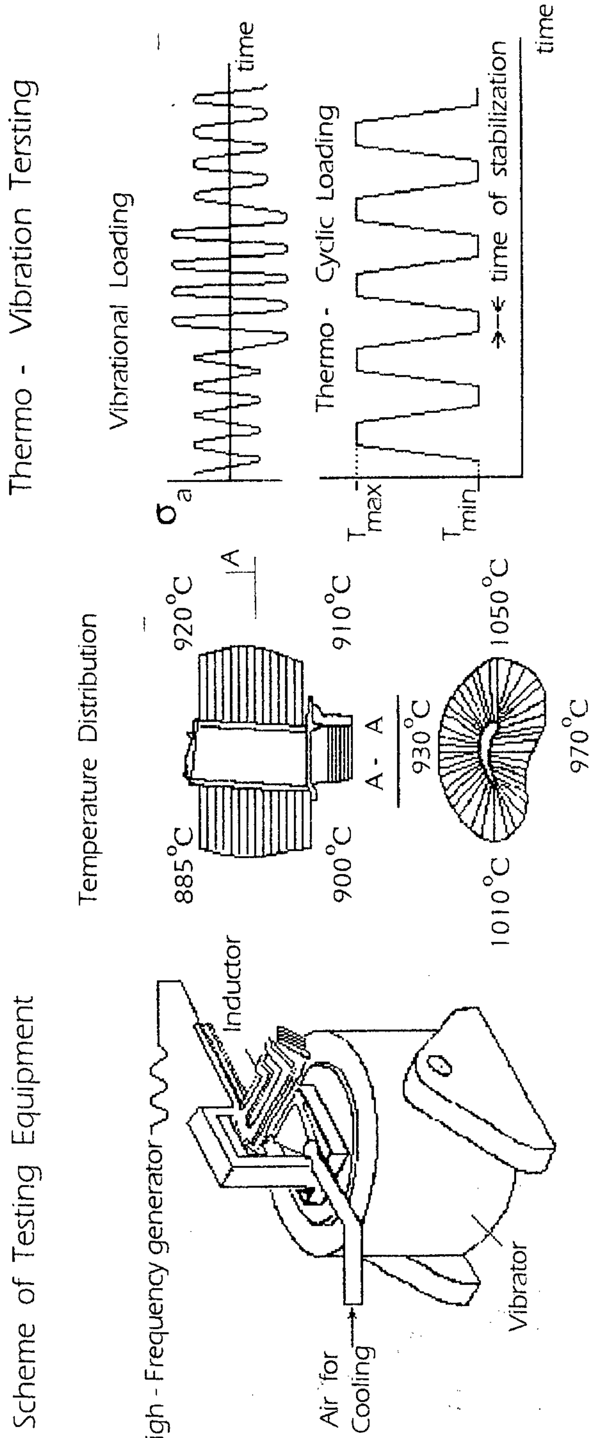
Scheme of Testing Equipment
High - Frequency generator
Inductor
Air for Cooling
Vibrator
Temperature Distribution
885°C
920°C
A
900°C
910°C
A - A
930°C
1010°C
1050°C
970°C
Thermo - Vibration Tersting
Vibrational Loading
σa
time
Thermo - Cyclic Loading
Tmax
Tmin
time of stabilization
time

Figure 5.

STRUCTURE	COMPOSTION	PURPOSE	THICKNESS	λ_{coat}, kkal/m*h*s
	Ceramics ZrO_2	heat conduction reduction	50... 150 mcm	1... 1,5
	Multilayer Ni+16...Cr+10Al+0.1... Ni Cr A 4 + 4 ZrO_2 layer thickness 0.5... 1 mcm	coat heat resistance	60... 80 mcm	4
	underlayer Ni+16Cr+2Al+0.054	blade surface coating crack imperneable barrier	30... 50 mcm	15
	blade 30 monocrystall	- - - -	Σ 170...220 mcm	2,5... 1,6

Figure 6.

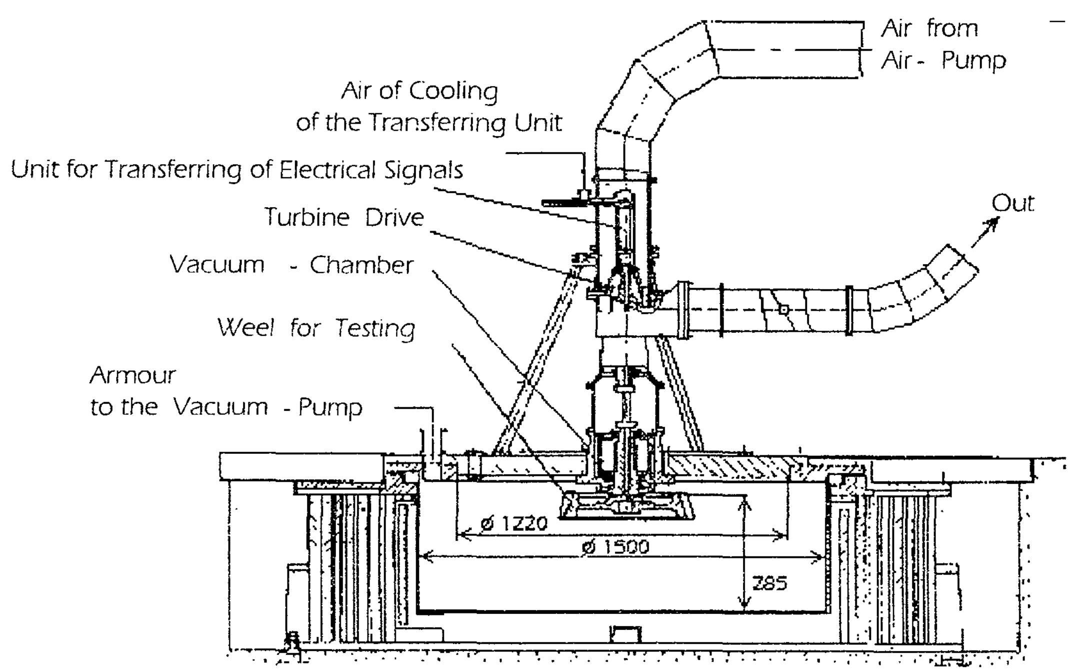

Figure 7.

TECNICAL CHARACTERISTICS

1. Power of Turbine Drive	N_{max} = 500 H P
2. Maximal Rotor Speed	n_{max} = 25000 rpm
3. Air Flowrate throw Turbine Drive	G = 6 kg / sec
4. Air Pressure	p = 5 kg / cm ^{2}
5. Thickness of Armour	30 X 7 = 210 mm
6. *Remaining Pressure in* Vacuum Chamber when Vacuum - Pump is Working	*p = 0.021 kg / cm*
7. Productivity of Vacuum - Pump	Q = 1480 m^3/ hour
8. Maximum Disk Temoerature	t_{max} = 750 ^oC
9. Power of Source of Heat	N = 150 kW

AIRCRAFT ENGINES CERTIFICATION

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Founded in 1956, the Omsk Engine Design Bureau (OEDB) started work on development of small-size turbine engines as early as 1957. At present the OEDB is a leading authority on small-size turbine engines in the Soviet engines production industry.

One of the main tasks of J-SC "OEDB" is the certification of the turboprop ТВД-20 engine An-3 and An-38 aircrafts. The certification oft the ТВД-20 engine was developed by J-SC "OEDB" in close cooperation with the CS "Kachestvo" in accordance with AP-23 requirements and confirmed by the Aviation Register of the Interstate Aviation Committee on May 1995. The ТВД-20 special features is that the engine can be installed on various types of aircrafts: An-3 (transport version), An-3 (agricultural version), An-38 (cargo and passenger version). It means that the development of over special 150 hours bong tests are considerably complicate flight performances and engine power ratings are also slightly different.

There is longitudinal section of the ТВД-20 engine on photo №1. This engine has axial/radial-flow compressor consisted of seven axial stages and single radial-flow stage. Annular combustion chamber is a semi-hoop type with rotary fuel nozzle. Fuel is supplied to rotary fuel along the fixed channels laid in the rear body of the compressor and labyrinth bush. Fuel combustion is provided by two starting units. Axial flow two-shaft turbine consists of two-stage compressor turbine and two-stage free turbine. Free turbine rotor drives the propeller by means of two-stage gearbox. The first stage of the gearbox is a simple gear transmission, the second stage is a planetary power is redistributed to the engine's driving box and to the unit box of the aircraft. The engine has automatic start from two electric starters ST-107B (An-3 aircraft) on starter-generator GS-12T, of the 3-a series, (An-38 aircraft). The engine is equipped with hydromechanical automatic control system (ACS). The An-38 aircraft may be also equipped with electronic-hydromechanical automatic control system.

Main data of the ТВД-20 engine

— take off equivalent power	1052 kW (1430 hp)
— take off propeller power	1011 kW (1375 hp)
— specific fuel consumption	0,306 kg/kW•h
— (0,225 kg/e.hp.h)	
— propeller driving shaft power in extremely condition	1103 kW (1500 hp)

(for the An-38 engine)

— -dry mass 285 kg

— -overall dimension (width, height, length) 850×845×1770 mm

At present the certification tests of ТВД-20 engine are conducted for 70%. In accordance with AP-21 the certification tests have been developed by J-SC "OEDB" and sent for agreement to CS "Kachestvo" and ARIAC.

In 1994 J-SC" OEDB" got down to development of the ТВД-20M engine modification with the AB-106 pushing propeller for a small cargo aircraft. M-102—"SARAS-DUET". At present time an order and amount of supplementary certification tests is under development and agreement with the Aviaregister(AR).

At the same rime when the TV-O-100 turboshaft engine for Ka-126 helicopter was certified by J-SC "OEDB", the certification basis was also developed in accordance with AP-33. The engine is under development since 1982. However financial difficulties stopped the developments until 1994. The developments of this engine was restarted only after the Russian Government Resolution № 878 November 17. 1992.

The ТВД-O-100 engine (Longitudinal section on photo № 2) is designed with free turbine. The free turbine power is transmitted to the main gearbox of the helicopter through the engine's driving gearbox which reduces the frequency of rotation down to 6000 R/min. Three-stage axial/radial-flow compressor, single-stage compressor's turbine and single-stage free turbine combined with reverse-flow looping combustion chamber make it possible to design compact construction with a small axial dimensions. Electric generator installed on the gearbox provides autonomous supply of electronic regulator of automatic control system. The gearbox has a free moving coupling separated the engine from the main gearbox of the helicopter in case of auto rotation. The gear box has also an installed gauge for measuring the twirl moment. The engine is equipped with electronic-hydromechanical automatic control system with the channel duplication equipment and hydromechanical manual control system used to restrict the unit parameters in case of failure of electronic regulator. The TV-O-100 engine has take off power 529 kW (720 hp) with specific fuel consumption up to 0,347 kg/kW•h (0,225 kg/e.hp.h).The specific fuel consumption is up to 0,401 kg/kW•h (0,295 kg/e.hp.h)(on the cruise regime with power of 338 kg (461 hp). Engine dry mass is up to 150 kg. Overall dimensions (width, height, length)—780×735×1275 mm.

The design of the engine has been considerably changes in the last years in the result of improvement (The income directional unit and the blades corner regulation system of the compressor have been excluded from the construction of the engine. The construction of the first stage of the compressor rotor has been changed and the widechord blade made in concert with the disk has been applied. The visual control system of the rotor and direct blades of the compressor has been also developed. The turbine disk of the compressor and the TK exhaust nozzle diaphragm have been made stronger; the construction of the TC rotor blade has been also changed. The bodies of the compressor have been made stronger, etc.) that made it possible to increase reliability of the engine and to improve its performances.

The ТВД-10В turboprop engine installed on An-28 airplane has been certificated in accordance with NLGS-2. In the result the flight certificated has been confirmed in 1985. There is longitudinal section of the engine on photo № 3. The construction of the engine has the following design features: axial/radial-flow compressor, annular combustion chamber with rotary fuel nozzle, axial-flow two-stage uncooled compressor turbine, axial-flow single-stage free turbine with the deflection of the power into the rear part, the transmission of the power on the gearbox of the propeller is provided through the high speed gearbox and external transmission. Takeoff propeller power — 960 hp, takeoff equivalent power — 1025 hp, specific fuel consumption is up to 0,255 kg(e.hp.h). Dry mass of the engine — 230 kg, overall dimensions (width, height, length) — 555 × 900 × 2060 mm. For the last two years J-SC "OEDB" in cooperation with "ROKS-AERO" and MAPO of Dementiev were carrying out the works on ТВД-10В attachment to T-101 "Grach" airplane. As a result flight tests have been started on December 1994. To certificate T-101 "Grach" airplane J-SC "OEDB" should carry out All-round tests of the engine to get the Engine type Certificate in accordance with the present rules AP-21.

The auxiliary power unit VSU-10 developed by J-S "OEDB" for Il-86 and Il-96-300 aircrafts was certificated in accordance with chapter 9 NLGS -3, as a result of this the flight Certificate was confirmed on December 1992. VGTD VSU-10 is developed on the base of gas generator of the ТВД-20 engine (photo №4). The power of the free turbine of the gas generator is transmitted to the additional axial-flow two-loop eight stage compressor providing air supply for a start of the main engines or into the air-condition system of the aircraft with the parameters:

air flow rate	3,5 kg/s
total pressure	4,75 kg/cm
temperature	217 C, max.

The electric generator installed on the gearbox pivots electrical power taken off parameters — 40 kW; frequency — 400 Hz , voltage — 208/120 V.

It is possible to install electric generator on the VSU-10 engine with electrical power take off — 60 kW.

Mass in state of delivery — 500 kg.

Overall dimensions (width, height, length) — 1035×1264×2394 mm

In connection with VSU-10 installation on Il-96T(M) it was necessary to carry out works for confirmation of the engine conformity to American standards TSO-C77a. J-SC "OEDB" in cooperation with

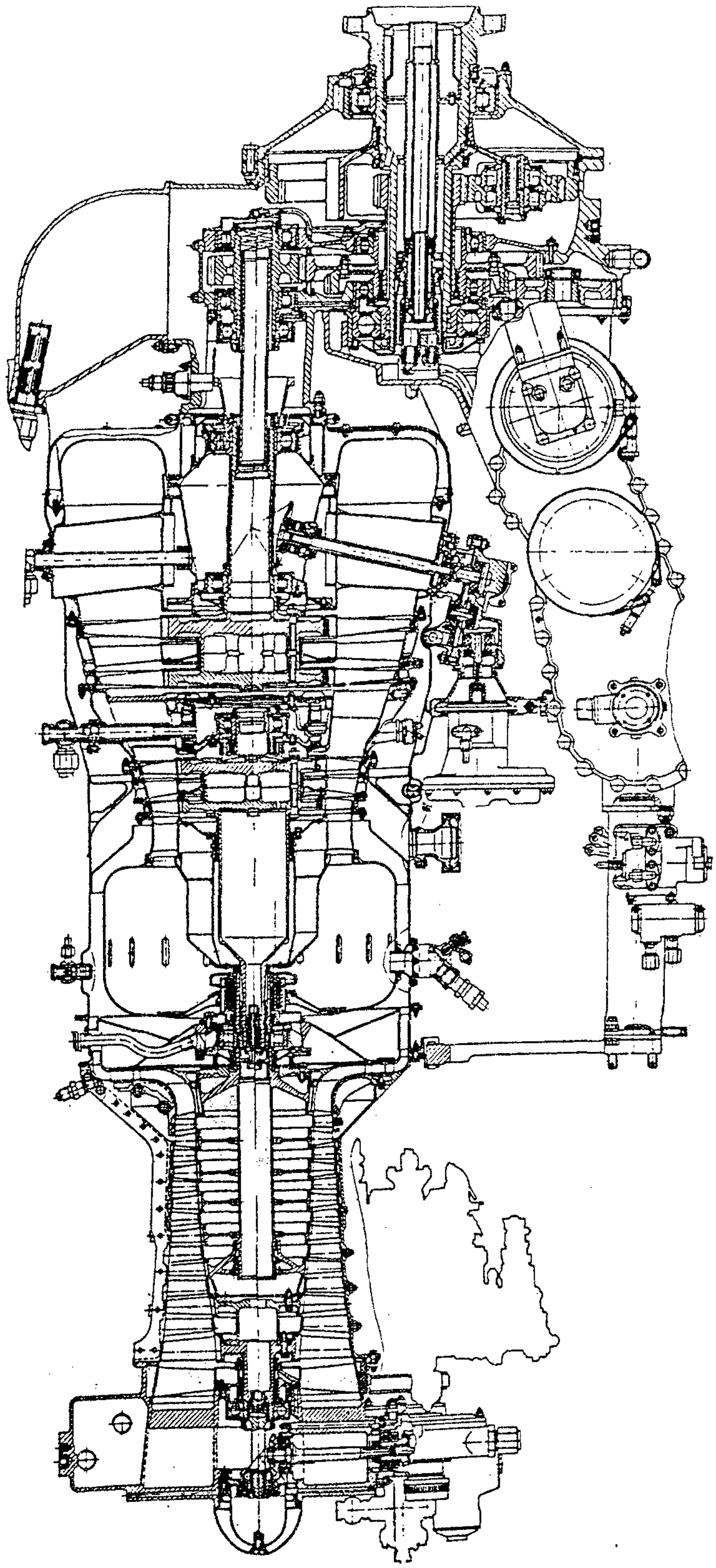
CIAM worked out the certification basis taking into account the TSO-C77a requirements. The certification plan has been worked out and discussed with FAA USA. However in accordance with the financial difficulties there have been taken the decision to certificate VSU-10 as a component of Il-96T(M) aircraft. To put VSU-10 in order to TSO-C77a standards it became necessary to improve the design of some units of this engine. The main improvements are connected with localization of fragments of the engine s rotor and with fire protection of the oil tank installed on this engine. The first requirement was carried cut by installing non pierced bond ring around the turbine unit. The second requirement was carried out by applying fire protective foam cover of the oil tank outside surfaces. In the aim of improvement of the design and certification tests of the engines there are following test-benches and installations in the I-SC "OEDB":

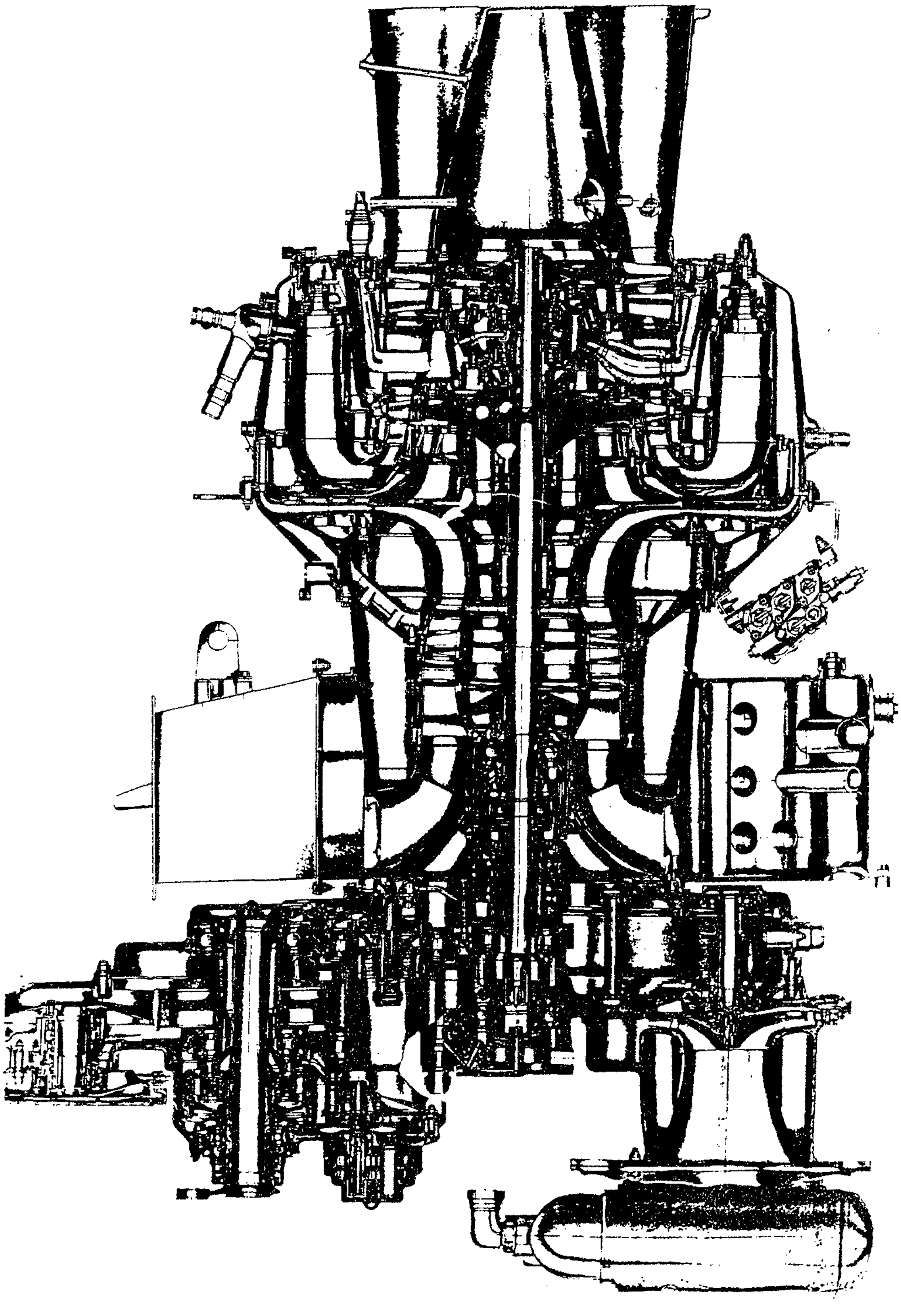
— three propeller test-benches for experimental tests of turboprop engines with airplane propellers in the ground conditions;
— test-bench for experimental tests of the auxiliary turbine engine VSU-10 with airplane incoming system in the ground conditions.
— hydrobrake test-bench for experimental and certification tests in the ground conditions with air heating up to +50 C.
— two electrobrake test-benches for experimental tests of turboshaft engines with air heating up to +50 C.

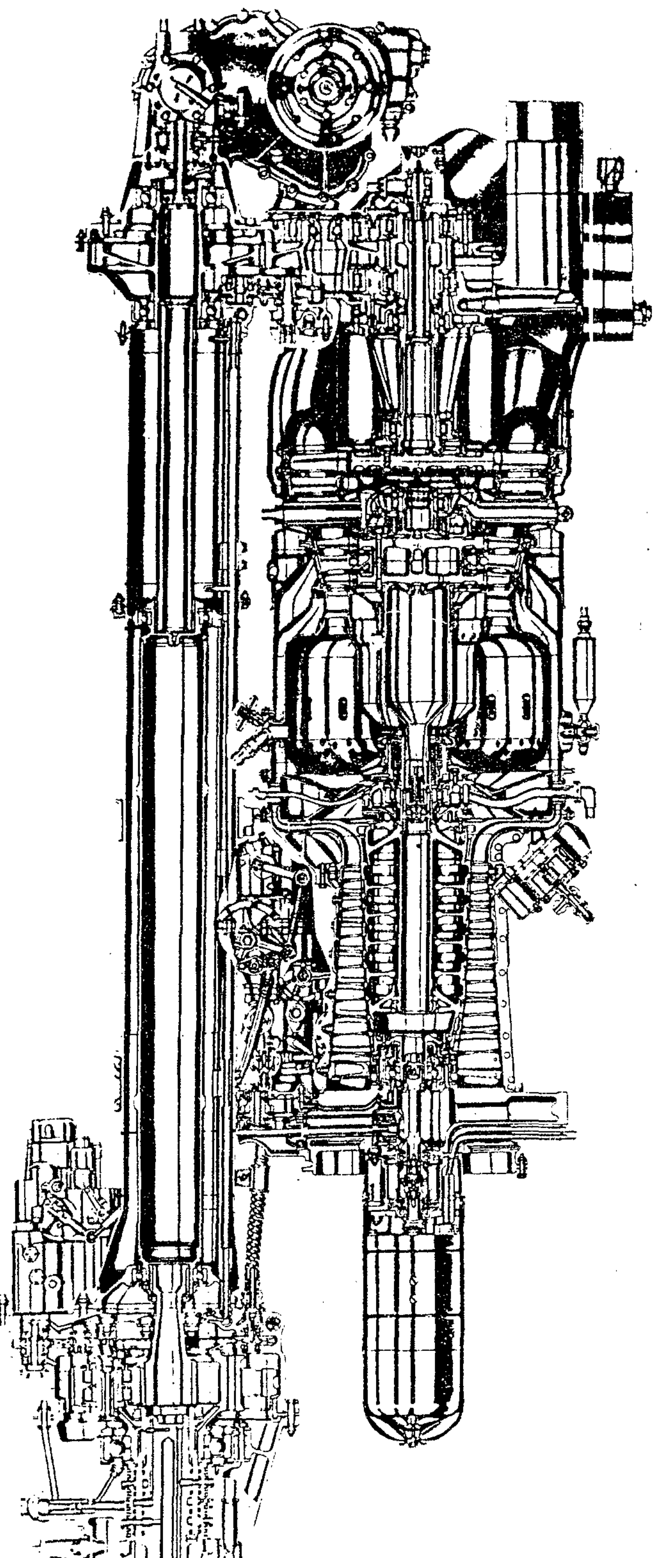
All the test-benches are covered equipped with the absorb and exhaust shafts (propeller test-benches with horizontal shafts), technological start and engine control system, and parameters measurement systems. The observer compartment is equipped with register and control panels. Test-benches are connected with computers which are used for automatic registration and estimating in the rate of experiment. To test engine units "The test experimental complex" of I-SC "OEDB" has aspeual equipment for testing compressors, combustion chambers, butt-end contact compressions, overtake couplings with eccentric rollers, scavenge nozzle systems, dynamic tests of compressor and turbine blades, control and ad just electronic, hydraulic and fuel systems.

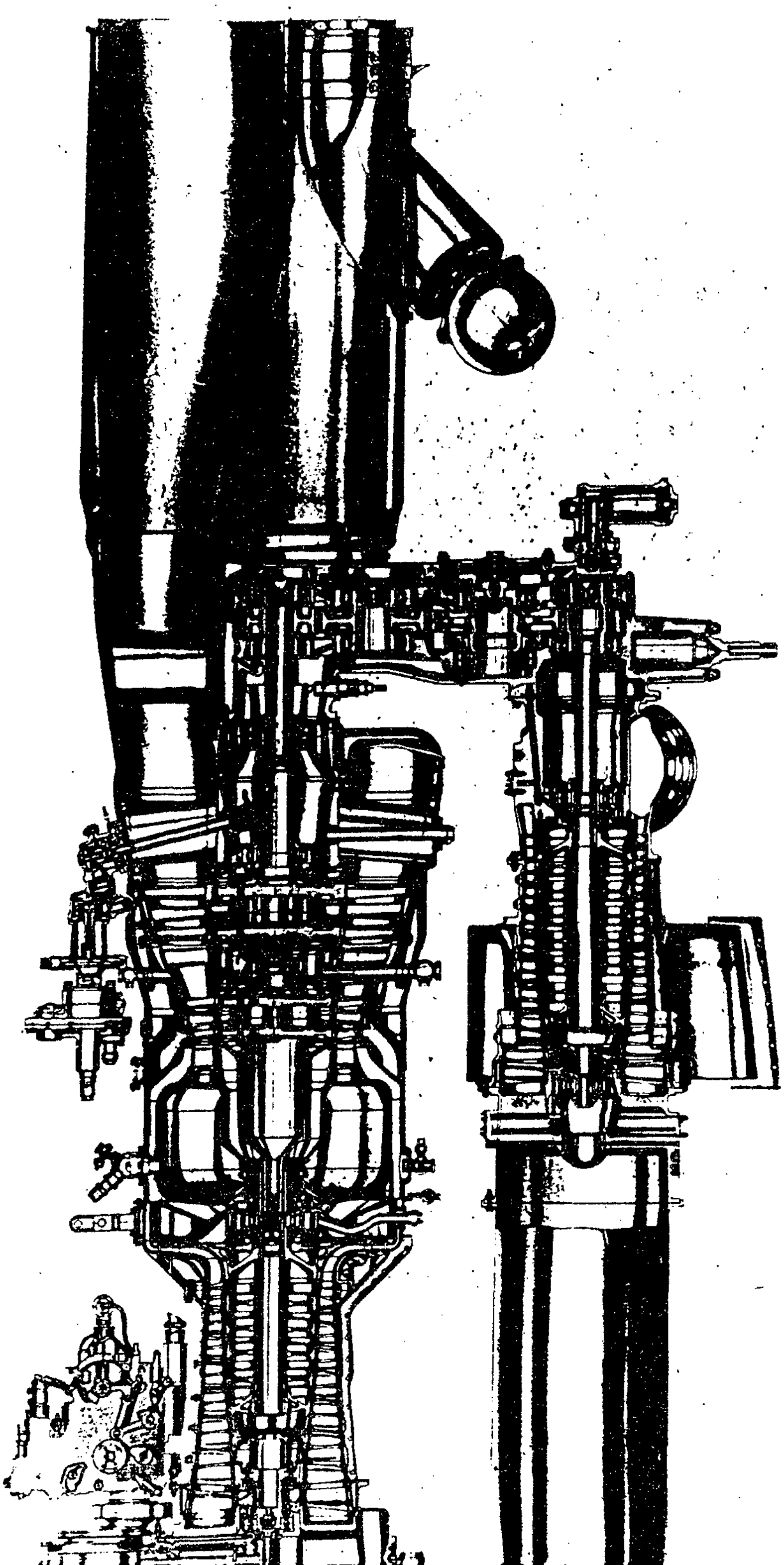
All the test-benches and installations make it possible to combine checks and tests determined by the present air rules except altitude, speed and climate imitations.

J-SC "OEDB" has to conduct testing of altitude and speed characteristics as well as to start and check engine s capacity for work in the possible ranges of speed, altitude and temperature using the thermal low-pressure chambers of the central institute of Aircraft Engine Production.









ELABORATION OF THE LIFE ESTABLISHMENT METHODS FOR ENGINES AND THEIR MAIN PARTS

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CIAM, Moscow, Russia

In recent years an aircraft engine life came to be given in our home aero-engine manufacturing at last the attention identical to that placed to other basic characteristics (such as thrust, specific weight, fuel flow, etc.).

Previously a new engine has been put into service with the achieved and demonstrated basic characteristics and small initial life of the order of 500 — 1000 hours, and subsequently while in service there have been realized the prolonged and expensive development works to attain a full declared life. Such approach to the life establishment and extension matter is presented in life normative documents and it has been used up to now.

Under new economic conditions and in the situation of aggravated competition between the aircraft engine manufacturing companies it has taken the elaboration of new methods to bring the engines to the declared lives at an accelerated pace that could take into account the gained home and foreign experiences.

Our country and foreign approaches to a life establishment matter differ greatly. In the home practice there are such notions as "life of the engine as a whole (specified between overhauls, first overhaul) and "life of main engine parts". Service is accomplished most often with the fixed lives of engine and its main parts, and an on-condition service is realized within the limits of fixed engine lives.

Abroad the "life of the engine as a whole" notion is unavailable, there are set up the limitations on lives of critical (main) parts. Engine comes into service, as a rule, already having large declared main parts lives and servicing is generally accomplished on-condition from the very beginning.

Upgrading of home methods of aircraft engine life management is underway in the following three directions:

- decrease of amount of testing of a full-scale engine while establishing the lives on retention of reliability of live estimations,
- working out of the experimental-calculational methods for establishing lives of main parts;
- upgrading of the on-condition service methods.

To date there have been worked out three strategies of live management:

— live establishment based on endurance tests of a full-scale engine and its main parts and servicing with the fixed lives of engine and main parts (for the aged production engines having the developed diagnostics system),
— life establishment based on endurance of main parts and on-condition engine servicing (for prototype and production engines prepared for servicing),
— calculational-experimental substantiation of main parts lives and condition monitoring engine servicing (for advanced engines and implementation of certain elements of this strategy for the engines recently came into service.

According to the first strategy, life is confirmed by tests of a full-scale engine with 20% margin on a number of test cycles and by tests of main parts with 2.5 — 3.0 margins on a number of test cycles depending on a quantity of the tested part specimens. Such testing types have been used previously when establishing the initial lives and in further their extending. What is new now, the emphasis is on the tests of main parts and the amount of the tests of the engine as a whole is decreased.

The technique for life establishment is changed significantly in case of on-condition servicing. If formerly every step on life extension called for a confirmation with the use of endurance tests of a full-scale engine, then currently the conclusion on a possibility to extend life of the whole engine fleet is made based on positive results of disassembly and flaw detection of the leader engines that have run a certain portion of the life.

The second strategy calls for testing main parts, and engine as a whole is tested only for a small portion of full declared life to provide the demonstration of attaining the guaranteed life.

A change-over to calculational-experimental method of establishment of main engine parts lives offers the greatest promise. The complex of works to be done to establish and extend life by a calculation-experimental method involves an analysis of operating conditions and flight cycle determination and refinement; an analysis of thermal-stress state; an accumulation of data bank on properties of used materials; a determination of plausible durability value and a life establishment.

Every third strategy stage includes a large amount of works which have been realized in any way in the home aircraft engine manufacturing. The suggested methodology generalizes these works and collects them in a unified strategy.

For instance, both engine durability and weight characteristics depend greatly on a proper substantiation of generalized flight engine cycle. The introduction of the life designing methods enables to seek for trade-offs between life and the remaining engine characteristics, but not having properly specified given data, the results may be wrong.

Much consideration must be given to analysis of thermal and stress-strain state and experimental confirmation of calculational results both by the way of conducting direct experiments and when applying the calculational methods to the engines being in service. The application of this calculational-experimental method is impossible without the up-to-date calculational methods being mastered in Design Bureaus.

The creation of data bank on materials is the most expensive in the calculational-experimental method. According to the foreign findings the acquisition of bank on one material costs more than 1 million dollars. Such high cost results from the need to take into account the influence of different factors made on a durability, such as stress concentration, manufacturing process, dissimilar types of damage (vibration, creep). In addition to the obtaining of the low-cycle fatigue curves for specimens whose form also makes an effect on test results, it is required to generalize the test results of actual parts.

The necessary data are available in an odd form in Design Bureaus, scientific-research institutes, plants. These data should be integrated.

The calculational- experimental method has been applied to a number of the PS-90A engine parts. The analysis showed that the application of new approach to the life establishment at early engine development stages enables to reveal possible life limitations without conducting the expensive tests.

EXPERIENCE OF DIGITAL DATA PROCESSING IN VIBRODIAGNOSTIC INVESTIGATIONS OF GAS-TURBINE ENGINES

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The modern experience of quantity production and assurance of service reliability of the aircraft gas-turbine engines, especially when they are operated on the 'on condition' basis, confirms actuality and necessity of wide use of various diagnostics means. The preference is given as a rule to methods and means of diagnostics which do not require disassembling engines, and the vibrodiagnostic is a typical example of these methods. The trustworthiness of diagnosis depends to a significant extent on the following factors:

- — high quality and qualified performance of a complex of researches at the stage of elaboration and verification of a method;
- — use of up-to-date diagnostic equipment permitting use of digital methods of signal processing.

The general requirement for methods and equipment is the possibility to receive and process during diagnostic process large volumes of diagnostic information, this also improves the reliability of diagnostic.

For application at the development stage methods of vibrodiagnostic of gas-turbine engines (GTE) and for implementation of these methods in service, the Scientific and Production Enterprise 'MERA', with methodological participation of the 'Motor Sich' JSC, developed and implemented into production the system of digital data processing for vibrodiagnostic investigations in stationary and field conditions.

The system consists of (ref. fig. 1):

- — a set of vibration transducers and unit of matching analog measured signals or other source of analog information;
- — an analog-to-digital conversion unit featuring a digital data processor;
- — a stationary or portable IBM-compatible computer;
- — SPP (signal processing package) software.

The system provides for the input, analysis and data presentation in real time conditions and with storage in the working memory or on the hard disk of personal computer.

The main working menu (ref. fig. 2) of the SPP package contains the modes of signal input, archiving and loading, visualization, mode of algorithms and their adjustment, mode of documentation and input of comments.

The system provides for 2-dimensional and 3-dimensional presentation of the analysis results.

The most typical examples are given below to demonstrate the application of the system for solving the problems encountered during vibrodiadnostic investigations on the full-scale gas-turbine engines.

The system permits to use both through- and indirect calibration of measuring channels.

Widely used during vibrodiagnostic investigations are 3-dimensional spectrum diagrams of vibrospeeds , measured at turbine engine rotor bearings, scanned in time or in main rotor rotational speed (ref. fig. 3—4).

These diagrams were used for investigations of vibration characteristics with the aim to identify the source of high vibrations and the conditions of manifestation of higher level resonance oscillations.

Realization of signals in time and their momentary and averaged auto spectra, as well as reciprocal and coherent analysis are used for revealing the diagnostic criteria when investigating diagnostically stressed conditions of turbine engine parts (ref. fig. 5—7). For example, when solving this problem for the body of accessory gearbox gear, the reciprocal analysis was used for natural oscillations of the gear caused by wide band pulse excitation, vibrations measured at gearbox casing as well as dynamic deformation of the gear body during operation of the gas-turbine engine.
The main task was to find diagnostic criteria in a vibration signal (ref. fig. 8).

The method selected for this purpose — construction of 2-dimensional reciprocal spectra and coherence function — made it possible to define, evaluate and visualize the coherent components of the deformation spectra and vibration spectra. The information obtained in this way was used for selection and evaluation of the information capacity of diagnostic criteria.

For mastering the technique of making diagnosis for slackening of tightening of high speed GTE rotor flanges, the method was used when the analysis was performed of vibration magnetograms with measurements taken at external casings of supports of running engine with proper and improper conditions of the rotor joint tightening (ref. fig. 9—10).

The essence of the method consisted in search of distinguishing features with utilization of noncoherent function, evaluation of non-reciprocal share of spectrum and noncoherent power. The obtained results made it possible to assess information capacity of the distinguishing features end select the optimum one to serve as the condition diagnostic criterion.

During the diagnostic investigations, the methods based on evaluation of parameters of the oscillating systems dynamic characte ristics are widely used, and these parameters are used as diagnostic criteria. For example, for evaluation of the predicted level of dynamic stress in GTE pipelines, the capabilities of SPP system are used for determining natural oscillation frequencies and dissipation characteristics (ref. fig. 11) and obtained information is later evaluated with the use of mathematical models. This provides the opportunity in many cases to reduce the scope of expensive and time consuming extensometric investigations. At present, at our Enterprise, in compliance with TsIAM recommendations, the system of accumulation of forestalling vibrodiagnostic information is introduced for the D-36/-136/-18T engines, this system is conditionally called 'vibrocertification'. The accumulation is effected at data base which is expected to be used for prompt performance of engine diagnostics with regard to newly developing defects.

The vibrocertification is realized on the basis of utilization of the above mentioned SPP software and hardware complex.

Mastering and utilization of the data digital processing system will permit to solve, in short time and at higher qualitative level, a number of pressing problems for assurance of operational reliability of the gas-turbine engines by using the vibrodiagnostic methods.

The typical example of wide use of potentialities of the digital reception and processing of data is the method, developed in cooperation with the TsIAM, of vibrodiagnostic of chipping of the D-18 T engine HPT support assembly bearing race in operation. In fig. 12 the appearance of defect is presented.

The technique provides for carrying out the following operations:

— manual cranking of HP rotor with the aid of special appliance;
— measuring, digitizing and storing in the equipment memory the vibration signal from the transducer fitted to the support assembly casing;
— processing of the signal based on use of digital filtering, peak detection, spectrum analysis and statistical analysis;
— calculation of values of four diagnostic symptoms connected with conditions of bearing races (in fig. 13 the results of signal processing and calculation of diagnostic symptoms are presented);
— operation of the expert system accompanied by issue of the resultant diagnostic statement containing information about the object for which diagnostics is performed, condition of the bearing and recommendations on actions to be taken with respect to the engine.

The diagnostic complex operates in automatic and dialog modes, which permits to use personnel without special training for performing investigations.

However, when elaborating the vibrodiagnostic methods problems arise of proper selection or rejection of symptoms, definition of decisive rule when forming multifactor diagnostic criterion, substantiation of selection of maximum permissible value of criterion, that is standard value, creation of continuously adaptable standard-mask which has the form of a 3D view complex surface permitting to monitor and evaluate vibration conditions of an engine within the whole range of operating conditions and wide frequency range.

A standard set of available software and hardware does not permit yet to solve the above enumerated problems effectively and at high technical and scientific level.

Further development of the software for solving the set tasks would give the users the powerful facilities for prompt and qualitative solution of vibrodiagnostic problems.

Figure 1. The Contens of Signal Procesing System "SSP"

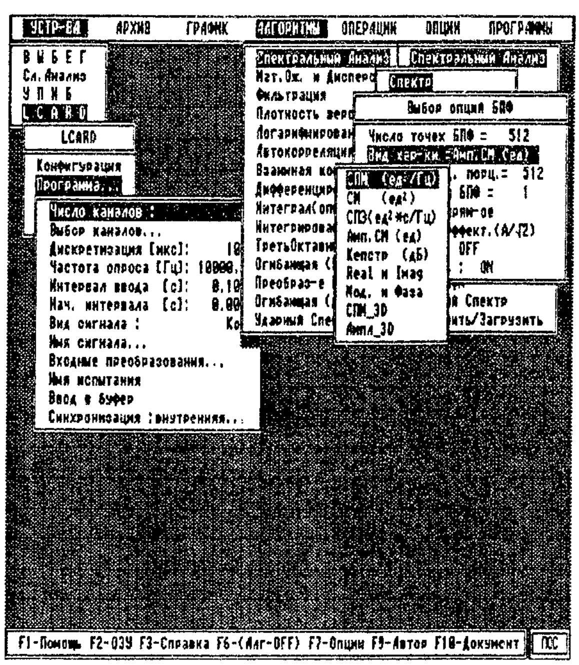

Figure 2. Main Working Menu of Processing System "SSP"

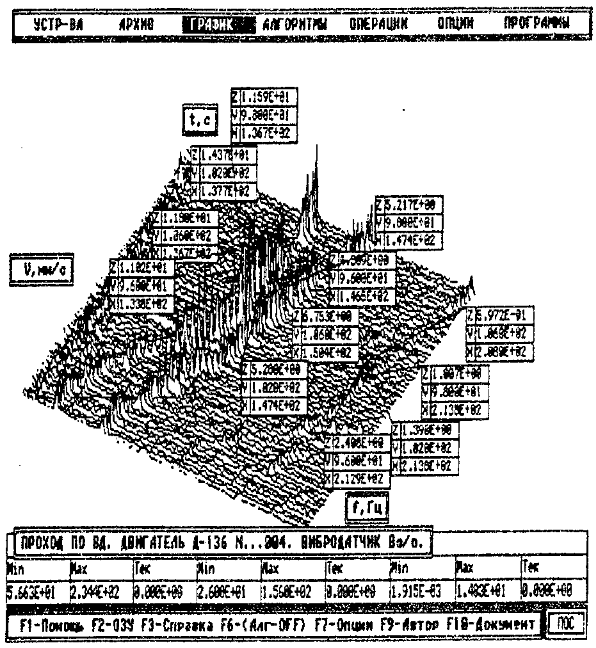

Figure 3.

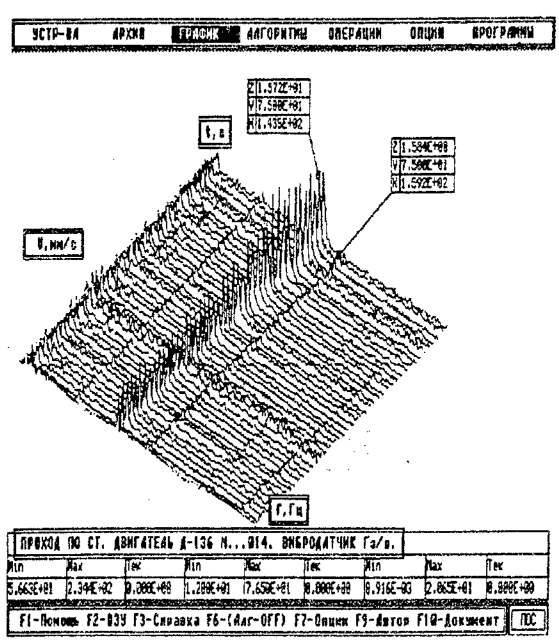

Figure 4.

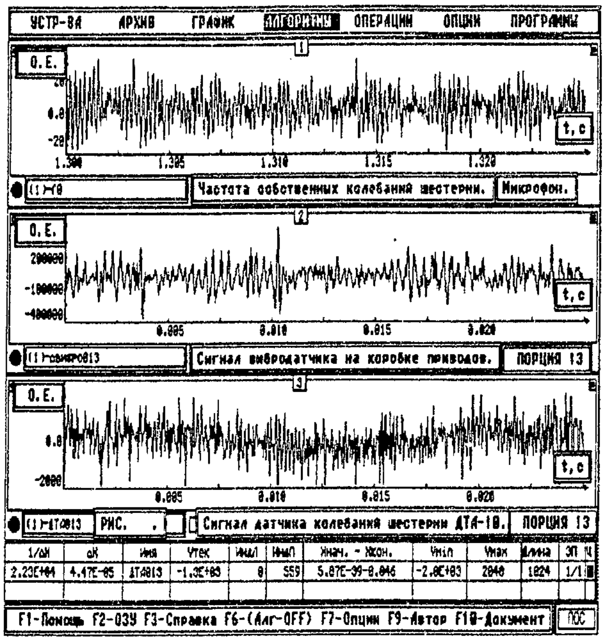

Figure 5.

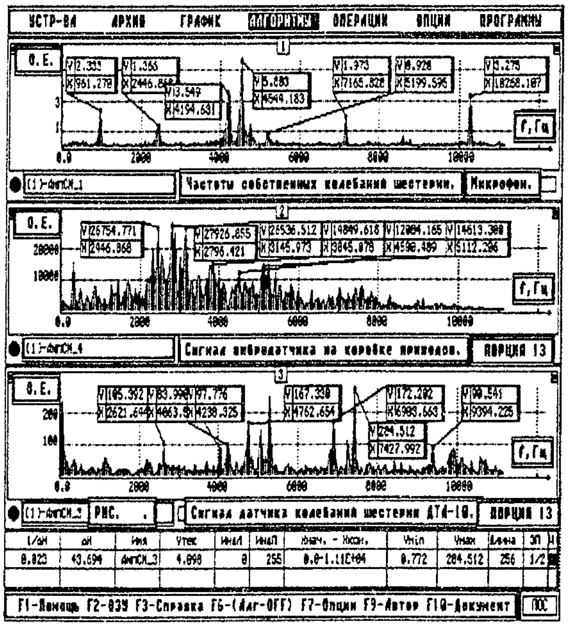

Figure 6.

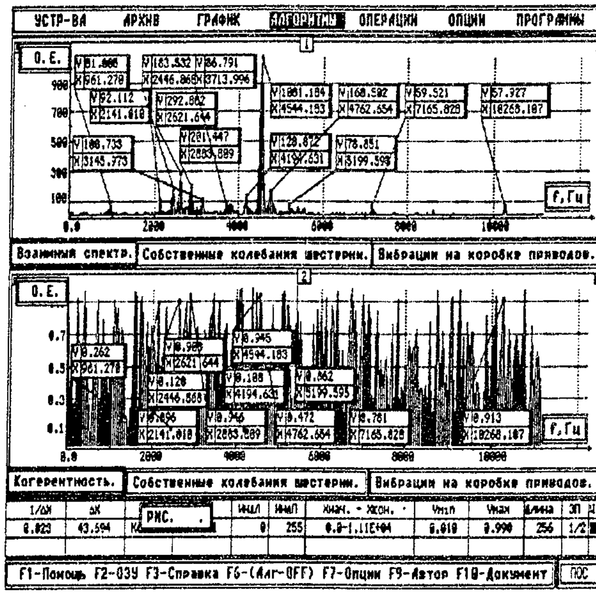

Figure 7.

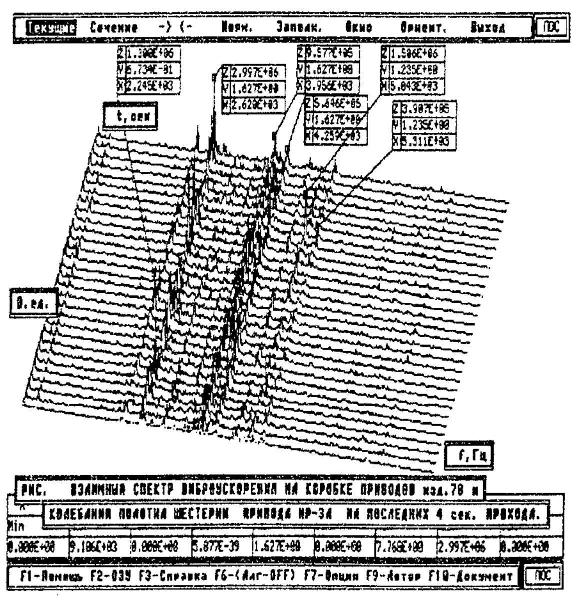

Figure 8.

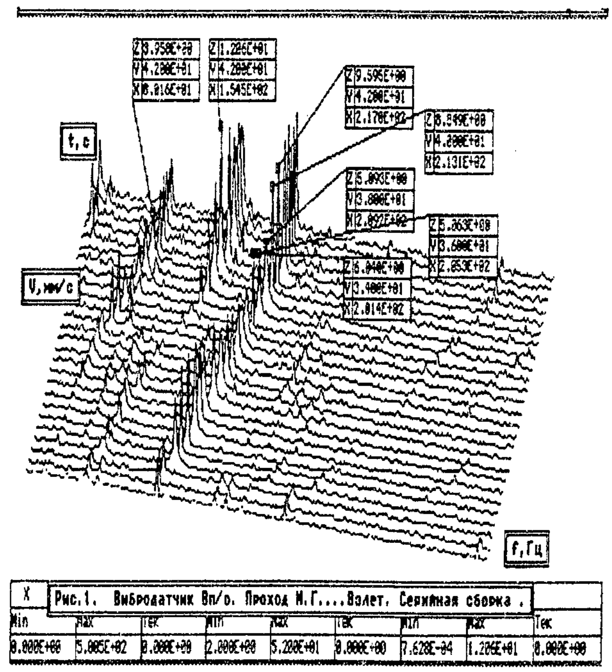

Figure 9.

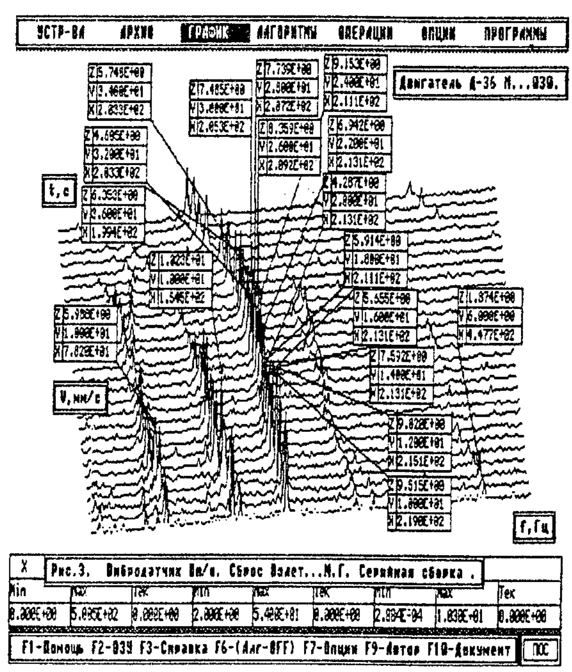

Figure 10.

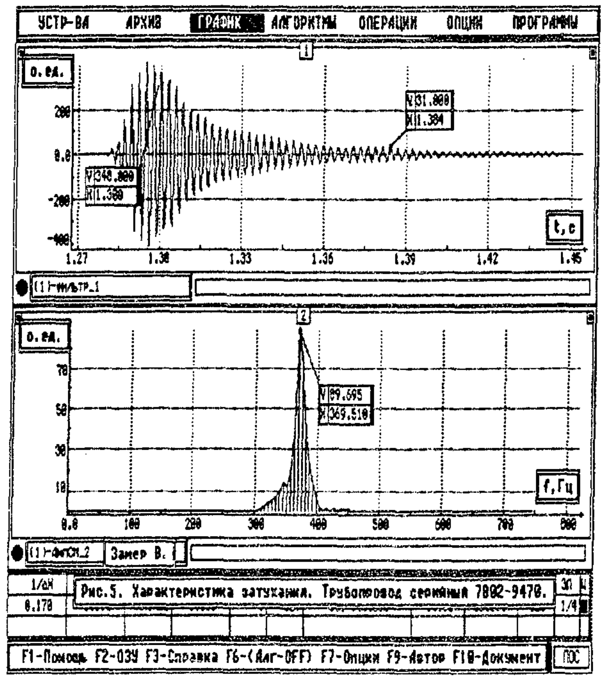

Figure 11.

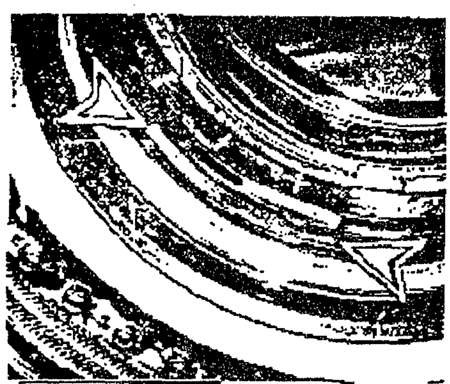

Figure 12. Example of daweged bearing

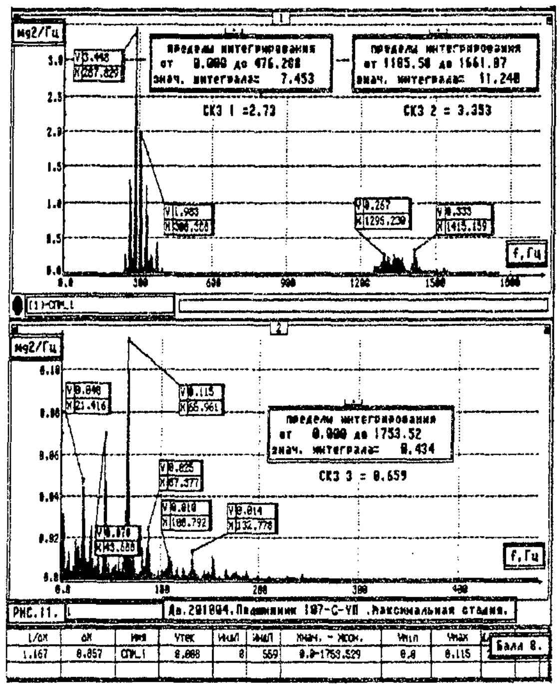

Figure 13. Diagnostic symptoms of daweged bearing. Maximum stage.

CERTIFICATION OF MATERIALS USED FOR AVIATION TURBINE ENGINE MAJOR PARTS AND REQUIREMENTS TO DESIGN STRENGTH CHARACTERISTICS

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The level of gas turbine engine safety and reliability essentially defends on the quality of materials used for engine major parts.

The certification of such materials is the important part of airworthiness evaluation of the engine and aircraft on the whole.

The material certification is the confirmation of material worthiness for part operation in the engine (at specified temperatures, loads, required life). The procedure of certification is required by "Regulation of Aviation Material Certification", developed by Certification Center "Material" which issues a certification of serial material or semiproduct approval and possibility of its application in aviation turbine engine production.

The following prolonged stages are proceeded this last one:

1) material creation based on the requirements formulated at the development of "aircraft-engine" concept, engine features determination, evaluation of its parameters, structural and strength analysis of major parts. The scheme of cooperation of the research centers at the development of the requirements for new materials that results in the complex target programme is shown in fig. 1. The major part multivariant strength calculations which are conducted to determine the required design strength are performed as a rule in CIAM using the automated systems functioning in a dialogue mode and having an information search system that uses the data base on the available material characteristics. The system includes the logic's of different levels for strength calculation and allows to establish the requirement to properties of new materials that satisfy the criteria

$$K_{\text{min parts}} \geq K_{\text{required}}$$

where $K_{\text{min}}, K_{\text{required}}$ — minimum and required strength margins accordingly,

2) the complex investigation of strength characteristics of the created material necessary for its operation being used in a part. The accumulation of statistical data on the investigated

properties, determination of their minimum values and development of specifications on material delivery by different manufacturers, establishment of design strength characteristics. The material passportization based on the results of the performed investigations.

On the whole VIAM is responsible for the fulfillment of this stage. CIAM takes part in the expertise of the characteristics, evaluation of values for providing the required strength margins;

3) the evaluation of technological manufacture capabilities of required geometry parts made of selected materials and confirmation of properties (on the samples cut out of the parts) specified at the passportization stage. This stage is carried by the material creator at close cooperation with technological institutes, billet and engine manufacturers;

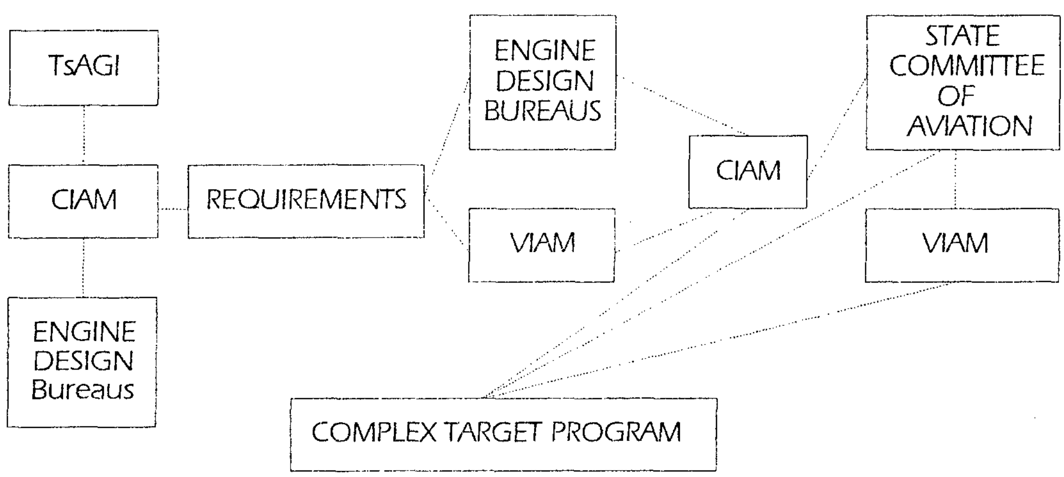

Figure 1.

4) the investigation of full-scale parts made of the selected materials at special strength test (spin and cyclic tests of discs, fatigue tests of blades, etc.). These tests are conducted at experimental design bureaus and engine manufacture plants according to the developed in CIAM Manuals and Standards. In some cases these tests are conducted in CIAM.

The mentioned above stages are the main ones which form a system of material quality provision. The results of the investigations performed according to these stages are considered during certification and (if they are satisfactory) allow to issue a certificate on the developed material.

However certificate receipt does not mean the accomplishment of activities on control and provision of material stability and quality. During serial production the continuous monitoring of material properties according to the indexes specified in the specification on delivery is performed. Besides the full-scale part periodic tests (e.g. fatigue tests of blades) and detailed investigations of properties (cutting into samples: discs, combuster casings) are carried out.

The results of the such investigations are subjected to statistical analysis that allows to monitor changes of the monitoring properties and compare them with the changes of the technological process parameters of a concrete production period. The statistical analysis of the check test results comes to drawing of monitoring characteristics distribution curves, determination of mean values ($\overline{X}$) and dispersion (S_X) and comparison of these parameters with those accepted at certification. The distribution parameters allow to evaluate the probability of failure using the value of strength characteristic specified in the specification.

The example of the such evaluation based on the statistical data on strength and plasticity characteristics, obtained at inspections of the titanium alloys VT3-1 and VT8 disc forgings performed at the engine-manufacturer is given in fig. 2 and Table 1. The curves of distribution of the ultimate tensile strength (UTS), elongation (δ), reduction of area (ψ) and shock bending (KCU) are presented in fig. 2. The values of these characteristics acceptable according to the alloy specification (X_{spec}) are given at the bottom.

Table 1

Alloy	Semiproduct typne	Property	$\overline{X}$	X_{spec}	$\frac{\overline{X}-X_{TY}}{S_X}$	Number of samples
VT-1	Disc forging	UTS,kg/mm	108,2	>96	5,3	69
		δ, %	15,8	>9	4,0	69
		ψ, %	36,2	>22	3,3	68
		KCU,kgm/cm	5,7	>3	5,4	69
VT8	Disc forging Vendor 1	UTS,kg/mm	105,7	>98	2,6	169
		δ, %	16,0	>10	3,3	168
		ψ, %	46,5	>25	5,1	167
		KCU,kgm/cm	5,3	>3,5	2,6	167
	Disc forging Vendor 2	UTS,kg/mm	109,7	>98	5,6	235
		δ, %	14,3	>10	2,3	248
		ψ, %	33,3	>25	2,7	236
		KCU,kgm/cm	4,9	>3,5	2,0	232

They are minimum allowable values that are considered as design properties at part strength evaluation. The relationship of minimum allowable (X_{spec}) and mean ($\overline{X}$) values of the considered characteristics are shown in Table 1, where $(\overline{X} - X_{TY}) / S_X$ is tolerant factor K of the equation $X_{min} = \overline{X} - KS_X$.

According to the requirements of the foreign regulations [1] for the group A parts the value of K is equal to 3. As it can be seen in Table 1 the values of K for the considered titanium alloys with the acceptable according to the specification characteristics are equal to 2...5,6; (in most cases $K > 3$) that affirm the possibility of using design strength characteristics of engine major parts in accordance with the foreign methodology.

The statistical evaluation similar to the mentioned above for the titanium alloys which are constantly performed during serial production allows to monitor material production stability based on the analysis of the parameters distribution $(\overline{X})$ and (S_X). If sudden value deviation take place the reasons must be revealed and in case of significant changes in the serial production a decision of recertification can be made.

Thus the conception of engine parts material is a capacious continuos process which establishes material condition and properties at the moment of material transfer to serial production, type, volume and quality of conducted investigations, acceptability of approved design characteristics, condition of control system and material quality, provision during serial production. During serial production the certification authorities together with the aviation research institutes (VIAM, CIAM) continue to monitor the stability of produced certified material based on the evaluations of values $(\overline{X})$ and (S_X) and hence the possibility of using in strength calculation the minimum allowable characteristics approved at certification.

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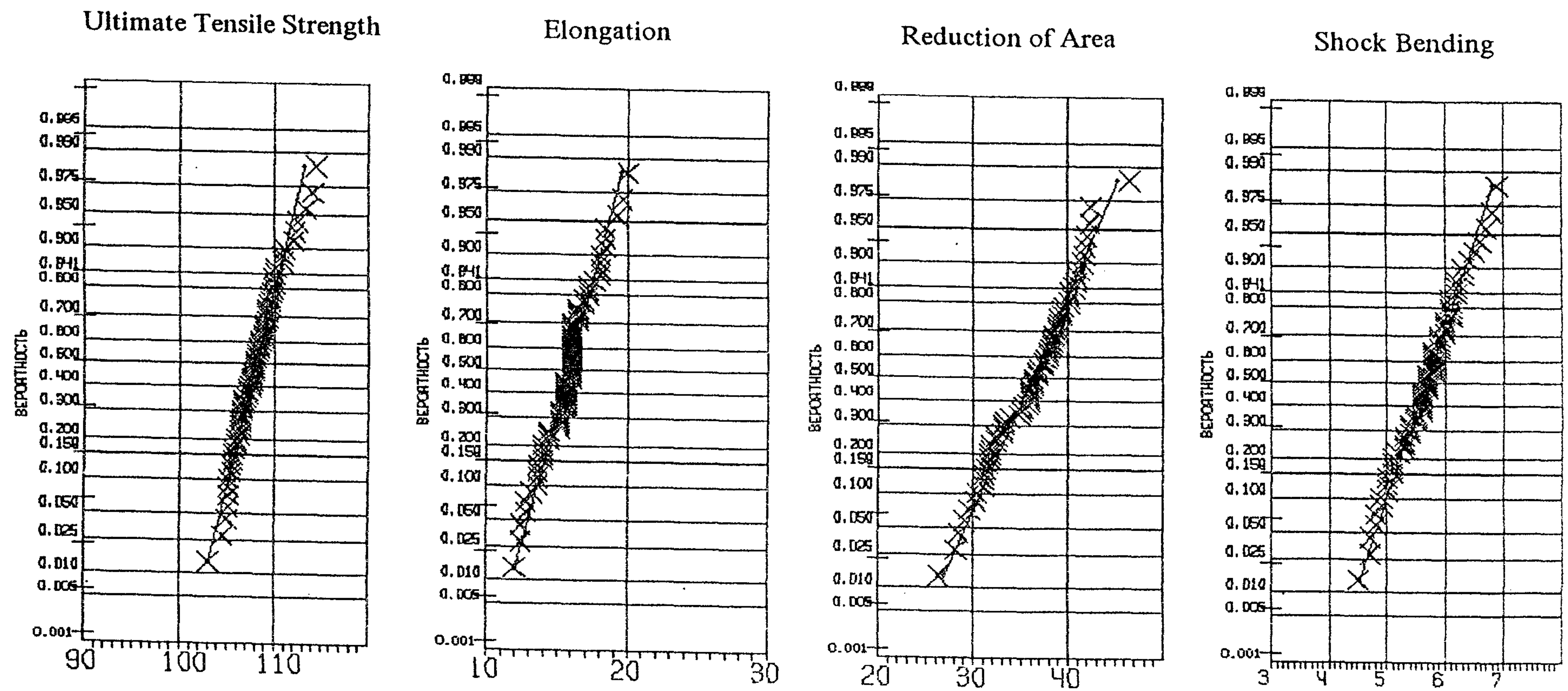
1. MIL-E-005007E (AS) 01.09.83.



SPECIFICATION: UTS $\geq$ 96kg/mm^2 $\quad$ $\delta \geq 9\%$ $\quad$ $\psi \geq 22\%$

Fig.2 Statistical distribution of VT3-1 alloy mechanical properties.

DEVELOPMENT OF NI-BASE SUPERALLOYS FOR GAS-TURBINE DISKS

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Aircraft engine industry in the process of its improvement has always specified stringent requirements to materials, what have had a stimulating influence on developments and introduction of new Ni-base superalloys for production of gas-turbine engine disks.

Optimum composition, structure and mechanical properties of Ni-base superalloy disks are obtained due to appropriate alloying of these alloys and process variables of their production. That is why development of these superalloys was carried out in two directions, i.e.:

1) improvement of production process and
2) an increase in the alloying degree to improve performance characteristics.

Each stage in the process of technology improvement allowed us to complicate chemical composition and vice versa, perfection of chemical composition of these alloys necessitates optimization of process variables.

One of such stages in the superalloy developments was introduction of vacuum arc remelting (VAR) for production of Ni-base superalloys early in the 60s. An improvement in grain structure due to VAR ensured a noticeable increase in plastic property level of the alloy. This allowed us to alloy such alloys as, for example, EI437B with main alloying element (Ti), to raise level of mechanical properties (Table 1) and also to begin producing more complex-alloyed alloy EI698 with higher Mo and Nb content and, thereby, to attain UTS > 1200 MPa.

Application of double vacuum remelting (VIM+VAR) opened the possibility for introduction of eve more alloyed alloy EP742 which is superior to EP437BU and EI698 alloys in mechanical properties (Table 1).

Unfortunately, an attempt to apply a new highly-alloyed superalloy EP741 for production of disks by conventional ingot metallurgy technique was unsuccessful, since such complication of alloy composition resulted in an increase in chemical and structural inhomogeneity of an ingot and impaired its ductility. Besides, narrowing of the homogeneity field hampered deformation and heat treatment processes used in component production.

Thus, progress achieved in the field of development of new Ni-base superalloys necessitated a new technology which could allow one to overcome noticeably chemical and structural inhomogeneity and ensure fine grain state and uniform distribution of dispersed phases formed in solidification process. All these problems could be solved due to application of a new process, i.e. powder metallurgy technique.

Mastering of powder metallurgy technique was began in the 70s using highly-alloyed EP741 which contained 55% of strengthening γ'-phase (Table 2). Slight correction of chemical composition of the alloy and development of new heat treatment conditions taking into account features of PM alloys, such as fine grains, dispersity of the strengthening γ'-and carbide phasis, etc. allowed us to introduce this alloy, known now as EP741P, for production of gas-turbine engine disks. This alloy shows ultimate tensile strength of > 1250 MPa and stress-rupture strength of > 900 and 600 MPa at 650 and 750°C respectively.

Essential advantages and potentialities of powder metallurgy technique in comparison with conventional methods were revealed in the process of development and mastering of gas-turbine engine component production by PM technique. A new stage in development of Ni-base superalloy was outlined. This new technology did not limited potentialities in the field of complication and improvement of alloy alloying.

The first alloy developed especially for production of gas-turbine components by powder metallurgy technique was EP741NP alloy widely used at present. In developing this alloy, content of main γ'-forming elements such as Ti, Al and Nb was increased for formation of 60% of the γ'- phase and temperature of complete dissolution of the γ'-phase was elevated up to 1185°C. This alloy showed a wide range of high mechanical property values after treatment according to standard and special heat treatment conditions, namely: UTS > 1300—1400 MPa, 0,2%YS > 90—100 MPa, 100 hr stress-rupture strength = 1030 MPa at 650°C and 680 MPa at 750°C.

In subsequent work concerning improvement of Ni-base superalloys, two trends stipulated by requirements of gas-turbine engine designers began to show. One trend was development of high-strength alloys for operation at temperatures of 650-700°C and another one was development of high temperature alloys for operation at temperature above 750°C and higher.

In developing the high-strength alloy, further strengthening of solid solution and preservation of rather high content of the strengthening γ'- phase (55%) were provided for. This allowed us to attain the following mechanical property values of this alloy; UTS = 1500 MPa and 100 hr stress-rupture strength at 650°C = 1050 MPa. In creating the super high-temperature alloy, the target was to increase an amount of the strengthening γ'- phase, while rather high plastic properties should be preserved. This new PM alloy EP975P in the case of γ'- phase content of ~ 64% showed high temperature strength of 740 MPa at 750°C. It should be noted, however, that this alloy, due to its limiting alloying, has susceptibility to formation of embrittling technologically close-packed (TCP) phases, in particular, μ- phase.

Thus, in the course of subsequent studies concerning improvement of alloy compositions the possibility of embrittling TCP phase formation, especially in the case of an increase in Cr, W and Mo contents, had to be taken in to account. Besides, it was necessary to ensure rather wide field of alloy homogeneity (at least 50°C) to have the possibility of controlling sizes of grains and the strengthening γ'-phase.

In view of the above mentioned, in developing a new PM alloy which could combine high strength values of EP962P alloy, excellent high temperature strength of EP975P alloy and universality of EP741NP alloy, a method of simulation of mechanical properties via regression analysis and a mathematical method of alloy evaluation in terms of the possibility of embrittling.

The latest studies resulted in development of a universal PM alloy (Tables 1,2) which shows a wide range of homogeneity (50°C), contains ~ 57% of the γ'- phase , has no susceptibility to TCP phase formation. Besides, this alloy shows high strength in range of operating temperatures of 650-750°C in combination with insensitiveness to stress concentrators.

One more method aimed at further improvement of mechanical properties of PM superalloys is deformation of HIPped powder compacts. There are some deformation techniques, but in our option, isothermal forging of extruded compacts is the most worth-while method. Table 1 shows mechanical properties of superalloys.

Table 1. Mechanical Properties of Ni-Base Superalloys

Production technique	Alloy	Mechanical properties at 20°C					100 hr. stress-rupture strength (σ)	
		UTS	0,2YS	EL	RA	KCV	650°C	750°C
		MPa	MPa	%	%	MJ / m²	MPa	MPa
Atmosphere melting	EI437B	820	600	10	13	0,25	-	300
Vacuum melting	EI437BU	1000	680	13	16	0,30	600	320
	EI698	1150	720	14	16	0,30	740	420
	EP742	1200	750	14	14	0,30	810	520
Powder Metallurgy (PM)	EP741P	1250	800	18	20	0,50	900	600
	EP741NP	1300	900	18	20	0,40	1020	680
	EP962P	1500	1150	10	12	0,23	1050	650
	EP975P	1400	950	12	14	0,25	1050	740
	EP962NP(new)	1500	1150	10	12	0,30	1100	750
PM+ isothermal forging	EP741NP	1590	1080	17	17	0,40	1050	-

Table 2 Main Phase Characteristics of Ni-Base Superalloys

Production technique	Alloy	Amount of the γ'-phase, wt %	Critical points, °C	
			$T_{\gamma'}$	T_s
Atmosphere melting	EI437B	10	950	1300
Vacuum melting	EI437BU	12	965	1300
	EI698	20	1020	1320
	EP742	30	1100	1280
Powder Metallurgy	EP741	55	1170	1280
	EP741NP	60	1185	1270
	EP962P	55	1170	1200
	EP975P	64	1215	1250
	EP962NP(new)	58	1200	1250

THE PROCEDURE OF ADMITTANCE OF FOREIGN FUELS, OILS AND GREASES FOR RUSSIAN-MADE AIRCRAFT ENGINES

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The expansion of international air-lines of AEROFLOT and an increase of Russian-made airplanes helicopters and aircraft engines export leads to the necessity of ensuring their performance reliability when using the foreign-made fuels, oils and greases. Due to the fact that Russian companies have started purchasing foreign-made aircraft nowadays it is of interest to consider the possibility of using the Russian-made fuels, oils and greases on foreign aircraft too.

The paper presents an adopted in the Russia procedure of permitting foreign-made aviation fuels, oils and greases to be used in Russian aviation.

In accordance with the results of experiments carried out by CIAM and other organizations it has been revealed that the foreign made types of fuels, oils and greases being an equivalent to some Russian-made product have essential differences from Russian products. The production was used when testing Russian-made aircraft and during the whole period of the utilization of the aircraft in our country now. Foreign specifications do not make restrictions on the production some features of which exceed the required limits of GOST(OST,TU). In such cases a tendency has been observed to the reduction of quality of some material (fuels first of all) which leads to lessening specifications requirements.

For reasons given above it was necessary to settle the procedure of using foreign products in the Russian aircraft. Thus, "THE STATE DOCUMENT OF THE ADMITTANCE REGULATIONS FOR FOREIGN-MADE FUELS, OILS, GREASES AND SPECIAL FLUIDS ON SOVIET AIRCRAFT" was worked out by CIAM in 1979 (the second edition was issued in 1987).

This document states the procedure of the admittance, the amount of the investigation and testing work of foreign-made products alongside with the list of organizations responsible for investigation, testing and issuing scientifically grounded recommendations.

So, our Institute, CIAM (The Central Institute of Aviation Motors) is one of the leading establishments making an emphasis in it's work on fuels, oils and greases for the aviation engines and their aggregates. Furthermore, "The State Document..."states the norms for the foreign fuels, oils, greases and special fluids in Russian-made aircraft to be admitted to the Russian market and it's procedure.

The specific types of foreign-made fuels, oils, greases and special fluids can be used in a Russian aircraft if these types are contained in the technical descriptions on the specific kind of an aircraft (airplane, helicopter or engine). A special procedure is worked out for making such a description which is a guide to admitting foreign-made special fluids to be used on board Russian-made aviation production.

"A GUIDE TO FOREIGN FUELS, ENGINE OILS, GREASES AND HYDRAULIC FLUIDS RECOMMENDED FOR USE IN RUSSIAN AIRCRAFT" and annual ADDITIONS to it are elaborated by CIAM at the initial level. These documents are approved by Goskomoboronprom. Depending on the results of laboratory experiments on foreign samples of fuels, oils and greases "A Guide..." states the possibility of their "constant" or "temporally" utilization on Russian-made aircrafts.

The next document is a special chemical list of fuels, oils etc. named "himmotologicheskaya karta". This list is made on the basis of "A Guide..." by the creator of aircraft according to the requirements of GOST 25549-82. The list is the main paper that states technical and economical grounds for using specific types of Russian- and foreign-made products for a definite model of an aircraft.

Finally, the mentioned types of foreign fuels, oils, greases etc. which are included in the chemical list must be included in the technical description of the definite model of an aircraft.

Therefore "A Guide..." is a key document as mentioned above.

"A Guide..." (and its export editions too) is published every 5 years (the 5-th edition — in 1993). The last annual Additions are published in 1994.

"A Guide..." is worked out on the basis of recommendations given in accord with the results of laboratory experiments of foreign-made products or the consideration of experience on using foreign types of fuels, oils etc. on Soviet aircrafts.

The foreign types of fuels, oils and greases are included in "A Guide..." may be divided in two groups. The first group includes foreign types of fuels, oils and greases that can be looked upon as the equivalent to Russian types. The second group includes foreign types of products with the deteriorated properties in comparison with the Russian equivalent types. In this cases limits should be set. For instance, the guaranteed service life of fuel assemblies of engine control system may be reduced due to poor lubricating properties of that sort of foreign fuels or its poor compatibility with some aircraft fuel tank sealants. In such a case the guaranteed service life is set jointly by an aircraft designer and manufacturer. This Decision should be approved by the staff of the leading scientific research institutes.

The procedure and the amount of the laboratory experiments of foreign products for them to be included in "A Guide..." correspond to the requirements set for the current procedure of admitting Russian types of fuels, oils etc. for Russian-made aircraft engines.

Russian-made experimental samples of fuels, oils etc. should pass the State Accepted Test before they can be utilized. It comprises 4 stages:

— laboratory testing,
— test on the engine,
— control-flying test,
— operational test.

It should be mentioned that in some cases the results of passing two stages of that Test are enough to make a decision for using them in the aviation industry.

The Special Complexes of Methods for Qualifying the production (CMQ) has been elaborated for carrying out laboratory tests (first stage) of Russian-made experimental products and some of the commodity (standard) products.

The latter are produced according to the requirements of GOST(OST,TU) with some unimportant deviations (negligible changes in chemical composition or technology etc.).

There are 6 CMQ being applied for aviation products (fuels, oils, greases etc.). One of them was used for laboratory testing of engine oils in gas turbine engines and helicopter gears. CMQ for engine oils includes 22 special methods in addition to the standard methods included in GOST(OST,TU).

As stated above foreign types of aviation products must be tested before using in Russian aircrafts. Note should be taken that the application of foreign products (engine oils, the commodity (standard) products) should be considered before their usage.

But however these oils are known to be already used on foreign-made aircraft engines. That is why the procedure of admitting these foreign products may be similar to the current procedure that the Russian-made products pass having some unimportant deviations from the requirements of GOST (OST,TU). These foreign products may pass the first stage of the Test in accord with the requirements of CMQ. It is not worth checking foreign-made production making it pass 4 stages of the Test entirely because these products are not experimental.

In our experience different types of foreign oils produced in accord with the requirements of the same specification but made by different companies might pass the Test having different results. This wide range of tests (CMQ) is intended for analyzing the peculiarities of Russian-made oils application in Russian aircraft. For instance four types of oils approved by DERD 2487 (Castrol 98, Turbonycoil 35A, Turbonycoil 35M, AeroShell Turbine Oil 750) were investigated but only one type was allowed to be used in the Russian aviation instead of Russian-made types B-3V, LZ-240. This type is in the list of "A Guide..." with certain limitations, however. Same results have been obtained when the investigation work has been carried out of oils produced in accord with the requirements of MIL-L-23699C. For example, Mobil Jet Oil II was admitted for use in our aircraft instead of Russian oils: IPM-10, VNII NP 50-1-4f, VNII NP 50-1-4u, 36/1-KuA, B-3V and LZ-240. ASTO 555 having a higher level of thermal oxidation stability in comparison with Mobil Jet Oil II is characterized by the increasing corrosion of Russian-made materials. For this reason ASTO 555 is not admitted for usage in Russian aircraft.

Whenever comparison is made on the possibility of substituting Russian oils for foreign ones and visa versa following the requirements of CMQ to four main performance parameters are considered:

- — thermal oxidation bulk oil stability according to GOST 23797-79;
- — anti-wear properties in accordance with the adopted method of qualification evaluation on special Gear Test (SH-3);
- — compatibility with the Russian-made metal-line materials and coats;
- — compatibility with the Russian-made elastomers.

15-year experience of the "State Document..." application (the one that has been elaborated in CIAM) has proved that the procedure of the admittance of foreign fuels, oils and greases for Russian-made aircraft engines is correct in general. In accordance with "The State Document..." more than 200 samples of foreign products were examined, then some of them were listed into "A Guide...".

As a conclusion it would be worth mentioning fields of further investigation work that might be done jointly with companies expressing interest:

- — working out the procedure of admitting Russian-made fuels, oils etc. for foreign-made aircraft;
- — the improvement of the current Test Methods for jet fuels, engine oils and greases in accord with the main physical, chemical and operational features.

DEVELOPMENT AND MODERNIZATION OF ICE PROTECTION SYSTEMS TESTS METHODOLOGY

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1. Analysis of icing conditions and requirements to engine ice protection system (IPS).

A lot of firms in Russia and abroad have been attending to problems of icing and in particular to certification IPS of air engines. Correctly conducted certification defines reliability of IPS work in service. The experience of work on certification of air engines IPS, which is accumulated in CIAM is generalized in given report.

Icing of aircraft and components of engines inlet can occur at operation of flying vehicles in difficult meteorological conditions. Icing of inlet elements arises by operation of a engine in conditions, determined by temperature of environment, content and size of water drops in atmosphere, by horizontal and vertical extent of a icing zone, by speed and altitude. These conditions are generalized on the basis of long-term supervision in the domestic Norms and Air Rules, in the similar foreign normative documents FAR and JAR.

Main requirement to IPS is maintenance of engine operation in icing conditions without essential deterioration of its parameters. The reliability of engine IPS operation should be confirmed by certified tests according to the acting specifications.

The analysis and comparison of domestic (NLGS-3) and foreign (FAR-25, JAR-E) normative documents was conducted. The domestic norms present more rigid requirements for realization of icing tests, and the foreign ones pay considerable more attention for calculation methods and for experience on similar engines.

The necessity in growing similarity of the domestic and foreign norms, mainly AR and FAR appear due to international cooperation in the aircraft area. The growing similarity of the domestic and foreign norms requires on the one hand greater development of calculation methods, and on other hand reduction of tests number and at increase of their quality and objectivity, deeper understanding of physical processes occurring at icing of inlet elements of engines.

The increase of quality of tests at reduction of their number is reached by following measures:

1) Application of calculation methods for prediction of the ice accretion size change of the characteristics of engine in icing process, that permits to use results of tests, received in

limited area to parameters, on all area of normalized icing conditions;

2) Using of calculation and experimental techniques, enabling to create and to supervise water-air flows with necessary spectral structure of drops, water content of flow before an inlet of engine and uniformity water-drop distribution during realization of certified tests;
3) Experimental determination of thickness and structure of ice layer, use of high-speed filming and video tape recorder for researching the process of drop flight, its interaction with surface, gear of ice accretions formation.

Principles and approaches to modernization of test base for IPS research developed in CIAM presents in the subsequent sections.

2. The analysis of domestic and foreign experimental base on maintenance of check of engines IPS efficiency.

The analysis of domestic and foreign experimental base for IPS tests was executed.

The requirements to modern test base was formulated on the basis of the conducted analysis:

1. The experimental base should provide the opportunity of realization of high-altitude and ground tests of engines and their elements pursuant to the acting domestic and foreign normative documents, as well as opportunity of research of physical icing processes. For these purposes it is desirable to have various installations for full-size and scale of tests, connected to altitude-compressoring station. In EX-SU the similar base is present only in CIAM.
2. Test base for research and certification of IPS should include special laboratory for graduation of injectors or other means for production of water-air flows. The equipment for atomizing of water drops should provide required water content, average size and distribution of drops on the sizes. The most advanced way of water-air injectors graduation application of laser particle size analyzer ("MALVERN").
3. It is necessary to create methods and means of control changing of water-air flow parameters during movement in supplying branch pipe.
4. Development and creation of modern means of the control at growth of ice accretion is necessary too.
5. All controllable parameters should be registered by computer system.

3. Creation of experimental base for engines IPS research.

Three rigs were created.

The first rig, intended for realization of certified tests of full-size engines. The second, scaling rig for realization of experiments on research of icing process of separate engine inlet elements of various geometry, processes of flight of super cooling drop and its interaction with a engine surface, process of drops evaporation in air flow, influence of height to icing processes and etc.. In third, laboratory installation for injectors and other means of generation water-air flow graduation.

Research of icing of full-size engines is developed on installation U-7 M of CIAM, enabling to conduct climatic tests of engines in altitude cell in earthly and high-altitude conditions. The installation construction permits to change distance from water-air injectors up to engine inlet, and to asses uniformity of water drop distribution of flow qualitatively on icing of technological grids, established on engine inlet.

For reception normative water drop size of flow are used injectors 11-524 and 11-525, graduated by analyzer "MALVERN-2600 C".

The CIAM laser analyzers water content and drop size of flow, are supposed to use, that will allow to supervise change of ones during realization of experiment, and to determine real distribution of drops on the sizes in flow in front of engine.

The rig is equipped by television installation, enabling to conduct constant supervision during tests at inlet part of air intake.

Application of high-speed filming is possible. The installation permits to conduct researches of icing of engine in all range of temperatures and water contents of flow, required by the Norms NLGS-3 and FAR-25 at altitude from 0 up to 7 km.

Installation in altitude cell for realization of scaling tests of engines elements was mounted. The installation has opportunities, similar to installation U-7 M. the Given installation permits to conduct comparatively cheap tests of air engines inlet elements, evaluating their efficiency for icing conditions. Realization of researches of different nose failings, with various speeds of rotations. The installation permits to make optimization of geometry of engine inlet from the point of view of icing danger minimization.

On the basis of installation U-281 rig, enabling to determine water drop size in water-air flows, created by pneumatic injectors, was developed. Researched injector is placed before of laser beam analyzer. After passage of laser beam, the researched flow arrives in air hole. Injector water and air, flow are measured by flow meter and by orifice plate accordingly.

For measurement of sizes of drops and their distribution on the sizes was used laser particles sizes analyzer of firm "MALVERN INSTRUMENT" (Great Britain) model "MALVERN 2600 C". This model is intended for measurement of the sizes of solid particles and liquid drops in air flow and in liquid mediums in a range from 0,5 mcm up to 2000 mcm.

As a result of realization of experiments the characteristics of number pneumatic injectors were received, that has allowed to determine parameters to receive arithmetic average diameter of drops 20 mcm, necessary for realization of certified tests.

Besides tests of several types fuel injectors with the purpose of determination opportunity of reception of spectra of drops, appropriate foreign normative documents (FAR, JAR) are conducted.

4. Development of methods of the control of parameters of water-air flows in channels of test installations.

Change of water content and drop size of flow as a result of water phase conversions drops is possible during flight of drops from injectors up to a engine inlet. Therefore during tests realization of valuation of influence of water drop evaporation in different phase condition, them water size drop are entry conditions on parameters water-air flows on high-altitude climatic test beds is necessary.

Mathematical model of water and ice particles evaporation for this purpose was created. The model permits to determine distance from water atomizing injector up to a engine inlet, necessary for reception of required super cooling of water drops. Besides it permits to calculate quantity of water, which up to engine will evaporate, and to evaluate change of drop size spectrum.

As a result of calculation was established that distance from generator of water-air flow up to engine inlet on a high-altitude rig could not less than 4,5 m for maintenance of equality of temperature of drops to temperature of air was.

The size of drops not essentially varies during stay in a flow (not more than on 2%), however water contents can change considerably (up to 7% and more). This effect can be compensated by inject in addition appropriate quantity of water in flow. On the other hand, for reduction of evaporation it is necessary to inject water at it is possible to low temperature. It is possible to recommend temperature not more than 20°C. The humidity of air flow should be not less than 85%.

The small size drops are cooled quickly, but lose large quantity of water as a result of evaporation. Therefore, the flow is enriched by drops of the larger size. On the other hand the larger drops are cold slowly and can have temperature above temperature of flow on 5-10° C. Hence at generation of water-air flow must to be achieved uniform.

So experimental installation and techniques are created enabling, first, essentially to facilitate creation and operational development of air engines IPS and, secondly, to conduct certified tests of IPS at high technical level, pursuant to the international normative documents.

CERTIFICATION OF QUALITY SYSTEM AND MANUFACTURE OF AEROENGINES AT MOTOR SICH JSC

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Motor Sich JSC is a leading enterprise in the CIS countries, manufacturer of high quality aeroengines, and it pays great attention to quality of the manufactured products and continuously improves its Quality System.

To retain the place in the market under the conditions of growing competition at the world aeronautical equipment market, we developed in 1989 the program of improvement of integrated quality management system effective at that time with the aim to modernize it on the basis of international principles stipulated by ISO Standards of 9000 series.

We were the first to solve this problem in our branch of industry and in the country, because at that time it was the very beginning of implementation of ISO series 9000 Standards in the USSR.

Therefore we had no one to give us recommendations, advise methodological assistance in correctness of understanding of the requirements stipulated by these Standards. We, by ourselves, studied corrections in standard documents of our Enterprise.

With this aim the following actions were undertaken: a working group was created for preparation for experimental approbation of ISO 9000 International Standards, a list of production items subject to certification was prepared, all engineers and technicians attended special training courses, a schedule of making corrections in existing Factory Standards and working out of lacking ones was prepared, a contact was concluded with NIAT for carrying out preliminary survey of the Enterprise to assess its readiness for the certification of production. In April 1990, NIAT in cooperation with NIID and NIISU carried out preliminary survey of the Enterprise and prepared the technical report.

When studying the NIAT recommendations on improvement of the Quality Management System effective at that time, the specialists of the Enterprise took notice of the routine approach to solving this problem at all the enterprises of this manufacturing branch which was manifested in use of obsolete administrative-and-command methods of management and which was unacceptable, since each enterprise had its individual distinguishing features; the necessity to take this into account is stipulated by the concepts of International ISO 9000 Standards.

After thoroughly studying once again ISO 9000 Standards and taking into account the recommendations of Branch Scientific Institutes we set about to elaborate Quality System documents.

When making preparation for the certification of production special attention was given to subject of Quality System documentation.

The Quality System of the Enterprise encompasses all stages of the 'quality loop', specified in ISO 9004 International Standard, from marketing to product utilization.

The foundation of the System is Enterprise Policy in the Field of Quality.

The main standard document of the System is Quality Manual which has the status of Factory Standard and serves as a ' Constitution' of the Enterprise in the field of quality.

When creating our Quality Manual in 1991 we chose the variant presuming detailed description of the requirements set for each element of the Quality System on the basis of ISO-9001 and 9004 Standards and with account of the requirements set in Regulations АП-21 worked out by IAC Aviation Register.

At the end of 1991 the situation was such that we required a competent agency, with sufficient experience, recognition and reputation in the field of certification, to evaluate our Quality System.

We selected Bureau Veritas of France to be such an agency.

In the autumn of 1991 we invited the specialists of this company to our Enterprise; they assessed our Quality System and suggested to enter into a contract.

In April 1992 the contract was concluded. It embraced the following items:

— Training of our specialists in Paris;
— Performing analysis of the Quality System;
— Rendering methodological assistance in refining the Quality System to bring it to the level required by ISO series 9000 Standards;
— Training of our auditors who carry out internal audit (and issuing Certificates to the auditors)
— Carrying out certification of the Quality System.

Concurrently, all the above mentioned activities were performed by the French company at the ZMKB "Progress" which is the principal designer of aeroengines manufactured by our Enterprise.

The program of work was expended to be completed in 7 months but due to various objective reasons the work was actually completed in 10.5 months

The first audit was a rather profound one. It was carried out by four specialists for a period of two weeks. As a result of the audit, the Quality System was evaluated at 93%, remarks were made and recommendations were given.

In cooperation with the French company and IAC Aviation Register we issued 2nd edition of the Quality Manual at the end of 1992.

The work under the contract was completed in May 1993; at that time four Bureau Veritas specialists within one week carried out random inspection of the documents pertaining to the principal directions of the Quality System and actual fulfillment of the documents' requirements in production units and verified implementation of all the recommendations which were given by the company at earlier stages.

As a result, on June 15, 1993 at Paris Air Show, La Bourget, the Motor Sich Enterprise and "Progress" Design Bureau were issued the Certificate for Quality System effective till May 14, 1994 and on July 15, 1994 the validity of the Certificate was extended till July 14, 1997.

We concluded the contract with Bureau Veritas for supervision over the Quality System and we fruitfully cooperate with this company.

After performing the work for modernizing the current Quality System to meet the requirements of international principles, the Motor Sich JSC entered into a contract with IAC Aviation Register for performing work on certification of production of aeroengines.

In the course of certification of the production, IAC Aviation Register, side by side with checking compliance of the Enterprise Quality System with ISO-9000 International Standard, attaches special importance to the reliability of the production items to be certified in accordance with the requirements of АП-21 Aviation Regulations

Members of the IAC Aviation Register commissions were the leading specialists of NIID and TsIAM and they, after thoroughly scrutinizing the Quality System documents, technical documents for engines, performance of operations and processes at work stations as well as information pertaining to reliability of the engines to be certificated, presented to the Motor Sich JSC the Audit Report, wherein discrepancies preventing issue of the Certificate were specified.

After elimination of the causes specified in the remarks and modernizing the Quality System in accordance with АП-21 Regulations, IAC Aviation Register issued to the Motor Sich JSC the Certificate for manufacture of the following engines:

Д-36 Series 1 and 2A

Д-136 Series 1

TB3-117 BM
TB3-117 BM Series 02
TB3-117 BMA
TB3-117 BMA Series 02

As a result of the work performed, the Motor Sich JSC was admitted to the World Trade Leaders Club (WTLC), which is a prestigious social organization supervising the quality of exported goods, and was awarded Gold Mercury Grand Prize.

FATIGUE RESISTANCE PROBLEMS IN GASTURBINE ENGINES (GTE)

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More than 60% of failures at a development stage and GTE operation are connected with strength problems and caused first of all by fatigue defects. By defect frequency of appearance in main engine parts the following row could be selected: compressor working blades, turbine working blades, compressor disks, bearings, combuster gas, turbine disks and shafts. Complexity of fatigue damage prediction of GTE parts is connected, from one side, with a variety of factors which influence upon mechanical strength of materials. These include technological features of parts manufacture, operational conditions, design and computation methods, etc. From the other side, it is necessary to take into account a possibility of appearance of dangerous level of variable stresses which are random time functions, operational conditions and design factors. The maximum carrying ability of a part can be provided by:

— technology upgrading at any manufacture stage which allow to maximum realize the strength properties of applied material;
— accounting at a design stage of part operational features both from the point of view of force action and of environmental influence;
— application of special technologies which secure additional increase of material strength characteristics and part carrying ability.

Significant expenditures are required to create structures having high carrying ability and to run special strength tests of materials and full-size parts with attraction of complex research by metallurgists, metal-physics and other specialists. However these expenditures can not be compared with moral and material losses which could have place during aircraft destruction.

Compressor blades. The only from of compressor blades destruction is fatigue. Defects resulting in destruction could be conditionally broken into:

1) design (existence of resonance, surge, connected vibration and so on);
2) technological (violation of punch or thermoprocessing regimens, deviation of part geometry from drawing specification, local structure non-uniformity, hydrogen material

saturation, unfavorable residual stresses, ridges overriveting and so on);

3) operational (foreign objects infrusion, violation of operation, conservation and storage regimes and so on).

The above mentioned defects are distributed as follows: 29% — design; 17% — technological; combination of design, technological and operational.

Turbine blades. Blade fatigue damages accounts for up of 33%; law-cycle fatigue damages — 9,3%; combination of low-cycle fatigue, vibration stresses — 41,7% and damages connected with reaching limit of song-term strength — 16%. The cases of blade damages can include:

1) reduction of σ_{-1} due to formation of micro cracks in the surface layer damages by machining;
2) increase of σ_v due to the increase of shrouds clearances because of contact spots wear or material creep;
3) disagreement of specifications concerning the shroud ten-n;
4) non-uniform temperature field at turbine inlet;
5) violation of operation conditions;
6) imperfection of cooling system;
7) change of excitation conditions nozzle vanes burn-through; choking of fuel nozzles, etc.;
8) non-uniformity of contact at root teeth;
9) violation of root profile cutting regimes;
10) imperfection of process of blade forging, thermal treatment or casting.

Turbine blade destruction caused by violation of manufacturing processes amounts to 25,1%, by designing imperfection — 28,2%, the rest 51,7% fall at unfavorable combination of structural and technological factors.

The destructions of disks and shafts, though they are encountered more seldom, result, as a rule, in grave consequences as 50% of these destructions are uncontained. The causes of disk and shaft destructions can be both structural, operational and technological.

The main technological factors influencing the GTE parts destruction include a number of signs of technological heredity. They include increased degree of cold-work hardening, local nonuniformities of structure or chemical composition, internal defects of material, residual stresses. Each sign accounts for about 20% of defects.

During the process of parts manufacture the sources of the "dangerous" technological heredity (depending on the material type which they are made of) can be the "high risk" procedures. Such procedures include: forging, casting, thermal treatment, milling or turning, development of parts with the help of abrasive wheels, pulling, forge rolling, polishing, coating.

Against a background of the negative technological heredity (unspecified and non-optimum properties of the surface layer, unfavorable shifts of tolerance fields, etc.). The structural defects resulting in crease of components and parts stress level are especially dangerous. The initiation of fatigue defects, for example, in compressor disks, can be connected with both the change of the operation conditions on the engine test level is also influenced by design features of the stage (including blades and vanes), disk, air bleed system and so on.

The structural steels and alloys used in the existing GTE are characterized by increased sensitivity to stress concentration (notches, mars, fillets, etc.) when the variable stress affect the part that is shown in the high material sensitivity to the technological heredity. The properties of structural materials manifest themselves not only in the form of technological heredity, especially during the high-risk manufacturing procedures, but in operational damage ability, when the sources of fatigue failures are: corrosion damages of which can be the same as the depth of marks from machining; damages caused by fretting-corrosion; damages caused by foreign objects (dents); surface layer cracking connected with the technological residual stresses or reaching the limit by plasticity or long-term strength and soon.

The most sensitive to stress concentration are the titanium alloys the elastic distribution of stress is fully realized in the stress concentration zone as the value of coefficient of sensitivity to stress concentration is close to 1 ($q = 1$).

Approximately the same properties are characteristic for corrosion-resistant martensitic steel, but these steels sensitivity to stress concentration depends on the regime of thermal treatment (tempering temperature). At the same time the high sensitivity of these steel to stress concentration remains also at operational temperatures (T = 300—600°C), and the sensitivity of 15Х16К5Н2МВФАБ steel even increases with the rise of test temperatures. Austenitie steel of ХН45МВТЮБР-ИД type has 0,9—0,7 sensitivity coefficient to stress concentration the normal test temperature (T = 20°C), but it decreases down to q = 0,71—48 at T = 600—700°C. Besides, this steel is well hardened that allows to reduce q to values of 0,6—0,12, and hardening effect remains the same in the whole operational temperature range.

The wrought nickel-based alloys have a rather high sensitivity to stress concentration at the symmetric loading cycle (q=0,75-0,8), but at cycle asymmetry and with the operational temperature rise the sensitivity to stress concentration of these alloys decreases by a factor of 2—3.

The sensitivity to stress concentration for casting alloys of ЖС type is not high and is equal to q = 0,22—0,32 at temperatures 650—900°C. The carrying capability and especially fatigue resistance of many parts is determined according to the surface layer strength.

So the problem of formation of a part surface layer with a high carrying capability should be solved at the design-work stage and then at the development of the technological process.

The part shaping is usually accompanied by plastic deformation and thermal effect on the surface layer. The plastic deformation at depth and on surface is nonuniform and is accompanied by structural changes caused by both force and thermal effect on metal.

After machining in the micro structure of the part surface layer one can observe the typical marks of plastic deformation.

During the electrophysical treatment methods, which are most widely used as the preliminary procedures of GTE parts treatment, the thermal effect of the procedure manifest itself in the thin part layers, and on a micro structure it is seen as a light, leaned by alloying components layer or a casting structure to a depth of 100 mkm. This layer is small during the electrochemical treatment. It is characterised by low surface roughness, but in this case the metal picking along the grain boundaries takes place. It reaches 20—30 mkm in wrought alloys and in casting alloys it is significantly deeper.

When the profile sections of blades made of wrought alloys are subjected to finishing treatments using the abrasive instruments (grinding, polishing) and during the development of cast blades edges the sharp temperature rise can cause the burns on the part surface which result in a spread in physical-mechanical and chemical characteristics of the surface layer, and also in reduction of fatigue resistance, especially in corrosive mediums. The most instable properties are observed during the manual blade treatment.

The surface plastic deformation (SPD) alloys to in crease the carrying capabilities of parts at the expense of maximum use of physical and mechanical properties of the surface layer. The SPD method selection depends on a part structure and dimensions, requirements to geometry parameters accuracy and surface roughness, operation conditions and life.

During machining, when the part surface experiences the mechanothermal effect, the incompatible processes occur in it strengthening and loss of strength, electrical and magnetic properties change. Conventionally the parameters of the surface layer in macro volume can be divided into:

1) physical — sizes of phases, grains and blocks, density of dislocations, concentration of crystal lattice;
2) chemical — including phase composition, element concentration in alloy volume, element concentration in phase;
3) connected with deformation — degree of material deformation (cold-work hardening), depth of its penetration and residual stresses;
4) geometric — which characterized by roughness that is determined by the aggregate of parameters of asperities forming the surface profile; the geometric parameters play the

role of stress concentrators, participate in forming the actual area of contact, they are determined mainly by the type of finishing treatment.

The deformation criteria are most wide used, because the methods of their determination are simple both using the specimens and parts. The deformation parameters can be controlled during the part treatment by means of changing the regimes and cutting conditions, thermal treatment or using other methods of surface treatment. When the severe regimes of treatment take place, for example SPD, the depth of the plastically deformed layer increases greatly and the layers parameters (deformation and physical) change. The significant plastic deformations also occur during the preliminary treatments, but in this case the tensile residual stresses are generated. Even under optimum cutting conditions and forced tool sharpening the depth of the plastically deformed layer in nickel alloys can be 50 — 100 mkm. After electropolishing or stabilizing thermal treatment the density of dislocations is minium, the deformation parameters (degree of cold-work hardening) decrease, the residual stresses are absent.

The residual stresses are either the result of technological heredity exhibition or the effect of operational loads and environment conditions. The value and depth of treatment regimes (cuttings rate and depth) and alloy structure. The value of residual stresses titanium alloys also depends on relative content β-phase. Conversion of β-phase into α-phase is accompanied by volume reduction and depending on the initial content-phase even the tensile stresses can appear in the surface layer.

The part surface layer is a carrier of technological heredity where all the parameters act simultaneously as many of them are interdependent and cannot exist separately , for example, deformation and physical parameters and so on.

Fatigue resistance depends largely on both the surface roughness and the observance of optimum regimes of treatment. The violation of treatment regimes during the preliminary procedures brings to nothing the role of treatment procedures for titanium alloys.

Reduction of R from 2,5 down to 0,63 mkm for 13X11H2B2MФ steel is accompanied by increase of endurance limits by 25—40%. Nevertheless, the tests of XH73MБTЮ alloy with different roughness parameters (subjected before tests to stabilizing treatment in order to relieve the residual stresses) did not show the large difference in endurance limits. This points to the availability of a stronger parameter affecting σ_{-1} than roughness R and substructure characteristics $< D >$,$< e >$.

It is determined that compressive stresses always favor the increase of endurance limit, but their effect on the smooth specimens fatigue is substantially less than the negative influence of tensile residual stresses. When the tensile stresses reach the yield strength, the endurance limit of only 10% from the specified value. The strongest effect of residual stresses on the fatigue resistance is observed at normal and moderate temperatures.

The significant role of residual stresses in forming the carrying capability of blades also was shown in simulation of the optimum technology process for manufacturing the turbine rotor blades from one -piece forged blank (ХН73МБТЮ alloy).

At normal temperature in the range of 10^5—10^8 cycles the following relationship is true for titanium alloys:

$$\frac{\sigma_{-1N\ell}}{\sigma_{-1N}} = 1 - \frac{|\sigma_z|^k}{\sigma_B}(b \lg N - 1)$$

where:

$\sigma_{-1N\ell}$ — restricted endurance limit of materials with induced residual stresses;

σ_z — induced residual stresses;

K, b — coefficients (for BT9 alloy: $K = 0{,}55$; $b = 0{,}25$)

With the rise of tests temperature the negative effect of the tensile residual stresses on fatigue resistance decreases.

The endurance limit depending on the surface layer parameters σ_z and R_a and the basic properties of material can be determined as

$$\sigma_{-1N\ell} = \sigma_{-1N}(1 - \psi_0)\frac{|\sigma_z|}{\sigma_B} - K_{R_a}(R_{a\ell} - 1.2)$$

where:

Ψ_0 — coefficient ($\Psi_0 = 0{,}175$) of compressive residual stresses effect ;

$\sigma_{-1N\ell}$ – endurance limit of stengthened material;

R_a — height of micro asperities after SPD;

K_{Ra} — coefficient including R_a effect ($K_{Ra} = 42$ MPa /mkm).

For the elevated temperatures the relationship includes the corrections for temperature, test duration and relaxation of residual stresses. Relationship $(\sigma_{-1})_{отн} = \sigma_{отн} / \sigma_{0.2}$ is described by functions of the from $y = (1 - x)^n$ or $y = [1 - (x)^m]^{1/n}$, where for the given examples $m = 1{,}6$; $n = 0{,}14$—$0{,}4$, when $T = 20°C$.

PERSPECTIVE LUBRICATING OILS FOR THERMALLY STRESSED GAS-TURBINE ENGINES AND DUAL-PURPOSE POWERPLANTS

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At present the situation with the production of aviation lubricating oils and providing the military aviation with them has become especially complicated.

This refers mainly to lubricating oils used for lubrication of supersonic aircraft thermally stressed gas-turbine engines (GTE).

The main conditions determining the operation of the oil in the engine consist in the following:
— thermal state of the engine;
— load on the surface of rubbing parts and units;
— amount of air entering the lubricating oil system;
— construction materials used in the oil system.

The supersonic aircraft gas-turbine engines are as a rule operated in a thermally stressed mode therefore the basic property of the lubricating oil used in these engines consists in its thermal stability at high temperature.

In the last few years gas-turbines have found wide application as drives of gas pumping units (GPU) by GASPROM JSC of Russia in the framework of military conversion programs.

Compactness and low weight characteristics of this engine class provide for high transportability, mobility of GPU to remote and difficult-to-reach regions of Russia.

The transport-oriented gas-turbine engines are characterized by a high capacity factor, reliability and simplicity of operation as well as a durable service life.

In the 90s due to the transition to market economy practically all of Russian and Ukrainian design and engineering bureaus engaged in developing gas-turbines for aviation started activities aimed at converting serial engines to be used in GPU. For example, the following aviation gas-turbine engines are now under development within the conversion program: NK-14SG (SKBM Samara), NK-36SG and NK-38SG (JSC SNTK 'NK Engines", Samara), PS-90GP-2, PNPP "Aviation

Engine", Perm), AL-ST-1, (JSC" A.Lyulka-Saturn", Moscow), D-336-1/2 (ZMKB "Progress", Zaporozhye).

In the nearest future the fleet of block-cast GPU will be enlarged by new more efficient units with driving facilities made on the basis of a new generation of aviation thermally stressed gas-turbine engines.

In the process of converting gas-turbine engines the designer succeeds as a rule in using from 60 to 70% parts of the base engine.

The conversion of gas-turbine engines for use as a driving facility of GPU consists in the following:

— a new unit is introduced into the GTE — a free turbine creating the torque for the compressor drive;
— the control system and the design of its units is altered (therewith it has to be noted that lubricating oil is used as a service fluid for the GPU driving facility, while fuel is used in the base engine for this purpose);
— the design of the combustion chamber is changed to provide for the possibility to use natural gas for the engine operation (in the base GTE liquid fuel is used);
— a number of changes are implemented into the engine performance system taking into account the specific features of the driving facility operation, therewith minor alterations are introduced into the design of the compressor, turbine gas generator and engine drive gear boxes.

For the GTE used as a driving facility of the GPU the specific feature of the lubricating oil system consists in the fact that it provides for the engine service life up to 100 thousand hours.

Up to recently synthetic hydrocarbon oil IPM-10 produced on to basis of the product from wax hydroisomerization and a complex of performance additives was used as the basic oil for the lubrication of modern thermally stressed engines in the temperature range from minus 50°C to 200°C.

Due to technical difficulties and raw material deficit at present the production of IPM-10 base oil was terminated and, as a consequence, the manufacture of IPM-10 aviation oil was stopped.

Research is being conducted at VNIINP aimed at developing an oil of IPM-10 type on the basis of more available technology using alternative raw material types.

An experimental lubricating oil formulation on the basis of low-viscosity polyalphaolefins PAOM-5 produced by JSC "Angarsk Refinery" subsequently subjected to adsorption treatment was developed.

The characteristics of the experimental oil are presented in the Table.

This IPM-10-type formulation favorably differs from the previously produced oil by its low pour point and high thermooxidation stability. Further bench testing of the experimental formulation as well as solution of the problem relating to elimination of raw material-alpha olefins-deficit and organization of the lubricating oil production are retarded due to financial issues.

An alternative way to resume the manufacture of IPM-10 oil consists in the use of the erected KM-3 unit and mastering the second (II) phase of the integrated lubricating oil complex at Volgograd refinery to produce the base oil for IPM-10 oil by means of hydroprocessing of paraffinic feedstock.

The research and development activities by VNIINP in this direction gave positive results. However final development of the technology for the base oil production conducted by VNIINP and finalizing the plant construction as well as mastering the production require considerable financial expenses.

Therefore the problem of developing the IPM-10-type oil, resumption of its manufacture. On a commercial scale is actual and is of strategic importance, on the basis of processing paraffinic feedstock from KM-3 unit by eliminating from the process the cost-effective stages of deep dewaxing and adsorption treatment the "Petrim" oil, inferior to IPM-10 by its thermooxidation stability, has been developed at VNIINP. The "Petrim" oil was developed by the technical assignment of KNPO "Trud" and the "Saturn" plant, coordinated with concern "GAZPROM" and TsIAM. This oil is intended for modern and prospective GTE used as GPU driving facilities. The "Petrim" oil can also be produced on the basis of the mixture consisting of super-refined mineral oil and 5% of diisooctyl sebacate. The "Petrim" oil was tested in the GPA-Ts-25 formulation at compressor station "Togliatinskaya" (operation under observation) for 3400 hours. In the process of testing the oil has not changed its physico-chemical properties. The oil exhibits the required thermooxidation stability and is suitable for use in nearly developed gas-turbine driving facility of new generation with a compression ratio above 20 in the compressor and gas temperature at the turbine inlet in the order of 1200°C (while first generation drives, now in operation, exhibit a level of parameters in the order of 9—12 and below 1000°C, respectively).

This oil proves to be a promising high-quality material, it does not contain any acute short supply components. The characteristics of "Petrim" oil are presented in the Table.

With the development of prospective aviation equipment posing severe requirements to the lubricants compared to the existing ones the problem of creating a new generation of lubricating oils has aroused where requirements for a high level of carrying capacity, thermal stability above 300°C, oxidation stability at high temperature (180—200°C), anticorrosion properties are mandatory.

At VNIINP a multipurpose motor-gear oil "Eridan" has been developed and tested. This oil is intended for the lubrication of prospective ducted fan jet aircraft engines as well as GTE and "Mi"-helicopter transmissions and prospective equipment, being developed by VNIITransMash (Saint-Petersburg).

Besides, "Eridan" oil can be used as a unified multipurpose oil to lubricate GTE and compressors on GPA and GPU with reducing gears on marine vessels, in power units in which GTE are applied.

The oil properties are shown in the Table. The viscosity of "Eridan" may range from 3 to 8 cst at 100°C as provided by the preparation technology without changing the quality of the formulation depending on the unit in which the oil is used. The oil is operable up to 180°C at the engine outlet and for a short period up to 200 C, it exhibits a high level of tribological characteristics (antiwear, EP-and antifriction properties) as well as a high level of protective properties, thus imparting the service-conservation function to the oil.

"Eridan" oil was successfully tested under different operating conditions within the first (I) phase as an aviation oil and on VNIITransMash benches as a motor-gear oil for prospective caterpillar vehicles.

For further testing of "Eridan" oil and for its implementation substantial funding by the Ministry of Defence and by JSC "GAZPROM" is required.

Main characteristics of lubricating oils for thermally stressed gas-turbine engines

N.O.S.	Properties	Norms of TU 38.1012-99 for commercial oil.	Experimental oil "Petrim"	Experim. IPM-10 formulation on the basis of PAOM-5	Norms of TU 38.401829-90 for oil "Eridan"
1.	Kinematic. viscosity, cst: at 100°C at −40°C	min. 3.0 max. 2000	3.4 5200	3.8 2800	min. 7.0 report
2.	Pour point,°C	min. −50	−50	−55 (below)	min. −50
3.	Flash point, °C	min. 190	186	208	max. 0.2
4.	Neutralization number, mg KOH/g	max. 0.05	abs.	0.01	max. 0.2
5.	Content of water soluble acids and alkalis	abs.	abs.	abs.	abs.
6.	Water content	abs.	abs.	abs.	abs.
7.	Content of mechanical impurities	abs.	abs.	abs.	abs.

8.	Evaporation losses under dynamic condition: under air bubbling 1.5 l/min, 175°C for 3 hrs	max. 8.0	5.5	4.0	max. 5.0
9.	Kinematic. viscosity at −40°C after evaporation, cst	max. 3000	—	3000	—
10.	Thermooxidation stability for 50 hrs:				
	under air bubbling 3 l/hr :	at 200°C	175°C	200°C	175°C
	-neutralization number, mg KOH/g				
	-kinematic. viscosity, cst,				
	at 100°C				
	at -40°C	max. 8.0	0.19	2.6	max. 1.0
	-metal corrosion, g/sm.^{2}:				
	steel ShKh-15	max. 4.5	3.45	3.90	max. 9.0
	copper M-1	max. 5000	5300	5900	—
		abs.	abs.	abs.	abs.
		$+2.10^{-6}$	abs.	abs.	abs.

EQUIPMENT AND PROCEDURE FOR STANDARDIZATION OF THE FATIGUE TESTS AND DIAGNOSTICS OF THE DESTRUCTION

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Vibration and fatigue tests with loading close to operational and standard loadings are still modern problems. To create the hardest vibration regime at a preset loading capacity [1,2] is a very difficult.

Up to now theoretical studies concerning the influence of the spectrum width on vibration loading, especially durability, have not been made. From [3] for an inertialess objects it is seen that when using the same capacity spectrum, the spectrum expansion increases the vibratory loading, characterized by the vibration velocity. There are very few experimental results on this matter in literature and they are very contradictive [4,5,6].

The statement concerning the object's vibration strength based on laboratory vibration test results is correct providing three strength failure regimes equivalence conditions are observed, such as: coincidence of the place and nature of destruction, time closeness up to destruction during tests and operation [7].

The lack of methods for direct measuring large deformations [8,9] distorts data on the material's properties. Sometimes the beam's free end vibration amplitude is measured and based on this, the stress value is deducted [10]. The experiment showed that at constant deformation this amplitude gradually decreases and before destruction constitutes approximately 60% of the initial value. Disregard of this factor gives erroneous durability estimations several times different from actual. This method cannot be used to determine stress during random and polyharmonic regimes.

A non-destructive capacitance sensor measuring curvature and bending deformations g_f flat and cylindrical surfaces almost unlimited in value of any scale of deformation has been designed [11]. An oscillatory state self-exciter — SAKR, a device for forming polyharmonic regimes has been designed [12] making it possible to test specimens simultaneously at three natural frequencies until full destruction at any preset deformation amplitude. Installations [1,10] are not suitable for such loading.

1. The object under study. A 363x20x4 mm cantilever beam made from AMg6BM material and fixed to vibration stand was chosen. The first three natural frequencies were 27, 175 and 485 Hz.

2. Test regimes. The spectra of the regimes imitate operational loadings (fig. 1). SAKR realized r.9 and 10 having spectrum as r. 1,2 formed by generators.

3. Basic measurements. Vibration velocity V_{av} was measured at the "input" and stresses G_{av} at the "output" as average values. A_{max} and f_g free end amplitude and frequency, P—electric current power of a vibrator coil and t_d—time up to destruction.

Basic research results.

1. Safe life diagnosis. Up to now two methods: non-destructive control and "safe life counters" have been used to assess residual material safe life [13]. It was determined that for materials with cyclic non-stable properties being loaded at constant vibration velocity the deformation amplitude gradually increases by 30—40% and after 70—85% of the loading time decreases (fig. 2). This property suggests a diagnosis method when 15—30% of life remains up to the end of the load-carrying capacity [14].

2. Equivalence of regimes during fatigue tests. Narrow-band three-modal r.4 (fig. 3) is most hazardous. Life service here is almost two orders less than with r.1 but power consumption increases (fig. 4). R.10 in durability is close to (r.7 and r.4; r.10 is much more hazardous than r.9 and rather more hazardous than r.2.

Conclusions:

a) it is possible to replace random regimes by fixed ones during tests;
b) polyharmonic regime r.10 in durability is closer to random ones.

It should be pointed out that in a small cycle domain there is no permanent equivalence regime, coefficient K^{eq}, and fatigue curves as $t_d \to 0$ "reduce" to value $\sigma = \sigma_{ts}$.

3. Equivalence of regime during vibration tests. No matter what random p.p. 3—8 is "operational" none of the determined types of loading with close durability values yields vibration velocity values commensurate with vibration velocity at random loading (difference 3—10 times, fig. 5). Vibration velocity at r.r.1,2 and 10 is almost the same and with random and determined regimes keeps at the same level only with substantially shorter durability at determined regimes (if curves 3—8 extrapolate downwards). However, in this case the determined regime will distract the test objects much quicker than the "operational" regime (5—20 times).

Conclusion: simulation of a random vibration tests by a determined one is not acceptable.

The jointly presented dependencies V_{av} on t_d of specimens for various regimes give the clearest picture of the objectives of vibration loading as a function of spectrum width (the number of harmonics) (fig. 5). These dependencies similar to a well known analogy can be called "vibration loading curves". Just as fatigue curves, they are different for each regime. The main requirements to regimes of laboratory tests on vibration are presented in work [15].

4. Existence of "band widths" ("natural bands") in mechanical systems. The experimental data given in fig. 3 and fig. 5 can be presented as a function of spectrum width [16]. Then the existence of extremes V_{av} and σ_{av} will be seen at the same Δf. With this spectrum width the t_d, of specimens is minimum. Using computer vibration loading for the same specimens in the domain of 14 natural frequencies was simulated. It appeared that for a beam with amplitude-frequency characteristics (fig. 6) and uniform spectrum loading having a preset value in the natural frequencies n (fig. 7) in the interval between $(n-1)$-th and n-th natural frequencies there are spectrum width values which are called effective (Δf_n^{ef}) leading to max V_{av} with σ_{av}= const and min σ_{av} with V_{av}= const (fig. 8) and min t_d. The values td can be interpreted as "band widths" of a mechanical system and the sense of susceptibility to vibration must be considered as an analog of natural frequencies named "natural bands" of a system.

It can be shown that every system has vibration loading parameter extremes as they correspond to power extremes of the mechanical forces (volumetric and surface) appearing in the system due to random vibration. From the point of view of the dynamic forces theory the extreme of mutual information between input and output signals is determined by the relationship in the sphere of practicability of a real mechanical system with a model system. The model can be considered consecutively as single-frequency, double-frequency, etc. The number of extremes (band widths) is equal to the number of the considered natural frequencies of the mechanical system [14,18,19].

The reliability of the result is confirmed by well known Rice formula (fig. 8) [17].

CONCLUSION

Based on these results the methods of diagnosing materials' fatigue failures (destruction) and vibration tests, as well as the procedure for determining the optimum and most severe vibration, loading regimes with wide-band random effect have been developed and a method of standardizing test regimes can be elaborated, as band widths are determining by parameters of a mechanical systems.

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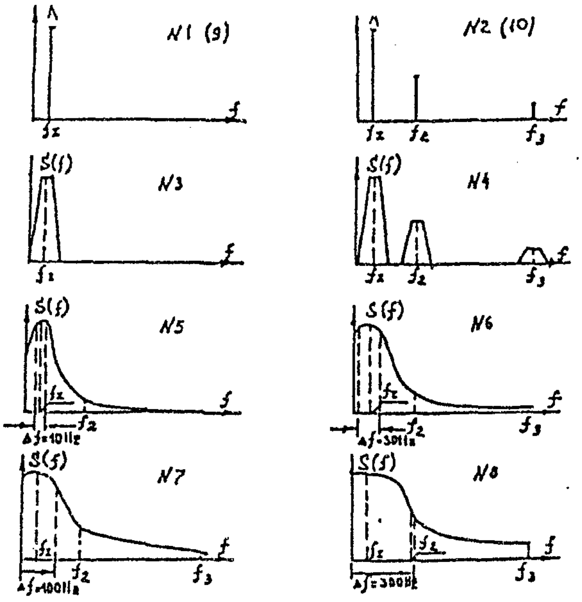
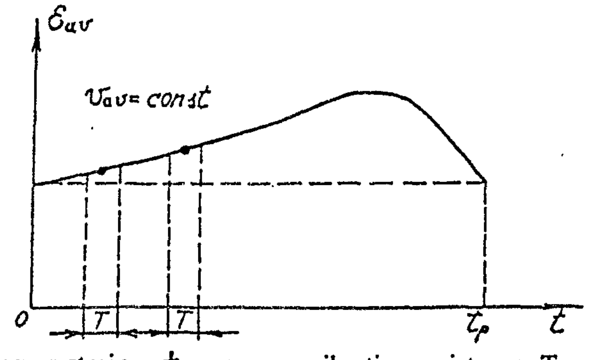


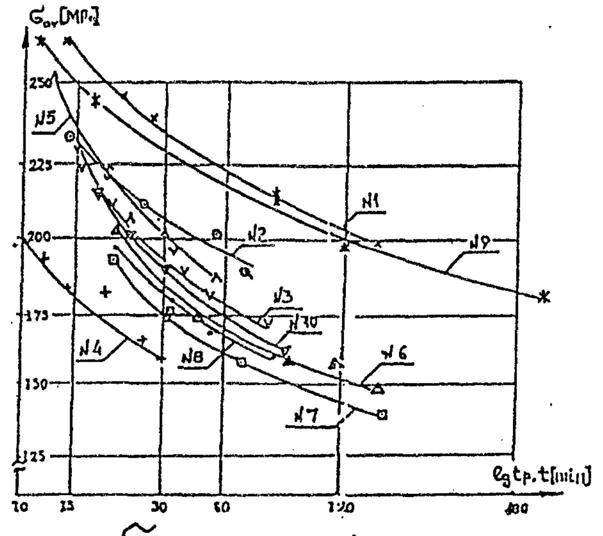
A - amplitude of oscillations, S(f) - spectral density, f_1, f_2, f_3 - the object's natural frequencies, f - spectrum width

Fig. 1



ε_{av} - average strain, t_p - average vibration resistance, T - strain-time measurement

Fig. 2



σ_{av} - average voltage

Fig. 3

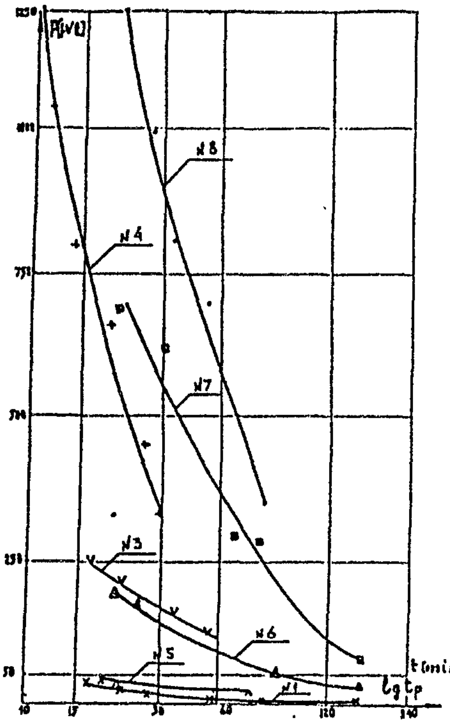
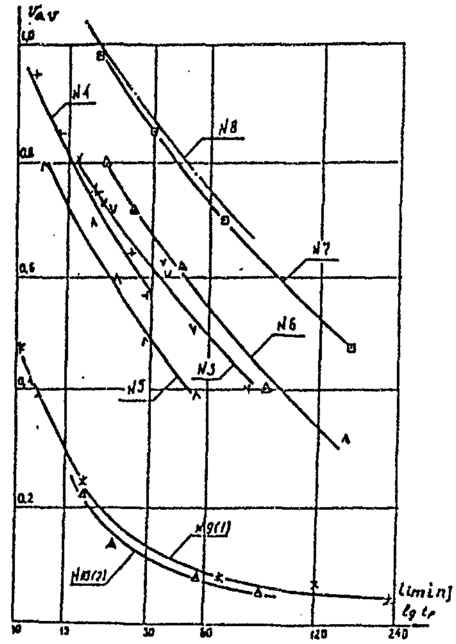


Fig. 4



ϑ_{av} - average vibration resistance

Fig. 5

Fig. 6

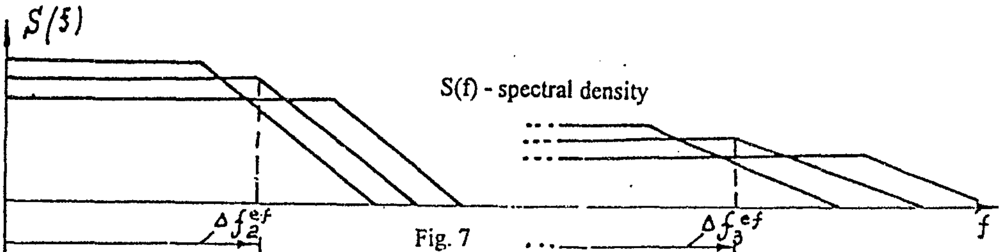


Fig. 7

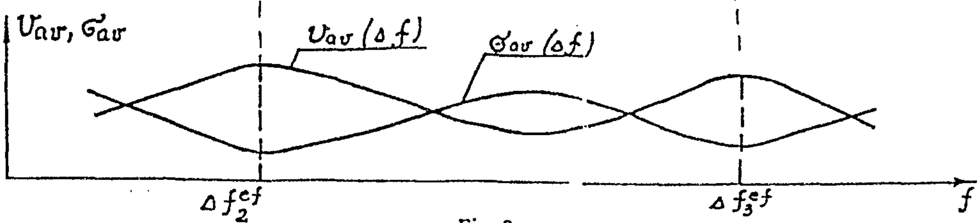


Fig. 8

CERTIFICATION OF AERODYNAMIC EFFICIENCY NOZZLE INSTALLATIONS

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Abstract

Estimation of aerodynamic efficiency of nozzle aircraft installations for the purpose of sertificating is suggested. A method takes into account of nozzle and afterbody external drag including airframe elements and jets interference. This method includes also characteristics of afterbody/nozzle with minimal external drag. This allows to estimate both nozzles and nozzle installations efficiency.

The nozzle installation effects are very important for providing excellent aircraft characteristics. We want to have minimal internal and external nozzle drag, minimal weight, simplicity of design and etc. But we do not know the minimal levels of characteristics in some cases. The best isolated nozzles have effective thrust losses no more then 1—2% in comparison with ideal nozzle. There is a number of difficulties to provide this nozzle performance level when engine is installed on an aircraft.

The problem of design of effective nozzle is a broad topic: internal performances, external and base drag, weight, cost of the nozzle research and design, life cycle and so on. Overall criteria of nozzle efficiency estimation may be passenger capacity, flight range, total fuel consumption and so on. But efficiency of engine installation on an aircraft — effective nozzle thrust (thrust minus nozzle or afterbody drag) — is one of important factors of the all above criteria.

The most complicity and importance of the problem of engine installation acquire on dry power engine regimes at subsonic and transonic speeds. In this case a nozzle throat area is 10—30% of an engine or fuselage cross section and nozzle/afterbody external drag in the engine installation on an aircraft may be highly large.

There are some ways of the presentation of the effective nozzle aerodynamic characteristics: thrust minus nozzle external drag; thrust minus nozzle/afterbody drag; thrust minus nozzle/afterbody fin and tail drag. The first way is more often used but this way does not always give clear picture about nozzle/engine installation efficiency.

Experimental investigations conducted in TsAGI last years allowed to appear main parameters and conditions for estimation and definition of perfection of nozzle/engine/airframe installations. Minimum level of thrust losses is defined at subsonic and transonic speeds. This level is achieved in case attached external flow on a nozzle surface at engine/aircraft installation.

Many interesting results about separation zones on the nozzle surfaces, about their development, influence of the nozzle installation on size of these zones were received. But the main attention in the nozzle aerodynamics is drawn to the possibility of attaining attached flow over external contour of the afterbody with jets, and to the definition of separated and attached flow boundaries. These boundaries for external turbulent subsonic and transonic flows over nozzles (isolated and installed) with and without jets are shown in fig. 1.

Interference of jets and attached external flow led to the favorable effects on external nozzle surface. This favorable interference decrease external nozzle/afterbody drag. High pressure on total external nozzle surface for the afterbody with large contraction ratio increases effective thrust and in some cases this thrust may be more than thrust of ideal nozzle, if presentation of the effective nozzle characteristics carry out the first way.

Experimental investigations shown that maximal effective nozzle thrust of the convergent afterbody with nozzle (effective thrust is gross thrust minus external drag of convergent afterbody from maximum cross section to the nozzle exit in this case) is approximately equal gross thrust at Mach Number $M < 0.9$ and nozzle thrust losses is about 1—2% from thrust of ideal nozzle (fig. 2). There is an attached flow on afterbody/nozzle surface, maximal favorable interference of jet with external subsonic turbulent flow and optimal geometric parameters of afterbody/nozzle configurations. This level of thrust losses is minimum and may be considered as standard. The excellent engine/nozzle installation on an aircraft has thrust losses approximately equal standard losses in fig. 2.

The characteristics of nozzle/engine installation on an aircraft are shown in fig. 3 obtained two ways: thrust losses as thrust minus external nozzle drag and as thrust minus nozzle/afterbody drag at $M = 0.8—0.9$ and pressure ratio $P_{tj}/P_{\infty} = 3—3.5$.

Parameter $A_{noz\ max}/A_{max}$ is ratio of the maximum of the nozzle cross section to the maximum fuselage (nacelle) cross section. Engine/fuselage installation has, as a rule, this parameter 0.5—0.7 and engine/nacelle installation has this parameter more then 0.7.

The punctuate lines in fig. 3 correspond to the level of minimal thrust losses from fig. 2: the first way is thrust minus external nozzle drag (fig. 3,a), the second way is thrust minus external nozzle/afterbody drag (fig. 3,b).

For example, the nozzle/engine/aircraft installation № 1 has $A_{noz\,max}/A_{max} = 1$ and the installation № 2 has this parameter 0.6. Theirs measured thrust losses at the first way are about 2% of thrust of the ideal nozzle. It seems, that the both installations (№ 1 and № 2) have equal aerodynamic efficiency of nozzles if punctuated curve A is absent. Nevertheless, the installation № 1 has not penalties with compared to standard (isolated) nozzle/afterbody and the installation № 2 has the great penalties in the presence of the punctuate curve A in fig. 3,a.

The second way of presentation of nozzle thrust losses shows the great penalties of installation № 2 with compared to the installation № 1 and with compared minimal level of thrust losses of standard (isolated) nozzle/afterbody configuration (the punctuate curve B in fig. 3,b).

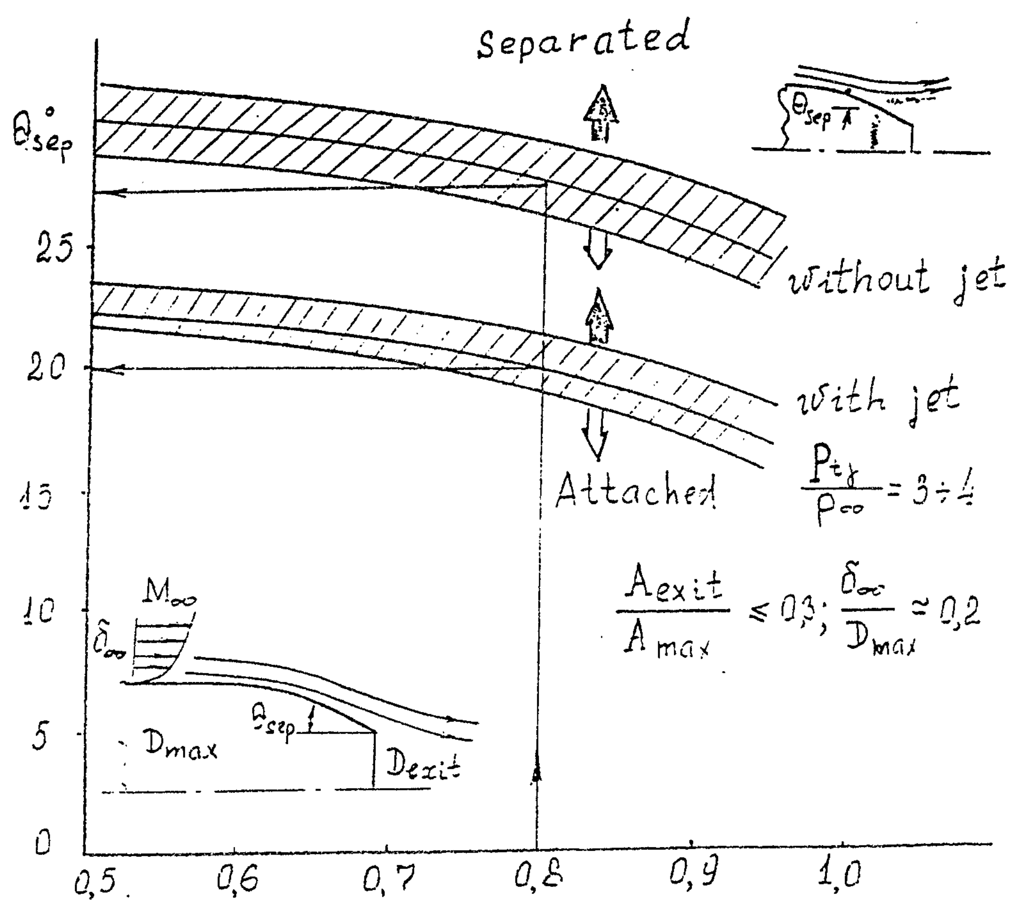

Fig. 1 Separated/attached flow boundaries

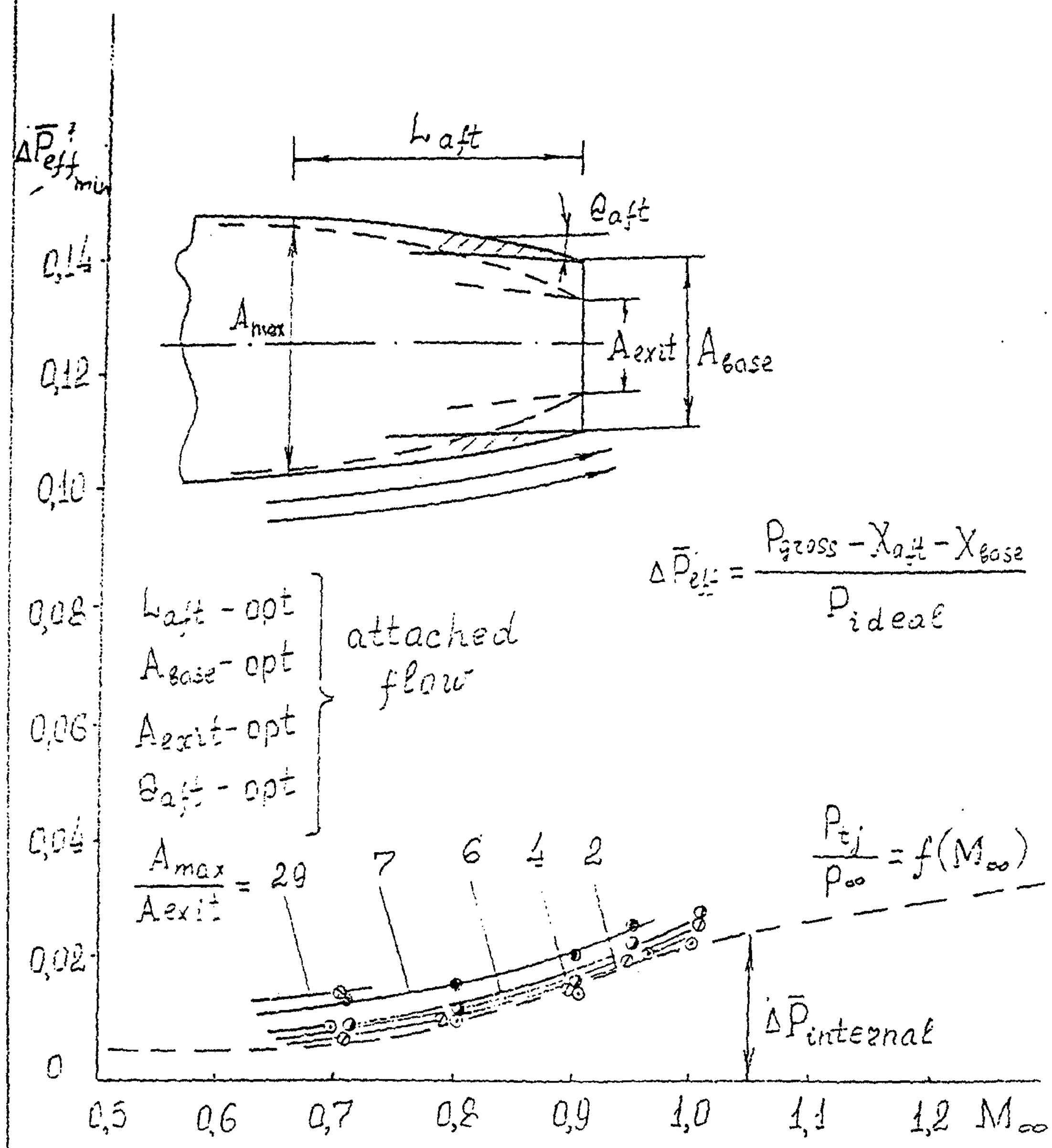

Fig. 2 Minimum nozzle thrust losses

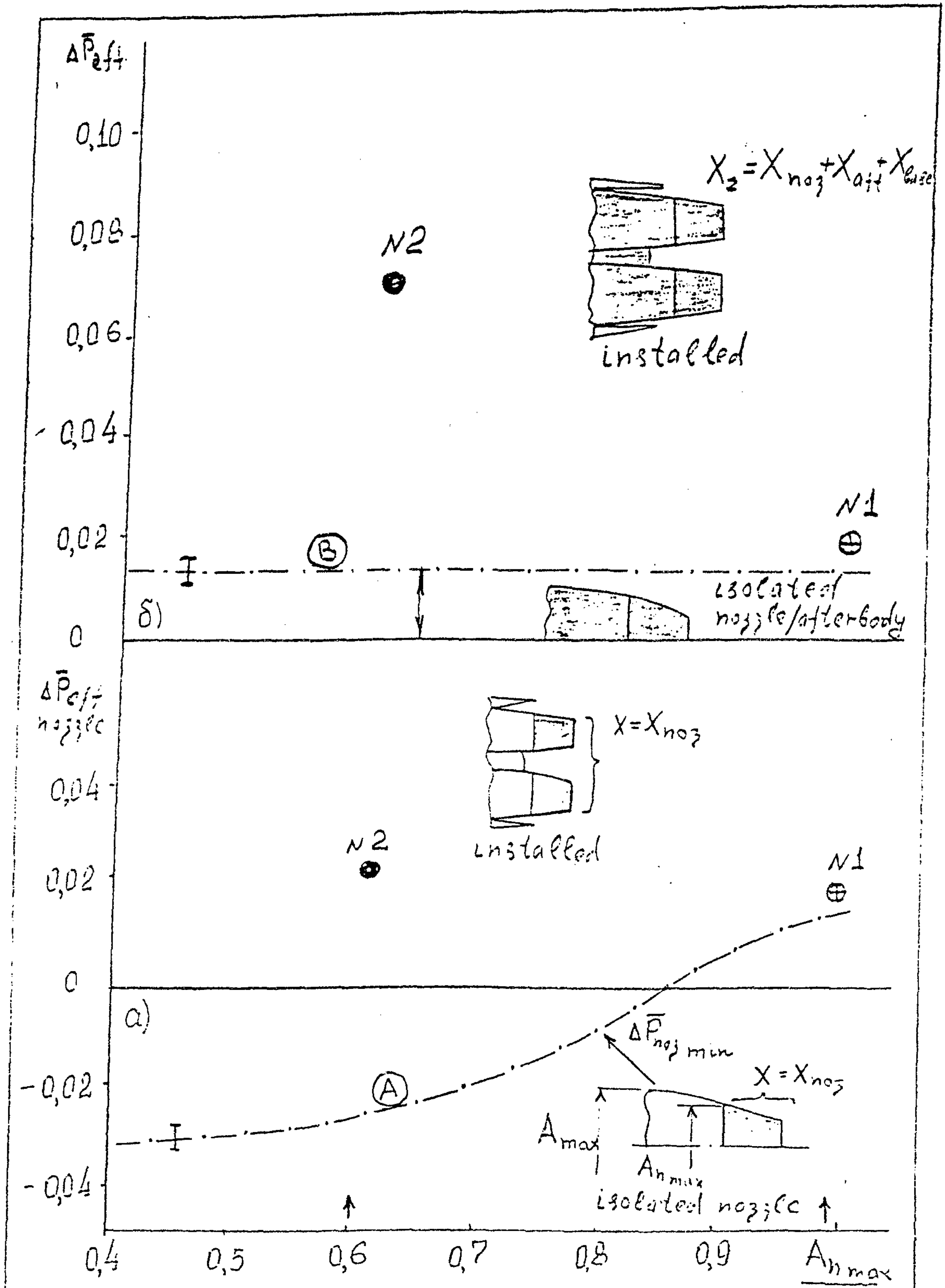

Fig. 3 Nozzle thrust losses.

SECTION 4

Aeromechanics

ASSESSMENT OF RUSSIAN VSTOL TECHNOLOGY EVALUATING THE YAK-38 "FORGER" AND YAK-141 "FREESTYLE"

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BACKGROUND

The dissolution of the Former Soviet Union (FSU) created new relationships between the world superpowers. Overnight, the Commonwealth of Independent States (CIS), formed from the remnants of the FSU, began the difficult transformation to a free market society, Indirectly, one of the world's supreme arsenals became virtual surplus. The chief military customer, the Soviet government, no longer existed and the smaller states could not afford expensive hardware. This shift in philosophy halted major development programs, suspended production lines and forced the design bureaus, now independent "companies" to explore new commercial markets. The design bureaus were no longer subsidized and now forced to deal with the western problems of competition, profit motive and cash flow.

Military hardware that had once been highly classified and the basis for our own defense planning, was now openly marketed at airshows around the world. "Test" flights were available for potential customers and cooperative partnerships were explored between former adversaries. Almost anything and everything could now he bought outright, including much of the FSU's premier achievements of technology. In many respects, it was a buyer's market.

This environment permitted a visit to the Yakovlev Design Bureau, (Yak) for a VSTOL technology assessment. Yakovlev is the FSU's sole Design Bureau with experience in VSTOL aircraft and has developed two flying examples, the Yak-38 "FORGER" and Yak-141 "FREESTYLE". This visit was the first time that westerners were permitted such candid insight into these two previously classified aircraft.

The Yak-38 FORGER became operational around 1975 aboard the carrier KIEV, and is chronologically equivalent to the western AV -8A "HARRIER." However, FORGER is markedly different from HARRIER and utilizes different technology for flight controls and vertical lift. FORGER has two lift engines, used only for vertical flight, embedded in the forward fuselage and a single lift/cruise engine in the aft fuselage for both vertical and wingborne flight. This multi-engine configuration is dubbed Lift plus Lift/Cruise (LLC).

The FORGER, built exclusively for shipboard operation, was originally designed as Vertical Takeoff/Vertical Landing (VTOL) only, but developed a Short Takeoff (STO) mode in response to western criticism of limited capability.

The Yak-141 FREESTYLE is a research and development testbed, designed to succeed FORGER with major advancements in flight controls and performance. It builds on the experience of the Yak-38 and uses a nearly identical LLC configuration with a much larger cruise engine providing a flying supersonic VSTOL aircraft. After its first flights in 1988, FREESTYLE claimed 12 world records for VSTOL aircraft, previously held by HARRIER. Additionally, this aircraft has many features highly integrated into the flight control system to simplify the pilot tasks and reduce workload, especially during VSTOL flight.

FREESTYLE and FORGER were displayed at the Farnborough Airshow 92 and there Art Nalls contacted representatives from the Yakovlev Design Bureau about a visit to study the two aircraft. The initial contact was quite successful, resulting in a complete detailed tour of the aircraft on the flight line including a cockpit orientation. More detailed discussions were held later in the United States and resulted in an official invitation from Mr. Alexancder Dundukov, Chairman and Chief Designer of Yakovlev, for a Navy/NASA team to visit Russia. This initial visit also included familiarization flights in the Yak -38U, two-seat FORGER. These flights were the first time western test pilots would be permitted in any Russian VSTOL aircraft.

DESCRIPTION

Yak-38 FORGER

The Yak-38 is a 3-engine, subsonic, maritime combat aircraft. It has a shoulder-mounted, modestly-swept, low aspect ratio, heavily loaded wing. Two 7800 lb thrust Rybinsk RD-36 dedicated lift engines are mounted in tandem behind the Cockpit and inclined forward 10 degrees. The 18,000 lb thrust Tumansky R-27 main engine flow exits through two low, side mounted, swiveling nozzles with a vectoring range of 90 degrees. A bifurcated inlet with a semi-circular cross-section has a single row of auxiliary inlet doors very similar to HARRIER. Other notable external features include the lift engine inlet and exit doors on the upper and lower fuselage, a flow blocker dam to protect the main inlet from reingestion, ventral strakes to trap the jet fountain and dorsal strakes to separate the jet fountain from the fuselage and protect the lift engines from reingestion. Additionally, the aircraft has a wing fold capability for shipboard operations and a drag chute used for conventional landings (see fig. 1). The flight control system is conventional hydromechanical in the lateral and directional axes and fly-by-wire in the longitudinal axis. It is a triplex, analog system that fully integrates the flight and propulsion systems. The single throttle lever controls the overall thrust level of all three engines in VSTOL flight and a bi-directional, spring-centered switch mounted on the otherwise conventional stick provides thrust vector control. Pitch response type is rate command in conventional flight and converts to attitude command, attitude hold (ACAH) in VSTOL flight. A bleed air reaction control system (RCS) provides lateral and directional attitude stabilization and control as the conventional surfaces lose their aerodynamic effectiveness in VSTOL flight. Differential thrust between the lift engines and the lift/cruise engine is used for maneuvering control and pitch trim in the presence of total thrust modulation and thrust vector deflection. It is remarkable that this fully integrated flight and propulsion system was developed in the early 7O's and operational in 1975.

The cockpit is quite roomy but lacks a built-in ingress/egress system. The canopy is hinged on the starboard side and manually operated. The cockpit field of view (FOV) in the two seat trainer is somewhat limited downward laterally at the canopy sill, which comes to shoulder level (see fig. 2), and aft by the ejection seat head rest. The lateral FOV in the single seat version is better. The KA-36, the Russian standard for ejection seats, is modified with a VSTOL variant and has automatic initiation in VSTOL flight, based on a combination of attitude and attitude rate for both pitch and roll. Automatice initiation of ejection is a concepte completely new to the western world and unique to the Yakovlev VSTOL aircraft. The general layout of the instrument panel is similar to the first generation HARRIER with the engine instruments on the right and flight instruments on the left. Of course, the labeling is in Russian Cyrillic and the units are metric. Notable features include a prominently displayed nozzle angle gage, dual tape lift engine RPM indicators, combination alpha and g-meter, combination VSI and turn-and-slip indicator, and numerous indicator, caution and warning lights. A vertical row of 7 advisory lights on the right hand side of the instrument panel advise of the systems necessary for vertical flight. All seven lights should be illuminated under normal operating conditions for vertical operations. A HUD is conspicuously absent. fig. 3 contains a more complete description and location of the instruments.

The left console contains the throttle quadrant and all other system switches (see fig. 4). The throttle has a rather short throw of approximately 5 inches at the throttle grip and lacks an arm rest to assist with precise throttle inputs. The 3-position lift engine start lever (OFF, STOL, VERTICAL) is located inboard of the throttle and has a large, square knob with a lock mechanism which provides unmistakable tactile cueing. The right console contains the usual communications and navigation equipment and is otherwise unremarkable.

The center control stick and rudder pedals are conventional except that a hand-actuated brake lever is provided on the forward side of the control stick rather than toe-operated brakes on the rudder pedals. Steering is accomplished through differential braking which is commanded by a combination of pedal deflection and brake lever application. The main engine nozzle angle is controlled through a bi-directional switch on the center stick that commands the main engine nozzles at a constant rate in the commanded direction. Aft actuation lowers the nozzles from horizontal, forward actuation moves the nozzles back to the horizontal conventional thrust position. To condition the propulsion system for VSTOL flight, the lift engine start lever is positioned to either STOL or VERTICAL. This automatically opens the lift engine inlet, located on the upper fuselage behind the cockpit, the lower exhaust doors on the fuselage belly, and starts the lift engines. Lift engines are started with main engine bleed air on the ground and airborne, but will automatically air start airborne, if the bleed air system fails. When the lift engines are started, the flight control system automatically shifts to ACAH mode. Lift engine thrust is scheduled by the flight control computers as a function or airspeed, main engine nozzle angle and throttle position. As noted above, lift engine and main engine thrust are differentially regulated for pitch balance and at thrust vector angles greater than 60 degrees (measured from the horizontal) the total thrust is regulated to hold the total vertical component constant. The result is an aircraft with very pleasant handling characteristics in VSTOL flight.

For a STO, the nozzle angle is initially set at 30 degrees (with respect to the waterline) and the lift engine thrust is just above idle (point A, fig. 5). When full throttle is selected to begin the takeoff roll, the lift engine thrust is increased to the maximum corresponding to the 30 degree nozzle deflection (point B, fig. 5). At the proper speed, based on measured longitudinal acceleration and ambient temperature, the flight control computers automatically increase the nozzle deflection to 60 degrees and the lift engine thrust to maximum (point C, fig. 5) and the aircraft becomes airborne. As the aircraft accelerates through a speed of 250 km/hr (135 kts) the pilot adjusts the nozzle angle to 45 degrees and holds it there until a speed of 300 km/hr (162 kts). After that, the nozzles are adjusted aft and the lift engines are shut down manually. Lift engine operation is limited to 3 minutes for cooling considerations.

Yak-141 FREESTYLE

The Yak-141 is a prototype, supersonic, carrier-based fighter intended to replace the Yak-38 (see fig. 6). Aircraft development is currently suspended, due to lack of funding and commitment by the military. The propulsion concept and control scheme are similar to the Yak-38, but several of the pilot functions are more highly automated and refined and a much larger cruise engine provides a supersonic dash capability. The shoulder-mounted wing is moderately swept and highly loaded (greater than 120 lbs per sq. ft at projected combat weight). Twin vertical tails are boom mounted and straddle the main engine nozzle. The fixed inlet is bifurcated with initially rectangular cross-section and auxiliary inlet doors on the upper and side surfaces. Lift engine inlet and exit doors are mounted and operate in the same fashion as on the predecessor. A forward flow dam is located aft of the lift engine exit to trap the Jet fountain and minimize reingestion. Side-mounted strakes, a wing root leading edge extension and the sharp upper corner of the main engine inlet serve to separate the jet fountain from the fuselage and protect the lift engines from reingestion.

The main engine, designated R-79 and of the 35,000 lb thrust class, was developed by the Soyuz Design Bureau and incorporates a single vectoring nozzle that permits afterburner operation at any thrust angle. Significant attention was paid to stability of operation during rapid thrust changes as RPM and afterburner fuel flow are modulated simultaneously in VSTOL flight. Thrust vectoring is accomplished by means of two counter-rotating "stove pipe" sections between the core and afterburner and the vectoring range is from -10 degrees to 95 degrees with respect to the aircraft waterline (see fig. 7). The rotating nozzle is convergent only and when rotated down is very close to the landing surface, creating a very harsh ground environment. Surface requirements are reported no more stringent than for the non-afterburning Yak-38 which requires either a tile or steel surface, however very little testing has been done in this regime.

The Rybinsk Design Bureau was responsible for the two 9200 lb thrust RD-41 lift engines, which were adapted from a conventional, horizontally mounted engine. Modifications to convert the engines to vertical lift and minimize weight included the elimination of the high pressure fuel pump (high pressure fuel is supplied by the main engine, replacement of the closed-loop lubrication system with a light weight, sacrificial oil injection system, replacement of the mechanical starter with a bleed air impingement start system and installation of the nozzle system with vectoring capability of ± 12.5 degrees. The combination of the vectoring capability and the installation angle of 10 degrees forward provides a total vectoring range of 2.5 degrees forward to 22.5 degrees aft, as measured from the vertical (see fig. 8).

The flight control system is nearly identical to the Yak-38, except for improvements to automate STO functions and reduce pilot workload. It remains a triplex, analog, FBW in the longitudinal axis, and hydromechanical in the lateral and directional axes, although a digital system was reported to have been in development. As before, all engines are integrated into a single throttle and the thrust vector angle of the main engine is controlled through a thumb actuated switch on the control stick, similar to the Yak-38. A schematic of the longitudinal flight and propulsion control system is presented in fig. 9. The only mechanical link in the entire system is the throttle to main engine and it is used for backup. The response type is similarly ACAH in VSTOL flight using differential engine thrust for pitch control and trim. A bleed air RCS provides control in the lateral and directional axes. The first aircraft had a tail boom ejector RCS for directional control, which was detailed in "Aviation Week." However, directional control was reportedly poor and the second flying prototype was improved with a nose-mounted RCS which provided greater control power through a longer moment arm.

The cockpit of the Yak-141 is similar to the Yak-38 in many respects. It is equally roomy with a starboard hinged, manually operated canopy and the KA-36 ejection seat headrest remains an obstruction to rear FOV, although the lateral FOV is improved through a lower canopy sill. The general layout of the instrument panel is the same and consists of all round gages. Reportedly, the instrument panel was borrowed "off the shelf" from a MiG-29 to save prototyping costs for the first flying model, with plans for a modernized "glass cockpit" for the second and subsequent models. Warning, caution and status lights are quite numerous, with nearly 50 lights scattered throughout the main panel. A HUD was planned for the aircraft, but in its place is a large panel with a flight test airspeed indicator and a small array of indicator lights, likely associated with test instrumentation. The instrument panel and the location of the instruments is presented in fig. 10.

On the left console, the throttle has been raised and operates in linear fashion on a rail mounted on the cockpit sidewall. Total throw is approximately 10 inches, with the last 2 inches dedicated for afterburner modulation. A small gate must be passed to select afterburner and must be intentionally de-selected to prevent inadvertent canceling of afterburner during critical VSTOL operations. This was reportedly the cause of the second prototype sustaining a very hard shipboard landing, resulting in strike damage and an automatic ejection of the pilot. The VSTOL selector knob is again located inboard of the throttle and operates in the same manner described for the Yak-38 (see fig. 11). The usual communication and navigation equipment is mounted on the right console. The control stick and rudder pedals are conventional as noted for the Yak-38 and nozzle angle is commanded by a similar bi-directional switch on the stick. Hand lever operated wheel braking is retained but nosewheel steering through the rudder pedals is added and invoked through an additional button on the stick. A flight control "recovery" mode is provided whereby the aircraft recovers to straight-and-level flight and is summoned via a panic button on the stick.

The propulsion system is configured for VSTOL in the same manner and the lift engine start sequence is identical to the Yak-38.

For vertical flight the two lift engine nozzles are toed-in to prevent a jet fountain forming between them and causing hot gas ingestion of the lift engine cavity. Lift engine thrust is scheduled as described above and pitch is controlled through differential thrust but since the main engine is operated in afterburner for VSTOL flight, the ground environment is extremely harsh. The STO is accomplished in like manner to the Yak-38 through liftoff, however in addition to the manual accelerating mode, the flight control system can automatically control the main engine nozzle to achieve an optimum accelerating transition. In this automatic transition, nozzle angle is reduced in increments, of approximately 10 degrees, and aircraft airspeed and altitude referenced by the flight control computer to prevent a settle. Although supersonic performance is achieved and good handling qualities are retained, the landing environment is extremely hostile, due to the afterburner, and is a major deficiency.

FLIGHT PROFILE

The flights of the Yak-38U two seat aircraft were conducted at the Zhukovsky Flight Test Center from the Ramenskoye airfield. Zhukovsky, a community of approximately 100,000 people located approximately 20 miles southeast of Moscow, is the center for Russian aerospace research and development. major facilities include the Central Aero-Hydrodynamic Institute (TsAGI), the Gromov Flight Research Institute (LII), the test pilot school which is currently closed, and the test bases of the various aircraft design bureaus.

The first flight was flown the morning of 30 June by Art Nalls and Viktor Zabolotsky, the Chief pilot at LII. After a short turnaround of less than 30 minutes, Mike Stortz and Yuri Mitikov, the Yakovlev Design Bureau test pilot, completed a similar flight profile. The profile was chosen by the Russians and consisted of a conventional takeoff, a decelerating transition to a hover, followed by an accelerating transition to conventional flight and limited wingborne maneuvering flight. Recovery was accomplished via straight-in to a conventional landing. Vertical landings were precluded by a reported lack of a prepared, metal landing surface, The entire flight profile was approximately 30 minutes in each case.

The weather was fine with broken clouds at 15—2500m (5—8k ft) and 25—5000m (8—16k ft). The temperature was 20 deg Celsius (68 deg F) with light winds and unrestricted visibility.

START/TAXI

The aircraft was started by the front seat pilot and with a minimum of post start checks was ready for taxi in under 2 minutes. Taxi is straight forward and ground handling is surprisingly simple with the stick mounted brake lever and differential brakes. Despite a lack of nose wheel steering, tight turns were easily accommodated, although nose wheel steering would certainly be desirable for shipboard operations.

TAKEOFF

Maximum RPM for takeoff was 98/98% representing fan and core speeds. RPM appear to be matched above approximately 95%. Rotation was at 300 km/hr (162 kts) and unstick occurred at 350 km/hr (189kts). Takeoff roll took 17 seconds and was estimated at 3000 feet, as no runway markers were available. Gear and flap retraction was smooth and quick. Shortly after takeoff we were briefly permitted some very limited maneuvering before returning for a hover. First impressions of the aircraft highlighted a lack of control harmony. The aircraft was very heavy in pitch, requiring an estimated 20 lbs for a 3—4 g turn. Yaw control was also very heavy and the aircraft appeared directionally loose, yet very responsive in roll. After only a few minutes, the demonstration pilot Look aircraft control and returned to the reciprocal of the takeoff heading and positioned for a decelerating transition.

DECELERATING TRANSITION AND HOVER

At 600 km/hr (324 kts) and at about 3 nm straight in the landing gear were lowered, followed by starting the lift engines at 450 km/hr (243 kts) by moving the yellow cockpit lever to VERTICAL. Lift engine start is completely automatic, from opening the inlet and exit doors, starting engines,and reconfiguring the flight controls to ACAH laws. Pitch is surprisingly stable, with no tendency to "bobble" with the additional lift from the lift engines. Drag from the inlet doors is significant and unmistakable. At approximately 1.5 miles from the hover position, and still on a slight glideslope, the nozzles were lowered via the stick control switch. Deceleration was comparable to the AV-8A in a flat attitude and attitude was held constant throughout the transition. A slight flare brought the aircraft to a stabilized hover. In the hover, mild maneuvering was permitted in pitch, roll, yaw, and altitude. A stabilized hover was easily controlled. Pitch response was sluggish, but reasonable since pitch is controlled with differential engine thrust. Pitch is so stable that minor hover adjustments fore and aft, normally made with pitch in the HARRIER, were ineffective in FORGER and perhaps could be better performed with small nozzle nudges. Roll control, with RCS, is responsive and predictable, however yaw control is sluggish, underpowered and the major deficiency of hover control, although reportedly improved in later models. Surprisingly, height control with a single throttle controlling 3 engines was smooth and responsive. Overall, the aircraft felt smooth and stable in hovering flight with very low workload.

ACCELERATING TRANSITION

Since the lift engines are limited to three minutes of operation, we ended the hover with a 180 degree turn and initiated an accelerating transition on the original runway heading. The transition was accomplished in increments. The lift/ cruise engine nozzle was moved to 25 degrees (measured from the vertical) and held there until 250 km/hr (135 kts), then moved to 45 degrees until 300 km/hr (162 kts) was achieved. The change from 45 degrees to horizontal was made without further airspeed restriction. The lift engines were secured via the cockpit VSTOL lever and with gear and flap retraction the aircraft was configured for conventional flight. The entire transition took approximately 3O seconds.

MANEUVERING FLIGHT

Recommended maneuvering airspeed was 700 km/hr (405 kts), with an maximum airframe limit of 900 km/hr (486 kts). Several wingovers were performed, letting airspeed drop to as low as 300 km/hr or less over the top and controls remain effective in all axes through at least this range with no sideslip buildup. Accelerated stall warning is light to moderate airframe buffet around 4—5 g's (the rear cockpit g meter was inoperative) developing into mild wingrock as the stall progresses. Recoveries are instantaneous with relaxation of backstick. № l-g stalls were performed. Roll performance is very good, exceeding 300 degrees per second, although adverse yaw is very evident. Roll attitude capture requires a large opposite stick "check", but is overall satisfactory.

LANDING

At the completion of the maneuvering phase, a conventional landing was made from a straight in approach. Landing gear was lowered at 600 km/hr (324 kts) with the flaps following at 500 km/hr (270 kts). A long, shallow approach was performed with a reference airspeed on final of 350 km/hr (189 kts). The aircraft appeared to be quite loose about all axes and large, frequent stick inputs were required to maintain course and glideslope. Speed stability was good, as few throttle inputs were required to maintain speed. Crossing the threshold, the throttle was retarded to idle and a gentle flare to landing at approximately 300 km/hr (162 kts) was accomplished. Stick activity increased markedly during the flare, requiring frequent, nearly continuous large (near full) stick deflections. Touchdown was positive, but there was some "lurching" and roll control inputs were required to maintain wings level during the initial part of ground roll. The drag chute was deployed at approximately 275 km/hr (149 kts) and the deceleration was very effective. Wheel brakes were applied without limitation below 200 km/hr (108 kts) although there was no mention of an anti-skid system.

TAXI, SHUTDOWN, POSTFLIGHT

Taxiing with Differential brakes was again quite easily accomplished, in spite of a hairpin turn into the staging area. Large power increases were required to keep the aircraft rolling during the turns. The aircraft was secured almost immediately after arriving in the chocks and cockpit postflight appeared minimal.

GENERAL IMPRESSIONS AND CONCLUSIONS

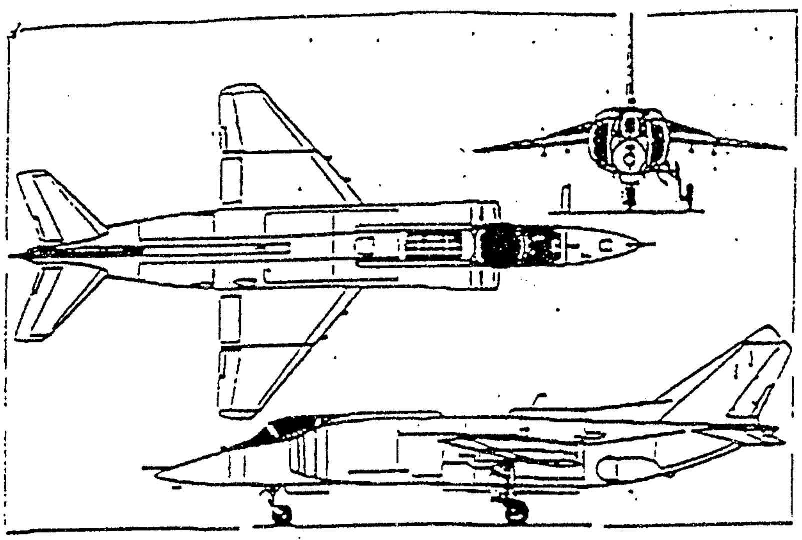
The handling qualities of the Yak-38U in VSTOL flight are excellent - low workload because of the attitude hold response type and high precision because of the superb integration of the flight and propulsion systems. Maneuvering performance was only adequate and wingborne handling qualities leave a lot to be desired.

The integration of the flight and propulsion systems for the Yak 38 was ahead of its time. The concept of lift thrust scheduling as a function of airspeed, lift/cruise nozzle and throttle is elegant in its simplicity. For its vintage (mid 197O's) analog, fly by-wire system of integrated flight and propulsion controls is a remarkable accomplishment. Credit is owed to the Russians for their ingenuity and persistence in making this complex system operational with such docile manners.

At the risk of oversimplification, the Russian design philosophy with respect to VSTOL flight appears to be a solution basked on brut force with only minor refinements in the development process. Short takeoff procedures were developed to minimize the ground environment problem but no consideration has been given to alternate propulsion configurations.

In the spectrum of powered lift configurations, the Yakovlev aircraft provide valuable information on dedicated lift systems and the ground environment of thrust augmented lift systems. The Russians have made tradeoffs to acquire a supersonic configuration with a significant vertical payload. The downside of the tradeoff is a logistical penalty for multiple, dissimilar engines and a harsh ground environment that carries its own logistical requirement.

In summary, the experience afforded by this Russian VSTOL Technology assessment will guide us as we develop the next generation of powered lift, jet aircraft.



Length - 50.8 ft
Wing span - 24 ft
Height - 14.3 ft
Wing Area - 199 sq ft
Maximum Takeoff
Weight - 25,795 lbs

Figure 1. Yak-38 Three View—
(from Jane's All The World's Aircraft)

YAK-38U

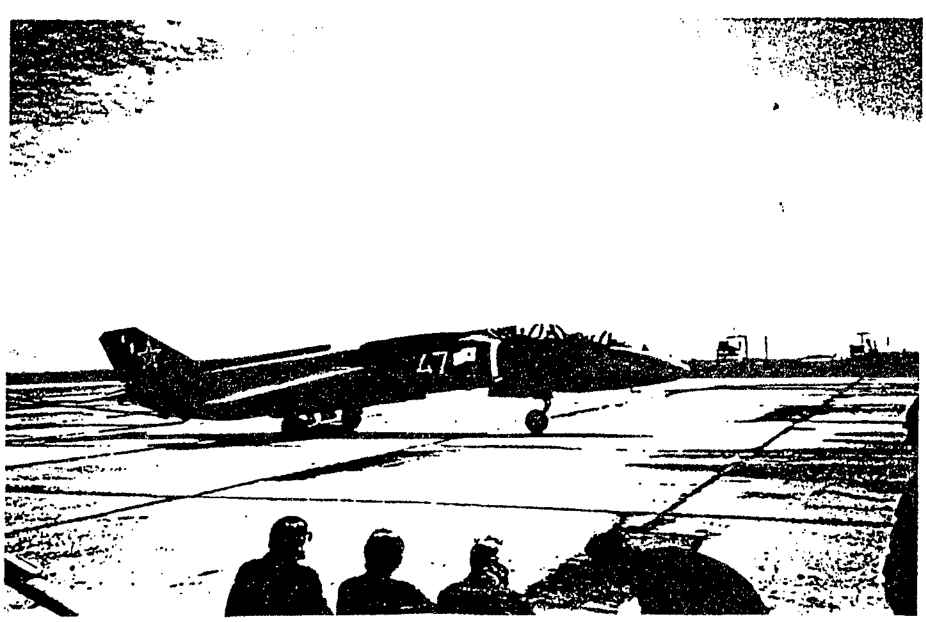
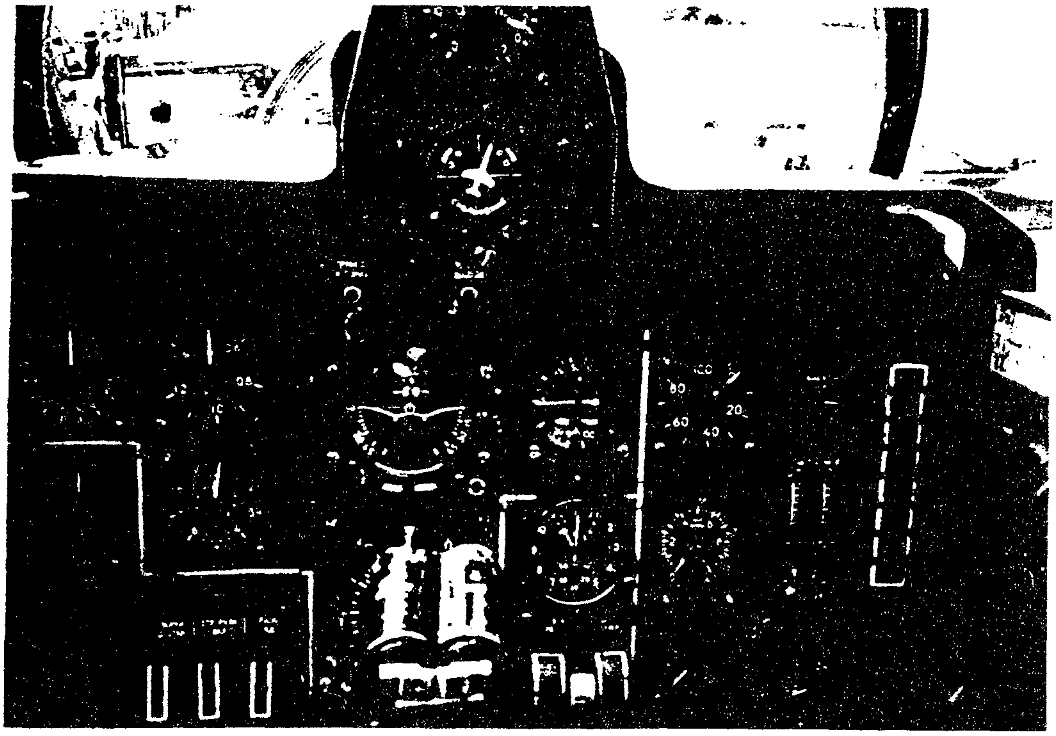
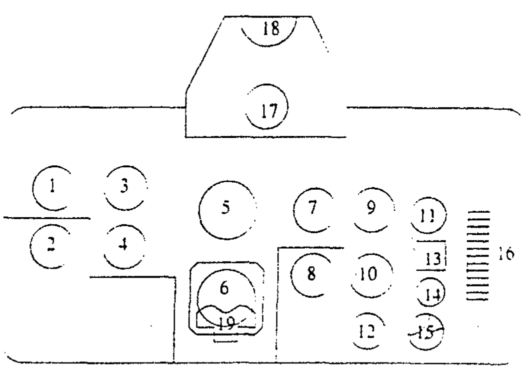


Figure 2. Yak-38 Three View—
(from Jane's All The World's Aircraft)

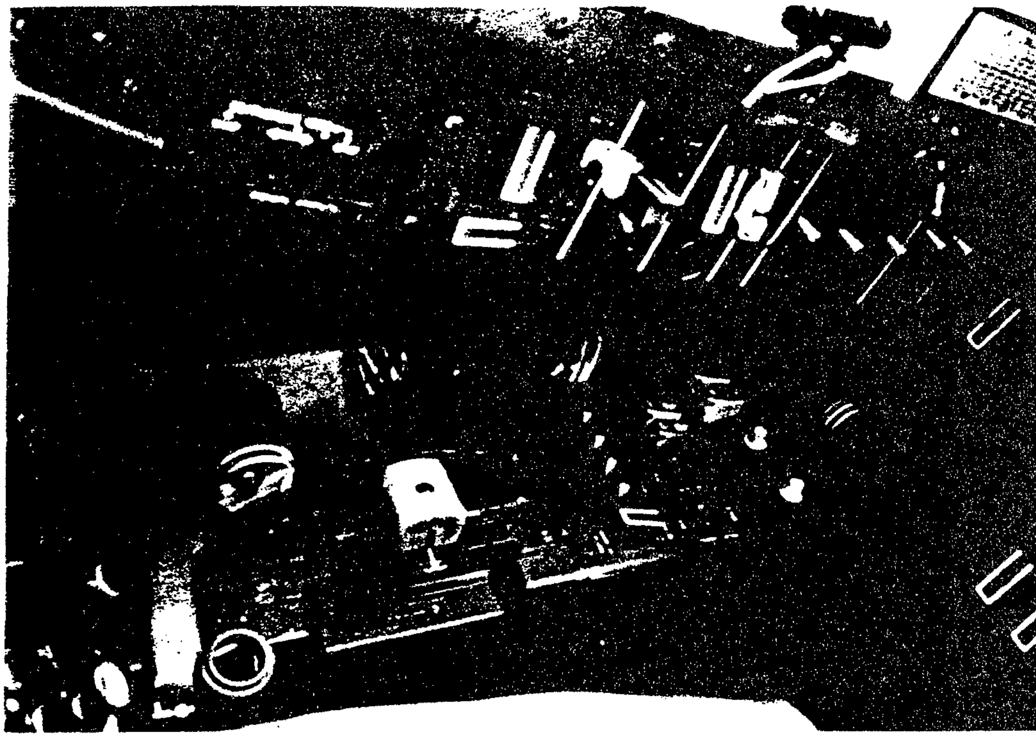


Legend



1. Airspeed Indicator
2. Radar Altimeter
3. Mach Meter (?)
4. Altimeter
5. Main Attitude Gyro
6. Horizontal Situation Display
7. Vertical Speed Indiactor
8. Clock / Stopwatch
9. Engine Tachometer
10. Fuel Gage
11. Nozzle Indicator
12. Exhaust Gas Temperature
13. Lift Engine RPM (?)
14. Lift Engine Temperature
15. Hydraulic System Pressure
16. VSTOL Advisory Lights
17. Sideslip Indicator
18. Combined G Meter / AOA Indicator
19. Nozzle Control Button On Stick

Fig. 3. YAK-38 Intstrument Panel



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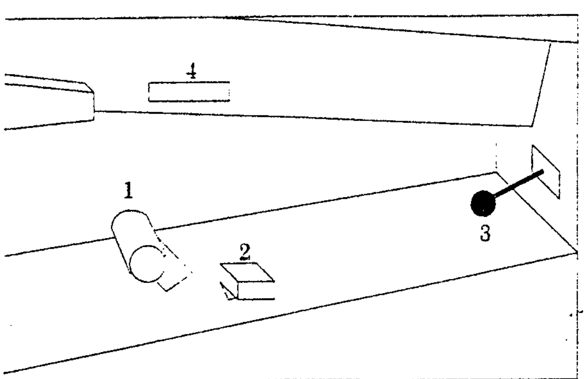

1. Throttle
2. VSTOL Mode Selector
3. Gear Selector
4. Canopy Control

Fig. 4. YAK-38 Left Console

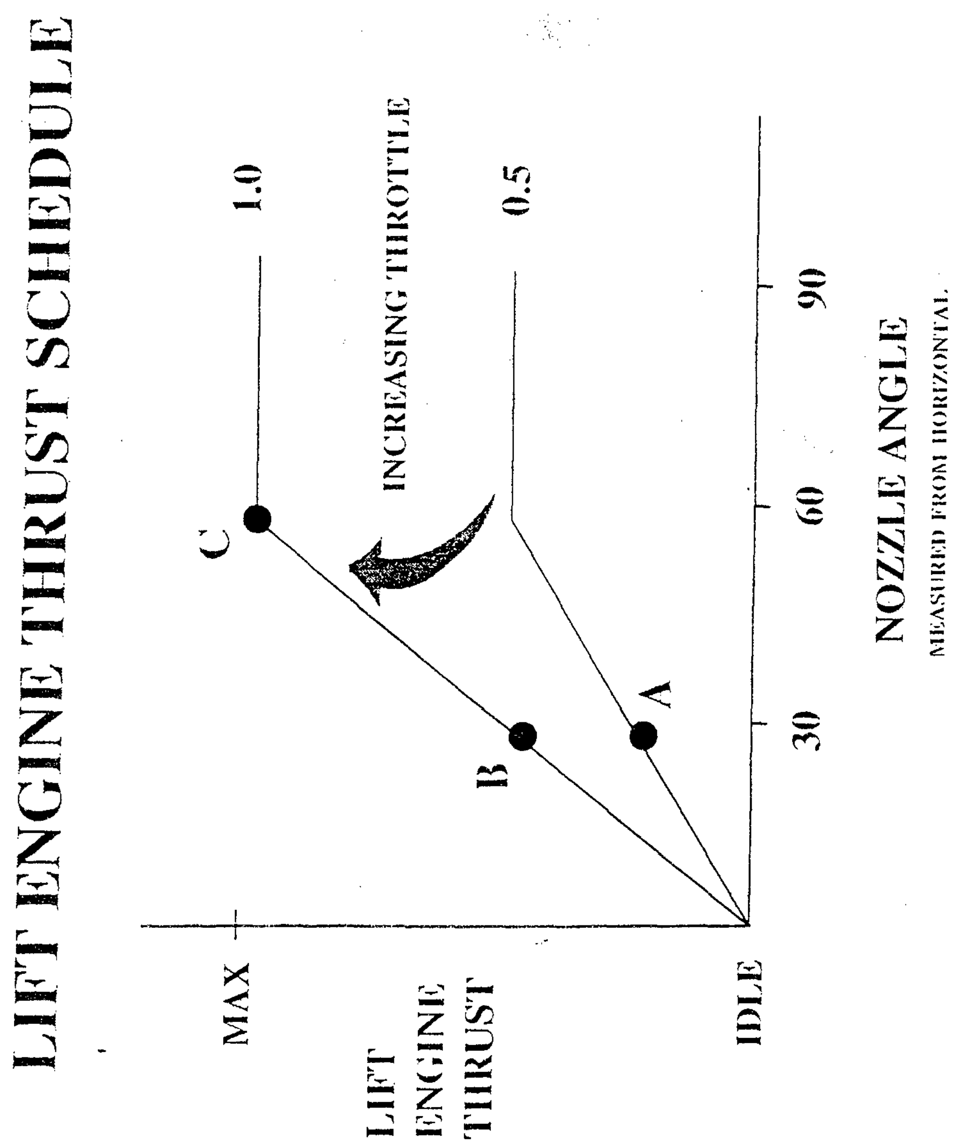
LIFT ENGINE THRUST SCHEDULE
MAX
LIFT ENGINE THRUST
IDLE
B
C
A
1.0
0.5
INCREASING THROTTLE
30
60
90
NOZZLE ANGLE
MEASURED FROM HORIZONTAL

Figure 5.

Length - 60 ft
Wing span - 33.14 ft
Height - 16.4 ft
Wing Area - 322.9 sq ft

Operational Weight Empty - 25794 lbs
Maximum VTO Weight - 34760 lbs
Maximum STO Weight - 42900 lbs

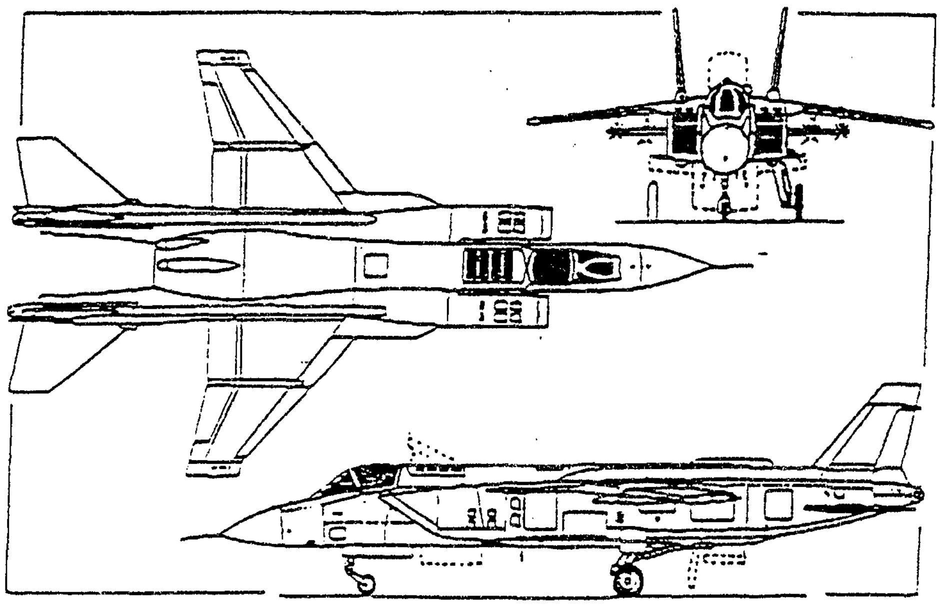

Figure 6. Yak-141 Three View
(from Air International)

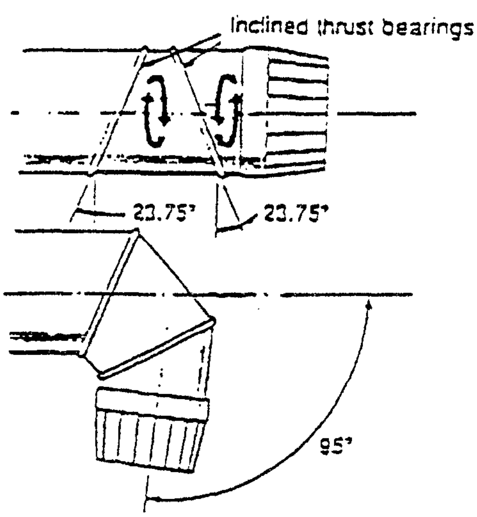

Figure 7. THRUST VECTORING NOZZLE OF THE YAK-141

POWERPLANT CONFIGURATION

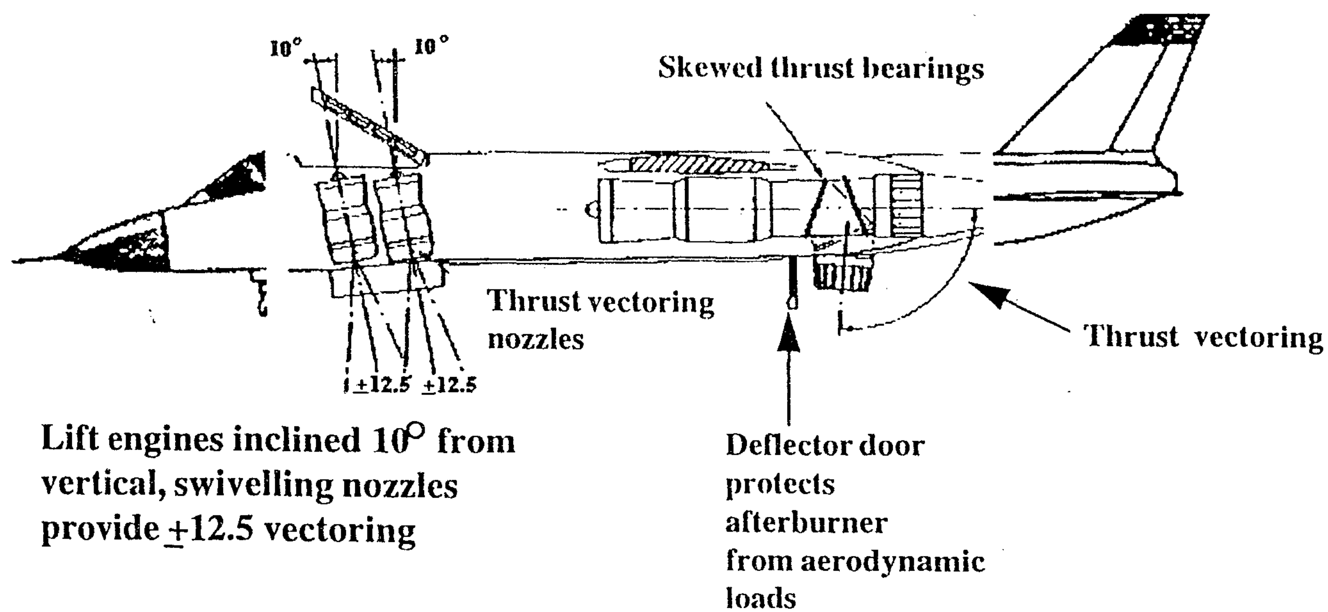

Figure 8

YAK-141 LONGITUDINAL FLIGHT/PROPULSION CONTROL SYSTEM

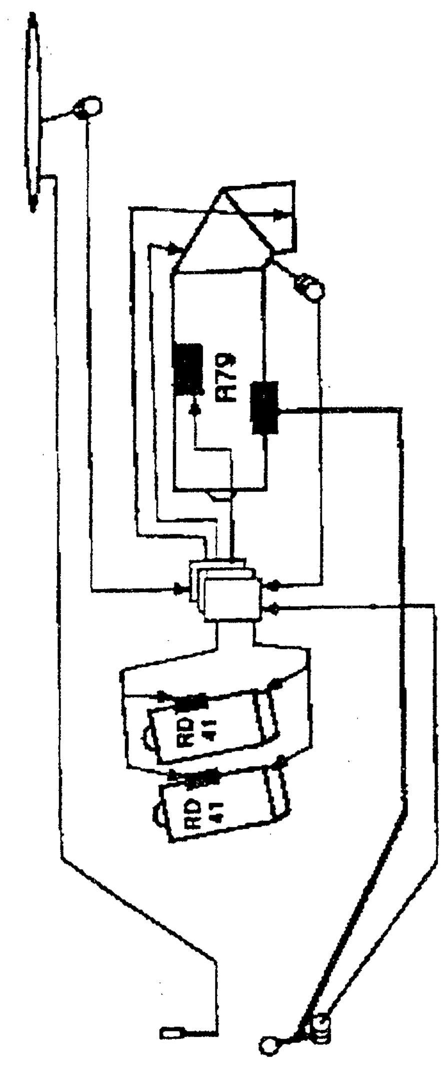

Figure 9

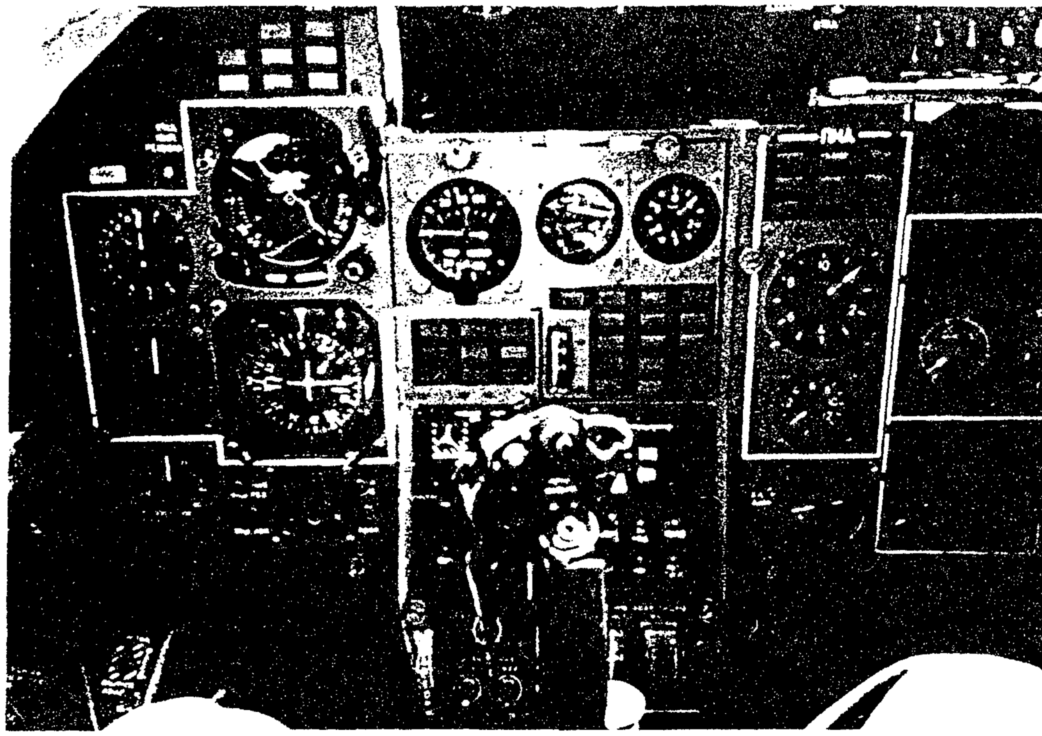

Figure 10. YAK-141 MAIN INSTRUMENT PANEL

Figure 11. YAK-141 LEFT CONSOLE

PARTICULARITIES OF NATIONAL ACCREDITATION OF AERODYNAMIC TEST LABORATORIES

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Abstract

Considered in this paper are the problems of development of legal, organizational/methodical and normative/technical documentation for the accreditation of aerodynamic test laboratories carrying out certification tests.

TsAGI disposes of a unique experimental research base including a complex of various wind tunnels and gasdynamic facilities with sub-, trans-, super- and hypersonic flow speeds. These TsAGI experimental facilities meet stringent requirements of fluid dynamics. There are wind tunnels surpassing similar facilities of Europe and the U.S. in many respects.

Solving scientific and applied problems in a complex manner at TsAGI has made it possible to establish a comprehensive data bank and use it in the design bureau of this country's aerospace industry. TsAGI's wind tunnels provide a high trustworthiness of experimental results. The provision of close agreement and repeatability of experimental results from testing in different wind tunnels is one of the most important problems. In this connection TsAGI begins the work to prepare experimental laboratories for national accreditation with the goal to carry out in its facilities certification tests.

Nowadays the requirements for production quality have been noticeably tightened abroad and some countries have developed national standards setting requirements for quality control systems. For adjustment and checking of quality control systems, the International Organization for Standardization (ISO) has approved a series of international standards, allowing the assessment of product/service quality control systems by independent agencies and their certification.

Since 1994, the "Aviation Engineering and Civil Aviation Objects Certification System" has been functioning in Russia. The System is developed by the Aviation Register of the Intergovernmental Aviation Committee in conjunction with the GOSSTANDART of Russia and includes the Rules of test laboratories accreditation. These documents are harmonized with the ISO standards. In foreign practice the ISO standards are a universally recognized base for national quality control systems. They have greatly influenced quality control methods and widely applied in economics of most countries. The ISO standards are used when concluding international contracts.

The evaluation of quality control systems by independent bodies (third party) and certification of quality control systems provide confidence in a product, process or service to meet a certain standard or another normative documents.

The world's practice of certification tests has clearly proved its efficiency as principal means of governmental quality control consisting in specifying one or several characteristics of a product, process or service being tested according to a prescribed procedure.

Wind tunnel testing is an important stage of development of aviation materiel, to which the requirements of certification systems should be applied. Such tests are exemplified by specific requirements for the characteristics of test medium being modeled, measurement and computational techniques. Certification tests should be carried out only at accredited test laboratories. Such laboratories must have the status of a corporate body. The laboratories' organizational and administrative structures should provide the independence of testing and preparing their results.

Any pressure on the laboratory's personnel should be excluded. If a laboratory is a structural unit of an organization, there should be a special clause in its regulations that stipulates its independence.

1. The accredited test laboratory must have:
- regulations determining the nomenclature of products and/or types of testing;
- test equipment certificate;
- composition, structure and functions of the laboratory;
- test procedures;
- status and responsibility of the laboratory's top executives;
- established rules of interactions with other bodies, organizations and enterprises during test activities.

2. Requirements for competence of the laboratory's personnel, test equipment and measurement technique composition, and work organization must correspond to a number of ISO standards determining:
- general requirements for the assessment of the technical competence of test laboratories (ISO 25);
- general requirements for the acceptance of test laboratories (ISO 38);
- guiding regulations of test results presentation (ISO 45);
- guiding regulations of the development of a quality control manual for test laboratories (ISO 49).

3. The personnel of a test laboratory must correspond in terms of composition, education and qualification to the tasks performed by the laboratory:

— duty regulations of the laboratory's workers specify their functions, status and responsibility, requirements for quality of work, education, knowledge of the job;
— the workers performing tests must be attest in the framework of acting attestation procedures;
— systematic improvement of professional skill should be organized.

4. The laboratory must dispose of documentation including:

— standards regulating technical requirements for products under testing and test methods;
— test programs and methodologies for specified types of product;
— service (usage) forms and records, maintenance documentation;
— quality control manual;
— time schedules of metrological certification and checking of measuring devices and certificates of measuring equipment;
— test methodologies;
— service documentation;
— documentation of a storage system for test results information.

5. The logistic base of the laboratory should correspond to set types of testing. It must include:

— measuring and checking equipment;
— test equipment;
— auxiliary means and devices;
— materials needed for testing.

6. The accredited test laboratory is obliged to:

— ensure a specified testing accuracy;
— maintain in appropriate condition test equipment, measuring means, provide their timely attestation and checking;
— observe specified or agreed dates of testing;
— establish an order of signing and approval of test protocols.

In accordance with these requirements a package of documents was developed for the accreditation of aerodynamic test laboratories, which incudes seven blocks:

1. Legal documentation;
2. Documentation for the products under tests;
3. Documentation for test equipment;
4. Documentation for measuring means;
5. Organizational/methodological and normative/technical documentation;

6. Staff documentation, technological and operational instruction for the personnel;
7. Methodologies of measurement and processing of test results;
8. Documentation on the registration of test results.

Concurrently with the package documentation's development a complex of investigations was carried out for development of methods and means to ensure reliability of aerodynamic test results. The basis of a modern test laboratory is the aerodynamic test complex including two major components:

— test equipment (wind tunnel, measuring and control means, metrological support);
— methodological means, i.e., algorithms for control and processing of measured information, test procedures, an archive (data base).

In accordance with the foregoing, goals were established and requirements were developed for metrological attestation of wind tunnels, measuring means and a methodological support of such equipment. An industry standards and reference devices base was created, which provide the transmission of units to measuring means in aerodynamic experiments.

Basic units in aerodynamic experiments are:

1. Force and mass;
2. Excess and absolute pressure;
3. Temperature;
4. Electrical units.

For reproducing and storing the unit of airflow speed, TsAGI together with VNIIM GOSSTANDART have established a state reference complex for a flow speed range of 10 to 100 m/s (GOST 8.542-86). Work has been performed to create reference means for measuring speeds up to $M=4.0$

The current composition of TsAGI's metrological base provides the possibility of conducting metrological attestation and periodical checks of the great majority of measuring means used during aviation materiel testing as well as in manufacturing processes.

For a number of units, TsAGI's reference devices have reached the level of working standards and are verified at GOSSTANDART's institutes of Russia using state reference devices. The rest of measuring means are reference devices and gauges of the highest grades verified in GOSSTANDART's institutes with the help of working standards.

An important goal of the preliminary preparation for accreditation is the development of unified (in terms to form and structure) measurement methodologies and test procedures. A carefully developed, experimentally verified and attested methodology, when strictly applied, provides the specified accuracy of test results and their convergence and reproducibility in repeated tests.

In the accreditation of a test laboratory all scientific experience and traditions stay as if in the background, but they should be transformed in strict metrologies precisely addressed to concrete performers. A common type of basic methodologies used in aerodynamic tests is developed:

— methodologies of metrological attestation of test and measuring equipment;
— methodologies of testing standard instruments;
— methodologies of type aerodynamic tests.

The unified structure of test methodologies includes four parts:

1. Requirements for test conditions;
2. Material support of testing;
3. Information about a method for obtaining a test result;
4. Test results.

With the aim of making the requirements for certification test more stringent, a system for quality control of test results was developed with the use of unified reference model groups. The system includes a computer analysis of results obtained, takes into consideration differences between results being obtained and reference ones, and in accordance with specified criteria, provides a conclusion about the state of the measurement technique.

When using reference characteristics the level of the test complex's state is assessed according to the following GOST 8.381-80 with the following parameters:

— accidental error, root-mean-square deviation of a measured result;
— uncorrected systematic inaccuracy relative to an reference characteristic;
— instability of a measured result during the period between checks.

The results for a check model reflecting the shape of a fighter aircraft, which was tested in a TsAGI low speed wind tunnel in 1989-90, show that the level of uncorrected systematic errors often exceeds the level of a measured result's root-mean-square deviation and the evaluation of the state of the wind tunnel measurement complex should be carried out with consideration for both factors.

Based on the considered developments, packages of documents for the accreditation of a number of TsAGI's test centers were developed.

The proposed documents may be used for accreditation of laboratories/centers of other aviation technologies.

ON THE USE OF TSAGI'S T-101 AND T-104 WIND TUNNELS FOR CERTIFICATION OF LIGHT GENERAL AVIATION AIRCRAFT.

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Activities are currently expanding in Russia for development of light aircraft to transport passengers and cargo and to perform other tasks as well. All aircraft of such a class before entering service must undergo certification flight tests according to the АП-23 requirements, which involves considerable time and financial expenditures.

To reduce the duration and amount of the certification flight tests of light aircraft, it is recommended to use the TsAGI T-101 and T-104 wind tunnels for testing full-scale aircraft. The tests in these wind tunnels allow one to obtain reliable aerodynamic characteristics, stability and control characteristics in takeoff, landing and cruise flight regimes with the operating powerplant for the entire range of angles of attack and slip angles, including post-stall ones.

According to the world-wide statistics, most of aircraft flight accidents and incidents (approximately 70%, see fig. 1) occur during takeoff and landing. Because of this, in developing aerodynamic layouts of general aviation aircraft it is extremely important to know aerodynamic characteristics, stability and control qualities with the operating powerplant in these regimes. Scaling lift and pitching moment data from model tests in the T-102 wind tunnel (model scale range of 1:10 to 1:5) to flight conditions can yield certain errors (fig. 2). A major concern relating to flight safety is the icing of the wing, tail unit and control surfaces.

As an example, fig. 3 shows the effects of ice simulators installing on the straight wing and empennage of a short-haul aircraft. These results were obtained by testing a mockup of this aircraft in the T-101 wind tunnel.

It can be seen that in the case of the flaps deflected for landing, the ice simulators significantly decrease the maximum lift and stall angle of attack and degrade longitudinal stability characteristics, especially at negative angles of attack.

The most reliable results on the influence of icing on aerodynamic, stability and control characteristics can be obtained by testing full-scale aircraft in the T-101 wind tunnel.

In addition to global aerodynamic characteristics, in the T-101 wind tunnel one can obtain loads acting on the aircraft s components (control surfaces, high-lift devices), stability and control characteristics, powerplant characteristics as well as pressure distributions. Emergency situations can be modeled too.

Currently, the nominal flow speed range in the T-101 wind tunnel is $V = 5—50$ m/s (to 180 km/hr), the maximum flow speed can reach 65 m/s, the dynamic pressure is up to $q = 156$ kgf/m, the maximum Reynolds number is $\mathrm{Re} = 3.4 \cdot 10^6\ \mathrm{m}^{-1}$, the angle-of-attack range is $\alpha \leq 40°$, the slip angles are within $\beta = \pm 180°$.

The dimensions of aircraft to be tested in the T-101 wind tunnel are limited by:

— wing span ≤ 16 m;
— wing area up to 30 m^2;
— aircraft length ≤ 18 m.

The mass of aircraft tested on the T-101 aerodynamic balance should not exceed 8000 kg.

These capabilities of the T-101 wind tunnel allow one to explore all takeoff/landing characteristics and partially or fully cruise flight regimes for practically all aircraft to be certified according to АП-23.

Results of testing a full-scale aircraft in the T-101 wind tunnel yield data needed for computation of technical/performance characteristic for mathematical modeling and testing on piloted simulator.

For calculating technical/performance characteristics, in addition to aerodynamic data it is necessary to know the aircraft s powerplant characteristics. As is generally known, computational assessment of powerplant characteristics may contain noticeable errors.

Losses of propeller thrust on an aircraft, obtained with various methods, can vary from 5 to 20 percent. This can results in noticeable errors in determining a number of performance characteristics. As an example, fig. 4 shows the takeoff distance, takeoff run and vertical speed as function of installation losses of propeller thrust.

In should be noted for the development of dynamic models forming a part of piloted simulators, the measured results obtained for an aircraft in the T-101 wind tunnel must be supplemented with data on unsteady aerodynamic characteristics, especially for the acrobatic category of aircraft flying at high angles of attack. In the final analysis the results of mathematical and pilot-in-the-loop simulation must be confirmed by proof tests under flight conditions.

Of great importance are the tests of aircraft in the T-101 wind tunnel for obtaining trustworthy aerodynamic loads, determining airframe strength.

Strain-gauge measurements during wind tunnel tests to study loads on the propeller of a normal category light aircraft may exclude the need in the same measurements in flight.

Prior to installation of an aircraft into the T-101 wind tunnel a corresponding preparation must be made.

In testing full-scale aircraft with operating engines in the T-101 wind tunnel, the aircraft s control can be fully simulated. For this purpose TsAGI department № 1 has developed and prepared for the "Mercury" aircraft a system to remotely control the deflectable surfaces and powerplant consisting of two piston engines with propeller.

This aircraft installed on the AB T-101 balance is connected with control consoles and an information/measurement system (fig. 5) through an electric command link. From these consoles, aircraft controls are commanded, and a mechanism in the cockpit provides required deflections of the control surfaces.

A computer-aided feedback monitors the control deflections, their hinge moments and forces on the levers in the cockpit. This provides a real picture of the forces and moments acting on the aircraft.

Another console is used to control the powerplant. Located on this console are the elements of the pilot s instrument panel related to the powerplant and the thrust lever.

Information about propulsion system parameters are indicated on the console and storaged in the computer.

The standard electric mechanisms of the МП-100, УР-10 and МПК-15 type are used as devices for actuating aircraft controls. Forces and moments are measured with strain-gauge transducers.

Fuel supply for the powerplant is provided from the aircraft s fuel tank with a capacity of 20 liters. Fuel consumption being large, an independent fuel supply system can be used. The aircraft must be equipped with a fire-suppression system.

The use of the T-104 wind tunnel is exemplified with the results of an experimental investigation of the small-sized remotely-controlled "Shmel" aircraft.

TsAGI s T-104 wind tunnel has an open-jet test section with circular cross section 7 m in diameter. The flow speed is up to 120 m/s.

The aircraft tested is of conventional tail-aft configuration with a rectangular wing and circular fuselage. A ducted propulsor with the contoured shroud and two-blade, fixed-pitch pusher propeller is mounted on the aft fuselage.

The П-032 piston engine is installed inside the fuselage ahead of the propeller. The finned cylinders projecting beyond the fuselage contours are covered with the cowling having a passage for cooling air.

Aircraft pitch and yaw control is accomplished with the aid of the plane elevator and rudder mounted within the propeller duct just behind the propeller.

The aircraft tested was installed in the wind tunnel on a pyramidal strut using a special holder (fig. 6). The aircraft angle of attack α was varied in discrete manner using a special device located on the base of the V-shaped holder while the slip angle was varied through the cabin of the wind-tunnel balance about the vertical axis.

Aerodynamic forces and moments acting on the aircraft were measured with the six-component mechanical balance. Engine and aircraft control surfaces were controlled remotely from the console.

Fig. 7 shows the values obtained of the resulting force P acting in the X-direction as a function of the flight speed V at various modes of engine operation-takeoff, cruise and idling. The resulting force P equals the difference between the thrust of the propulsor as an aircraft s constituent and the aircraft drag with the operating propulsor.

ΔP values are measured immediately with the tunnel balance and can be used for determining aircraft performance, in particular, the aircraft vertical velocity V. Also shown here are the values of the propeller speed n_p corresponding to the ΔP values obtained. At takeoff, that is at full-throttle conditions the fixed-pitch propeller speeds up as the velocity increase.

From the resulting force values obtained, an effective thrust of the aircraft propulsor was estimated. The effective thrust of the propulsor as a constituent of the aircraft was found to be 17 — 20% less than the thrust of the isolated propulsor. Such a drop in effective thrust is mainly, caused by the interference between the ducted-propeller propulsor and the airframe: the propulsor is immersed in the flow behind the fuselage, engine, ets.

The effective thrust is under the effect of the deflections of the controls located within the contoured duct and the effect of varying angle of attack (fig. 8). The elevator deflection to $\delta_e = 20°$ in the regime of $V = 0$, n_p = 6600 1/min reduces effective thrust by about 7.5%. An increase in angle of attack also leads to decreasing effective thrust of the propusor to lesser extent the larger is the angle of attack $\alpha_{f.d.l}$.

At the same time, the presence of the operating propulsor results in changing aircraft aerodynamic characteristics. Fig. 9 illustrates the comparison of aerodynamic characteristics of the aircraft with operating propulsor and without it. The changes of the flow around the duct and the controls induced by the propeller lead to increase in the derivative $dC_{y\alpha}/d\alpha$, production of a significant nose-down moment and increase in static margin.

The 1/2-scale model of the aircraft was tested before in the T-102 wind tunnel. From the experimental data presented in fig. 9 one can see a substantial discrepancy between drag characteristics: the model drag less by half than that of the full-scale aircraft. This results primarily from the unfavorable condition of the full-scale aircraft surface.

The flight-test results are in good agreement with the data of testing the aircraft in the T-104 wind tunnel (fig. 9).

The operating ducted-propeller propulsor influences the effectiveness of the elevator and rudder-mounted within the contoured shroud. Fig. 10 shows the variation of the c_m increments with the elevator deflection angle δ_e. With the presence of the propeller the elevator effectiveness is 4—9 times greater, at V = const it being the greater the greater is propeller rotational speed.

From the examples presented it follows that full-scale flight data for small sited aircraft with operating engine can in some instances appreciable differ from the data of model testing in wind tunnels of the T-102 type.

Conclusion

Tests of light aircraft with operating engines in the T-102 and T-104 wind tunnels allow obtaining credible aerodynamic, stability and control characteristics, and propulsion system characteristics over the wide operating angle-of-attack range including a post-stall region, which provides an improvements in flight safety and can result in decreasing the volume and period of certification flight tests.

Conditions of flight accidents

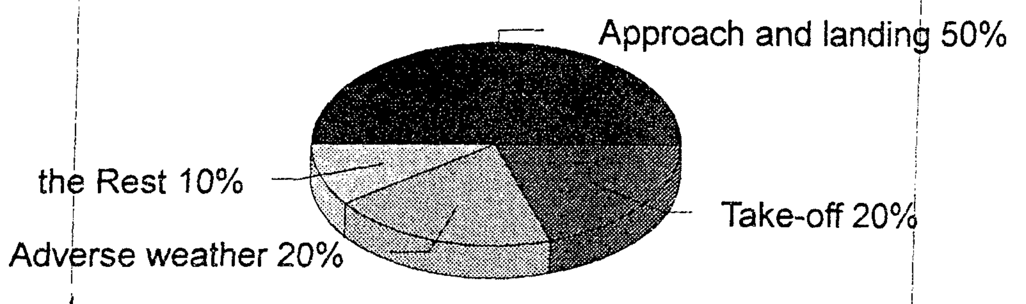


Fig. 1

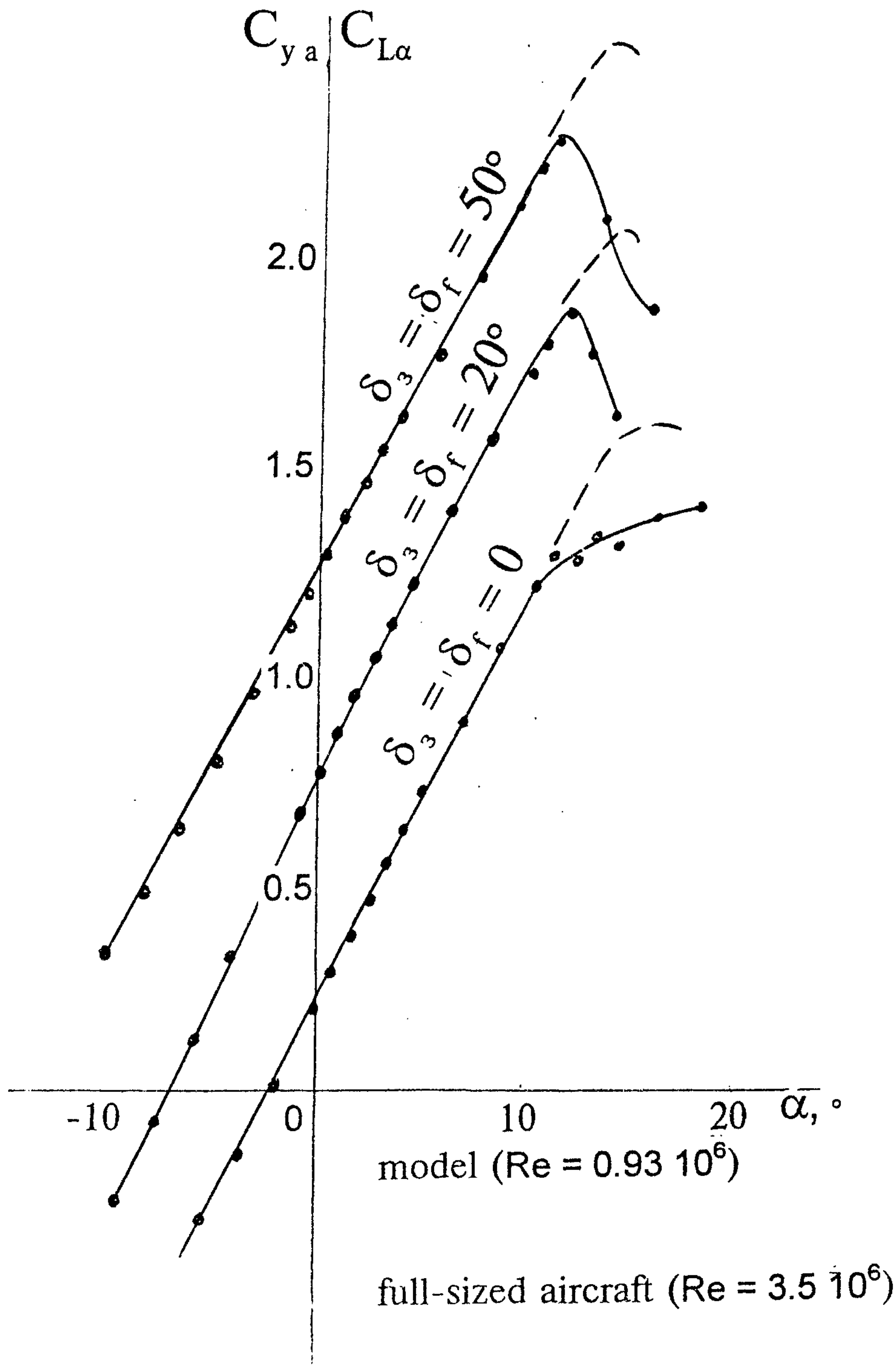

C_{ya} $C_{L\alpha}$
2.0
1.5
1.0
0.5
-10
0
10
20
α, °
$\delta_3 = \delta_f = 50°$
$\delta_3 = \delta_f = 20°$
$\delta_3 = \delta_f = 0$
model (Re = 0.93 10^6)
full-sized aircraft (Re = 3.5 10^6)

Fig. 2

Large-scale aircraft model

Scale 1:2 Масштаб 1:2

$$\Lambda_{l.e} = \chi_{п.к} = 5°20', \delta_f = \delta_з = 30°$$

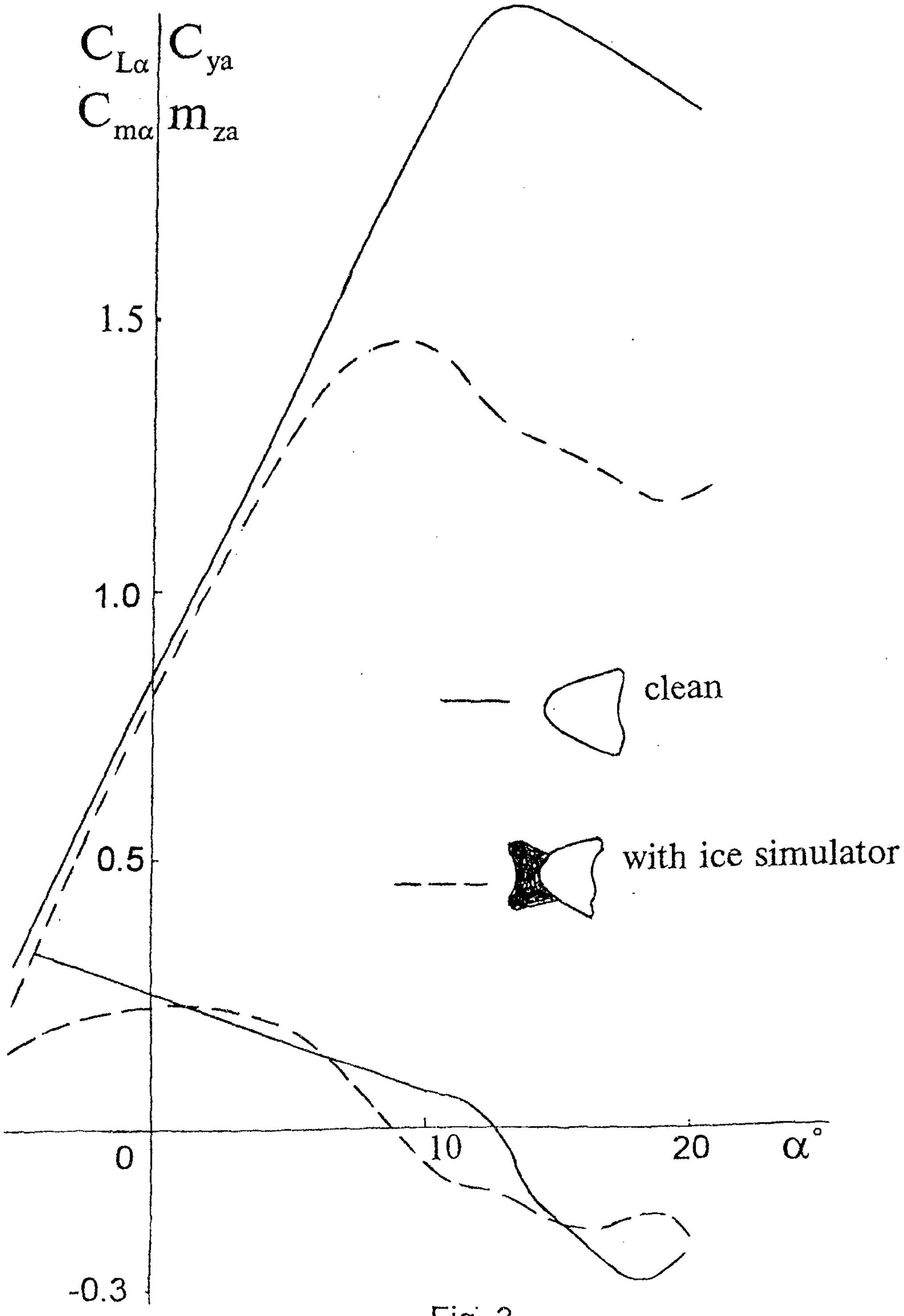


Fig. 3

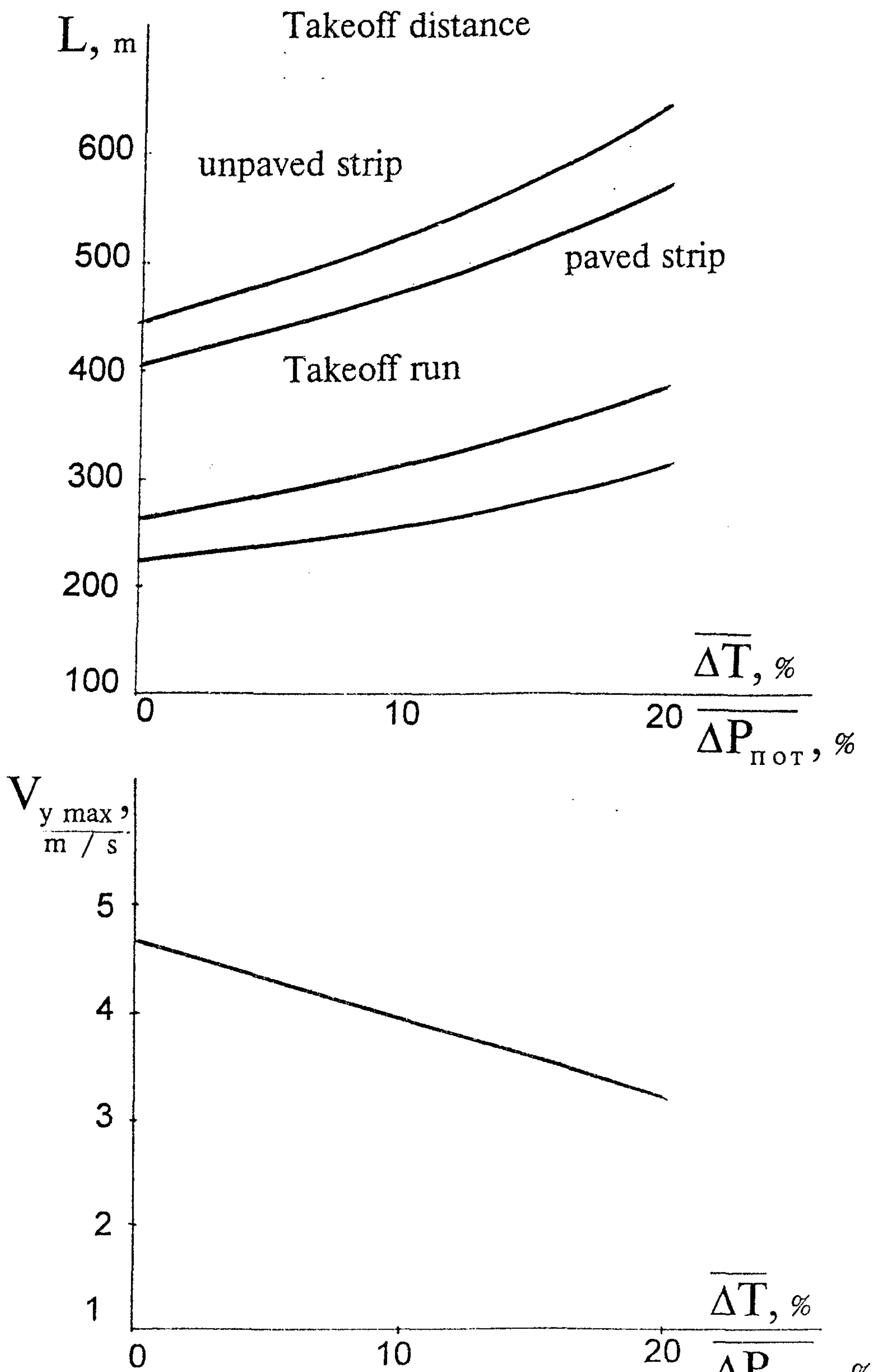

L, m
Takeoff distance
600
500
400
300
200
100
unpaved strip
paved strip
Takeoff run
0
10
20
ΔT, %
ΔP_пот, %
V_y max, m / s
5
4
3
2
1
0
10
20
ΔT, %
ΔP_пот, %

Fig. 4

Test of aircraft with the operating powerplant in the T-101 wind tunnel

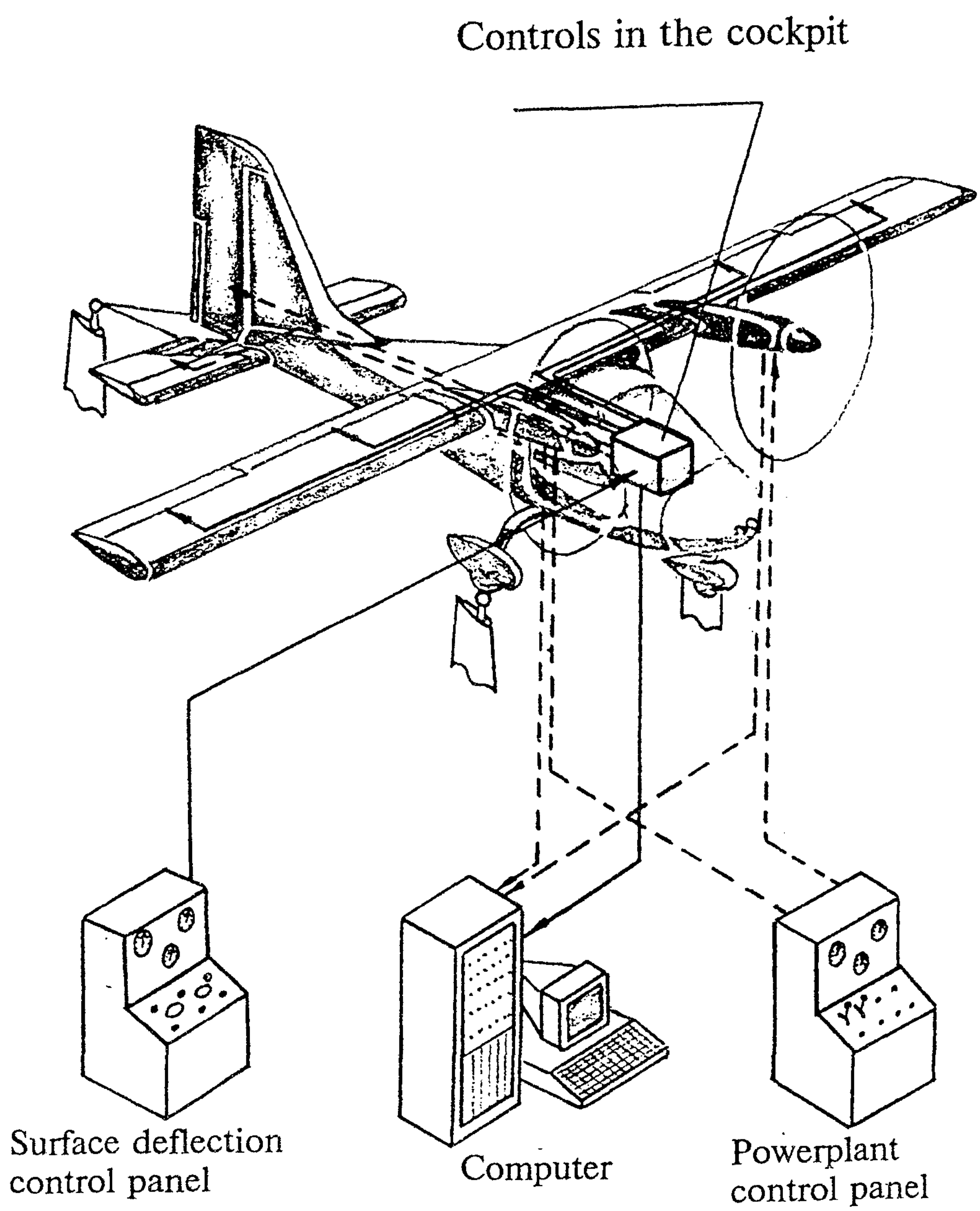


Fig. 5

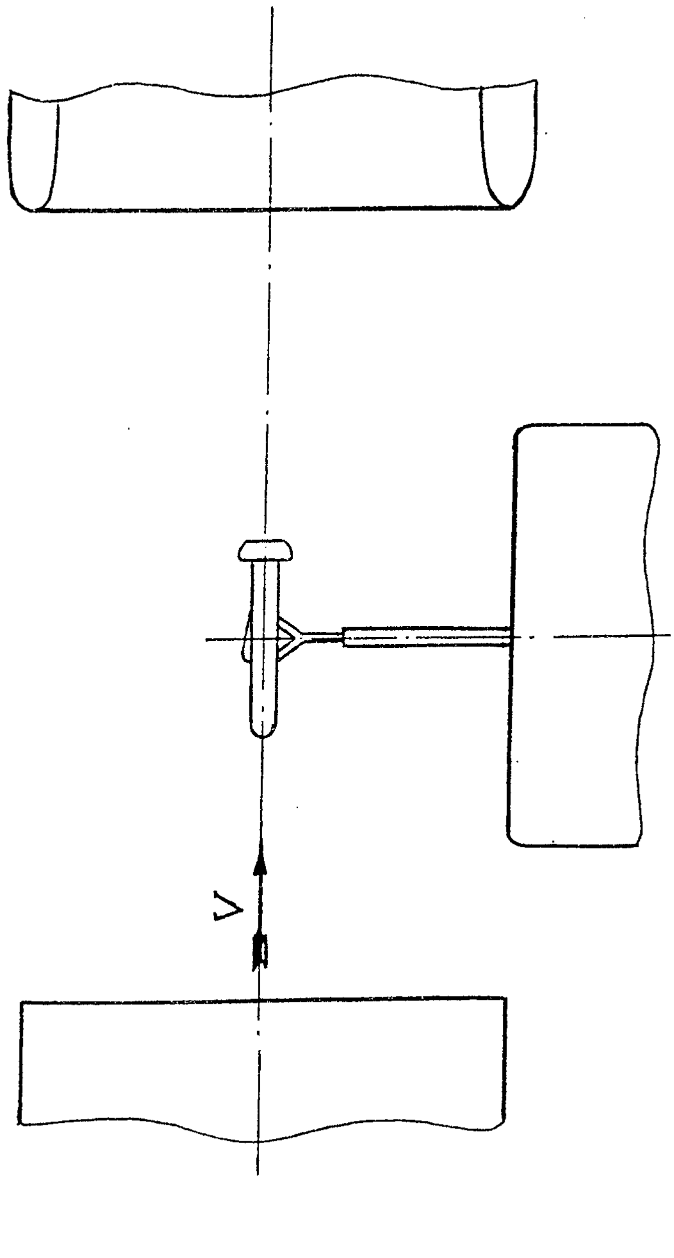
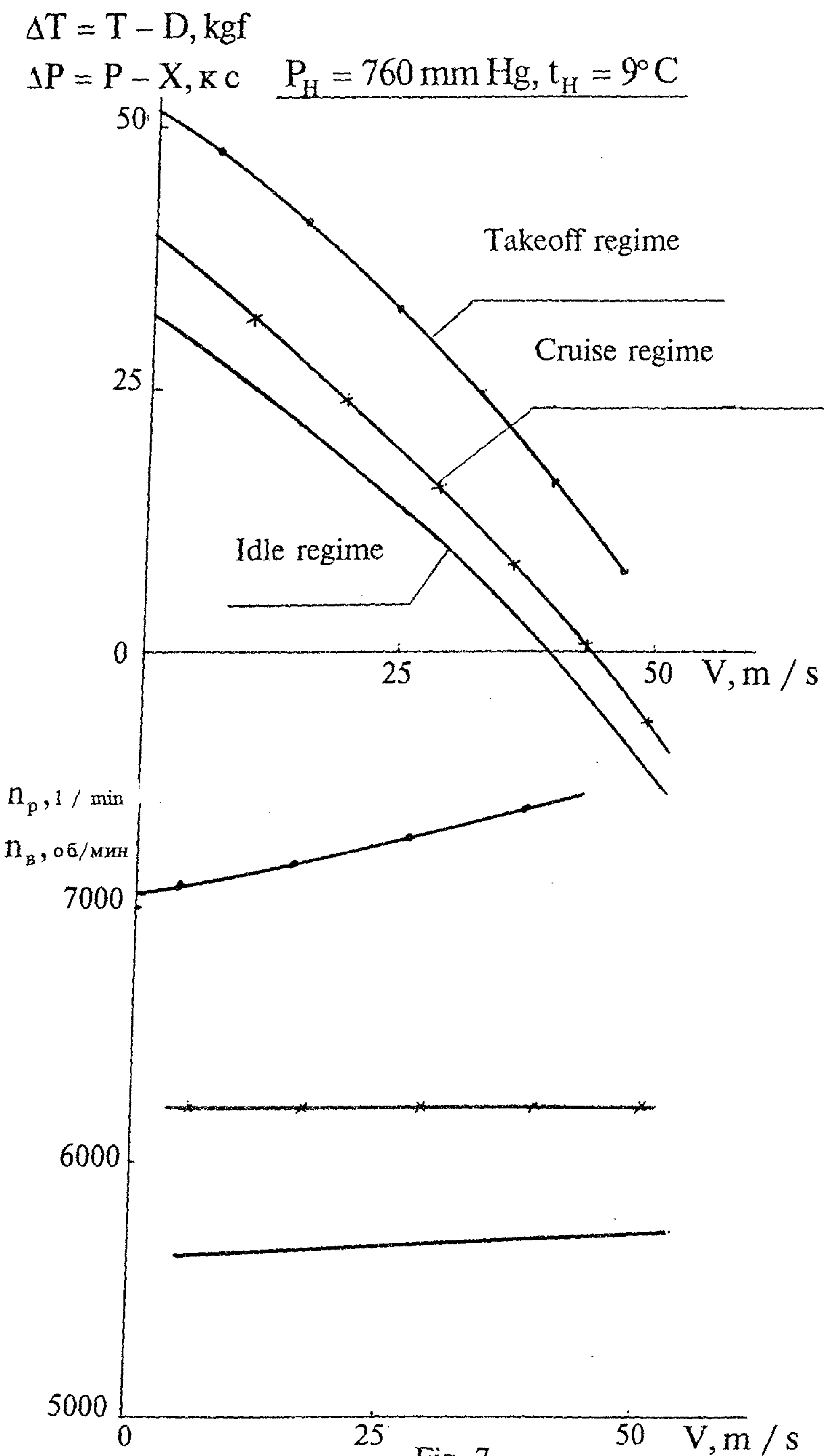


Fig. 6



$\Delta T = T - D$, kgf
$\Delta P = P - X$, кс
$P_H = 760$ mm Hg, $t_H = 9°C$
50
25
0
Takeoff regime
Cruise regime
Idle regime
25
50
V, m / s
n_p, 1 / min
$n_в$, об/мин
7000
6000
5000
0
25
50
V, m / s

Fig. 7

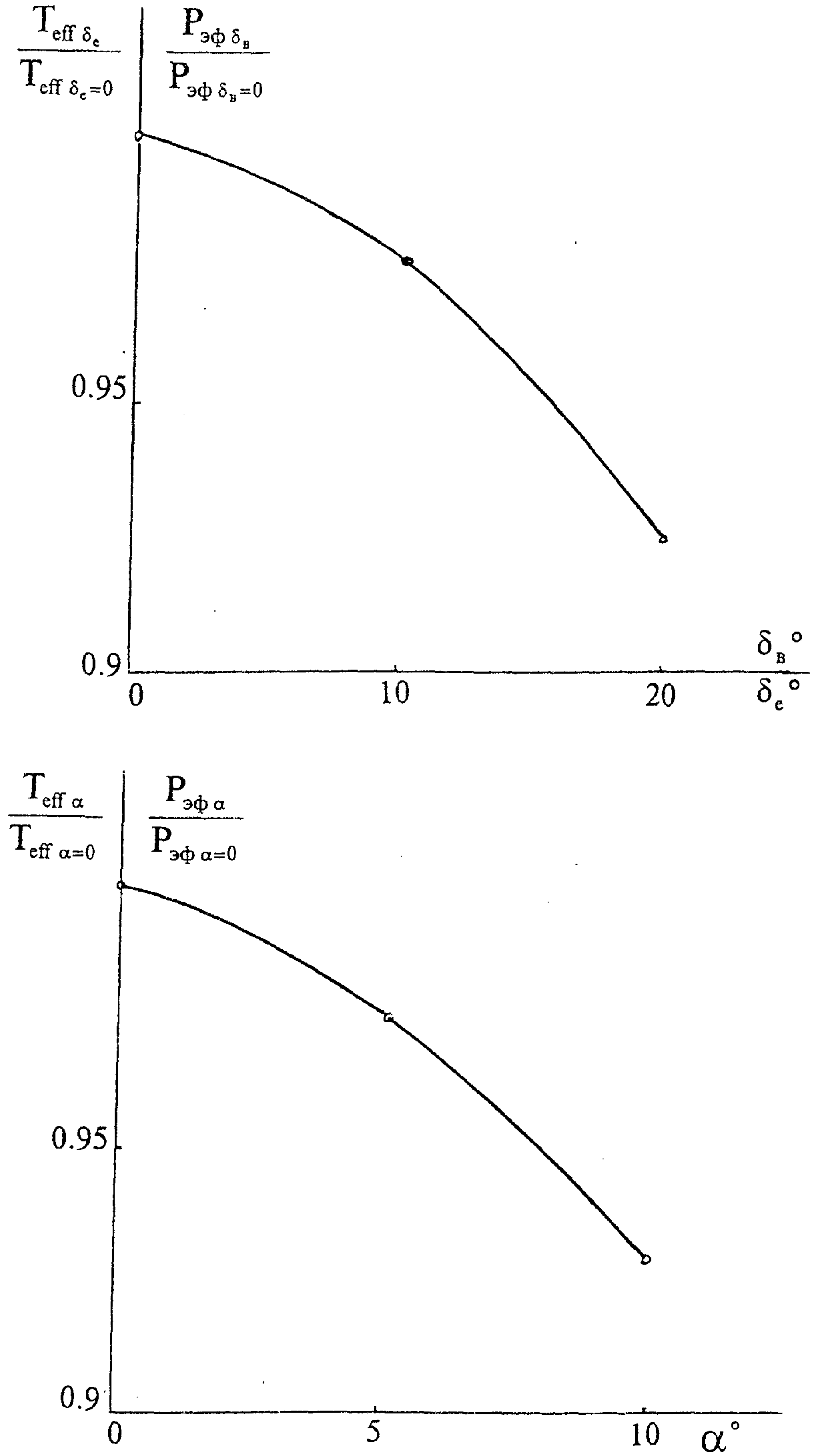

$\frac{T_{eff\ \delta_e}}{T_{eff\ \delta_e=0}}$
$\frac{P_{эф\ \delta_в}}{P_{эф\ \delta_в=0}}$
0.95
0.9
0
10
20
$\delta_в{}^\circ$
$\delta_e{}^\circ$
$\frac{T_{eff\ \alpha}}{T_{eff\ \alpha=0}}$
$\frac{P_{эф\ \alpha}}{P_{эф\ \alpha=0}}$
0.95
0.9
0
5
10
α°

Fig. 8

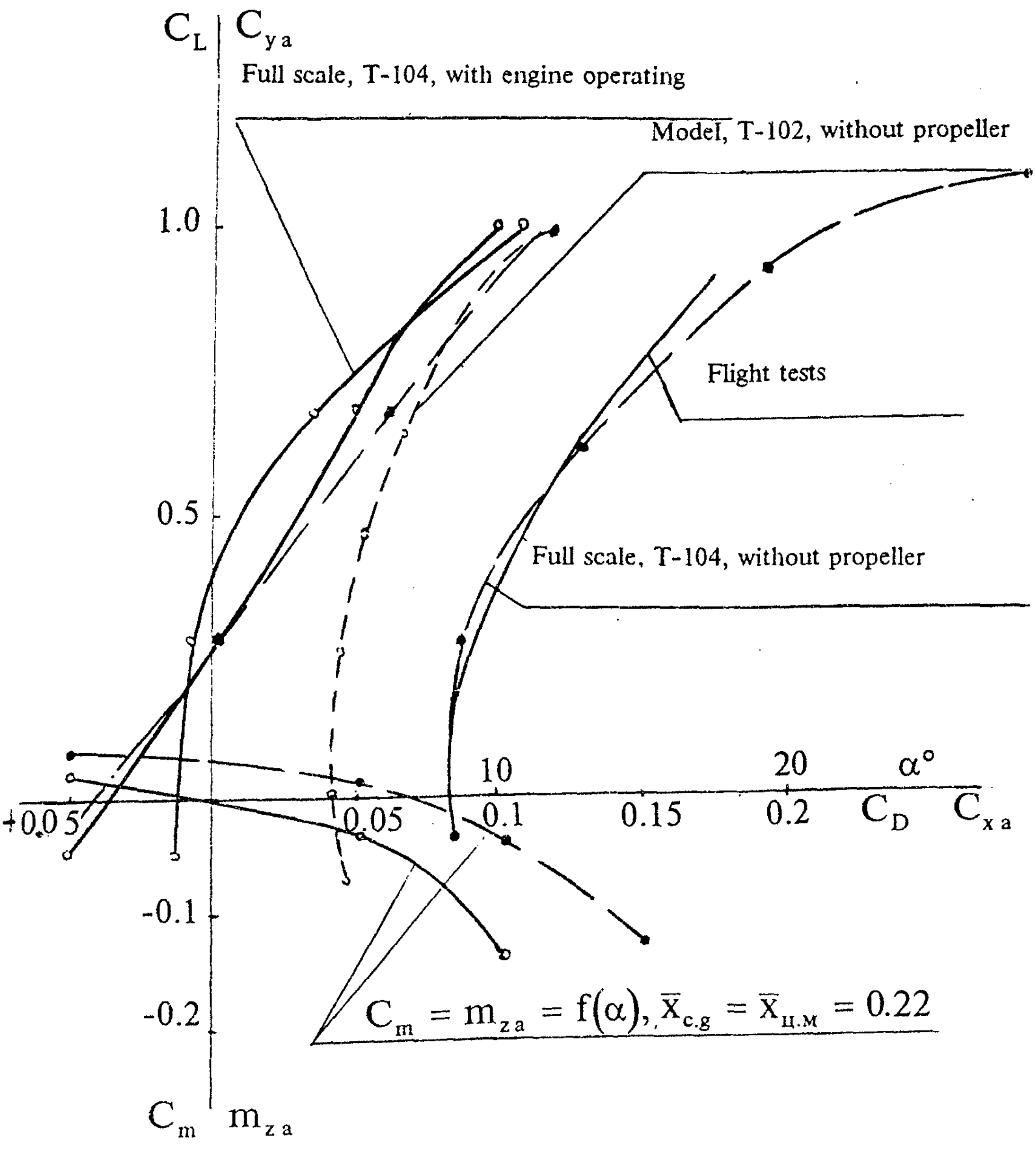
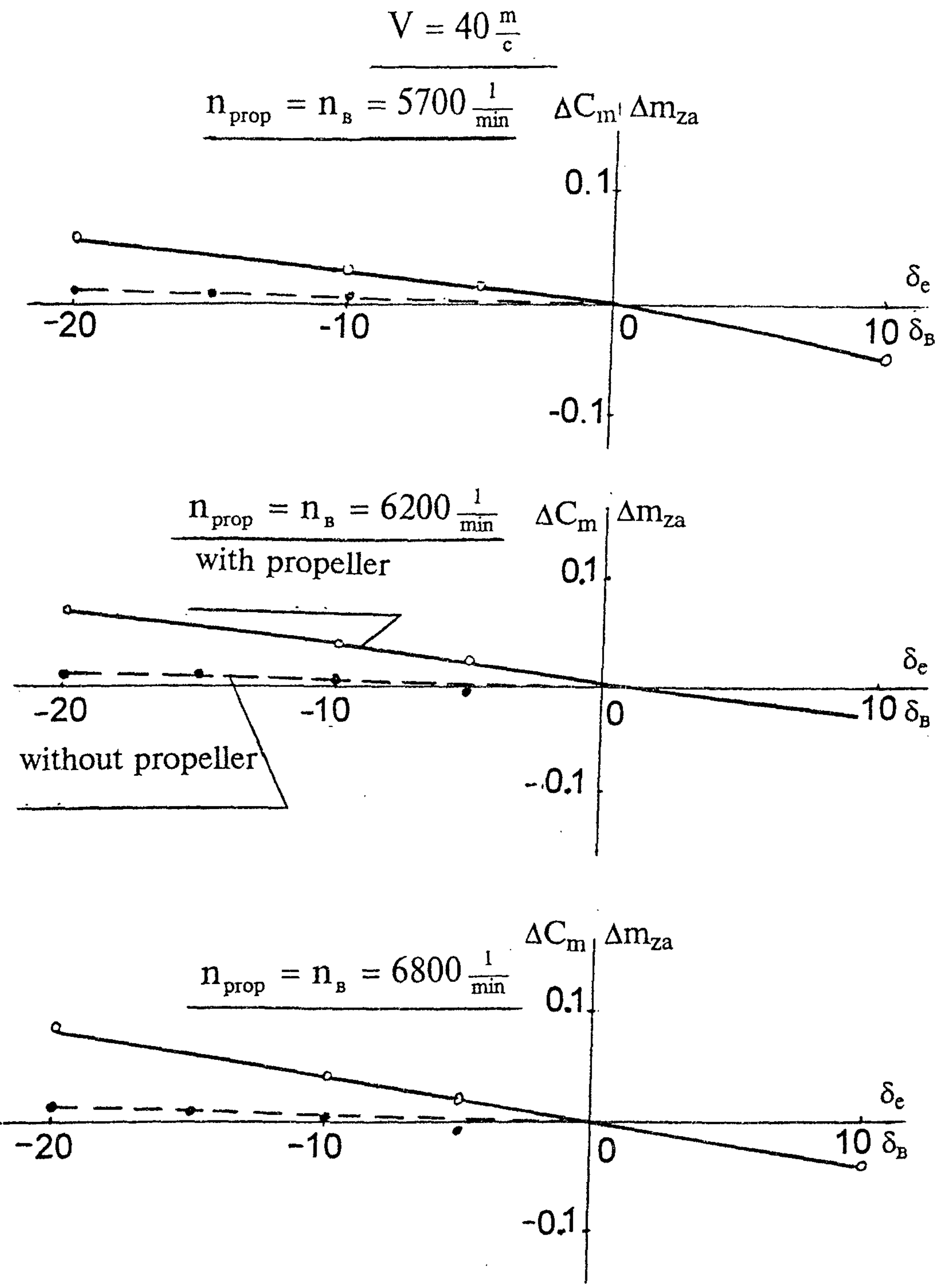

C_L | C_{ya}
Full scale, T-104, with engine operating
Model, T-102, without propeller
1.0
Flight tests
0.5
Full scale, T-104, without propeller
10
20
$\alpha°$
+0.05
0.05
0.1
0.15
0.2
C_D
C_{xa}
-0.1
-0.2
$C_m = m_{za} = f(\alpha), \overline{X}_{c.g} = \overline{X}_{ц.м} = 0.22$
C_m m_{za}

Fig. 9



V = 40 m/c
n_prop = n_в = 5700 1/min
ΔCm ; Δmza
0.1
-0.1
-20
-10
0
10
δe
δв
n_prop = n_в = 6200 1/min
with propeller
without propeller
ΔCm ; Δmza
0.1
-0.1
-20
-10
0
10
δe
δв
n_prop = n_в = 6800 1/min
ΔCm ; Δmza
0.1
-0.1
-20
-10
0
10
δe
δв

Fig. 10

SIMULATION OF FULL-SCALE FLIGHT CONDITIONS IN TESTING MODELS OF RAMJET AND SCRAMJET INLETS IN WIND TUNNELS AT HYPERSONIC SPEEDS

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There are considered plane inlets of Ramjet-Scramjet composite power plants of hypersonic aircrafts for a wide range of flight Mach numbers, respectively up to $M_\infty = 6—7$ and $M_\infty = 12—15$.

It should be noted three fundamental problems being essential for development of ways of hypersonic flow efficient deceleration in jet engine inlets:

1. Peculiarities of flow over inlets at flow Mach numbers more than design Mach number. Inlet design Mach number is usually taken as $M_d = 5—6$.
2. Determination of minimal allowable inlet throat area in decelerating supersonic and hypersonic flows.
3. Recalculation of inlet models wind tunnels tests results and calculations of flow over inlet models at $k = \text{const}$ for full-scale flight conditions providing for change in thermodynamic air properties at large flight Mach numbers $M_\infty \geq 5$.

1. In fig. 1, as characteristic example, there is shown design flow pattern of Ramjet plane inlet at entrance and in throat at Mach number $M_\infty = 7$, $(M_\infty > M_d)$. Inlet design Mach number is equal to $M_d = 6$. Three-step decelerating wedge angles are equal to 10°, 18°, 28°. Computations were run on the basis of solution of Euler equations for two-dimensional flow at $k = 1{,}4$ by marching procedure without regard for viscosity [1]. In fig. 1 there are shown isolines of Mach numbers $M_i = \text{const}$. It should be paid attention to formation of intense resulting diagonal shock waves arising as a result of intersection of shock waves from wedge steps and cowl contour deflections. At position of incidence of resulting shock waves to surfaces of cowl and wedge there are generated Mach shock waves reflection followed by high-entropy subsonic flows (see in fig. 1 crowding together isolines). At wedge behind contour deflection upstream from resulting shock wave from cowl there arises large zone with supersonic flow acceleration. Providing for viscosity in this region there arises boundary layer

separation. Observed design flow pattern is typical for Mach numbers M_∞ more than design Mach number M_d.

2. Ramjet and Scramjet inlets characteristics and, accordingly, engines thrust-efficiency performances depend substantially on relative inlet throat area $\frac{F_{th}}{F_0}$ (or $\frac{h_{th}}{H_0}$). There are considered relative throat area of inlet starting $\left(\frac{F_{th}}{F_\infty}\right)_{start}$ and minimal throat area of supersonic flow separation at entrance $\left(\frac{F_{th}}{F_\infty}\right)_{min}$. Here $F_\infty = F_0 \cdot f$ is air stream section area captured by inlet from free-stream flow, f is airflow rate.

Values $\left(\frac{F_{th}}{F_\infty}\right)_{start}$ and $\left(\frac{F_{th}}{F_\infty}\right)_{min}$ are determined by flow type realized upstream from considered inlet entrance section immediately before inlet starting and in inlet throat before separation. Experimental studies of different types of inlets showed that instead of relationships:

$$\left(\frac{F_{th}}{F_\infty}\right)_{start};\left(\frac{F_{th}}{F_\infty}\right)_{min} = f(M_\infty, \mathrm{Re}, \overline{T}_W, \text{inlet geometry}) \qquad (1)$$

it is expedient to use generalized relationships in designing inlets and in recalculating their characteristics for full-scale flight conditions:

$$\left(\frac{F_{th}}{F_1}\right)_{start};\left(\frac{F_{th}}{F_1}\right)_{min} = \varphi(M_1, \frac{\delta^*}{h_1}).$$

Also, instead of Mach and Reynolds numbers M_∞, Re there is used flow Mach number M_1 at inlet entrance and relative displacement thickness of boundary layer at entrance $\frac{\delta^*}{h_1}$. In this case it is possible to generalize relationship (1) for different types of inlets at regimes $M_\infty \le M_d$, as shown in fig. 2.

Curves $\left(\frac{F_{th}}{F_1}\right)_{\text{start}} = q\left(\frac{1}{\lambda_1}\right)$ and $\left(\frac{F_{th}}{F_1}\right)_{\min} = q(\lambda_1)$, shown in fig. 2, correspond to starting and separation of flow for inlet such as reverse convergent-divergent nozzle (in this case convergent channel at entrance).

If at inlet entrance before starting there is realized flow with bow wave and separation zone at wedge *A*, then experimental points are somewhat above curve $q\left(\frac{1}{\lambda_1}\right)$, that is $\left(\frac{F_{th}}{F_1}\right)_{\text{start}} \approx 1{,}05 \cdot q\left(\frac{1}{\lambda_1}\right)$ for $\text{Re}_{H_0} = (1-5)\cdot 10^6$ and temperature resistant wall $\overline{T}_W = \frac{T_{W_r}}{T_{0_\infty}} = \overline{T}_{W_r}$.

If at inlet entrance before starting there is realized flow with powerful zone of separation at wedge and diagonal shock *Б*, then it is possible substantial decreasing values $\left(\frac{F_{th}}{F_1}\right)_{\text{start}} < q\left(\frac{1}{\lambda_1}\right)$ (dotted line). For this type of flow it is allowable further converging channel at throat over longitudinal and lateral sections. Relative throat area is approximately estimated by formula:

$$\left(\frac{F_{th}}{F_1}\right)_{\text{start}} \approx q\left(\frac{1}{\lambda_1}\right)\cdot q\left(\frac{1}{\lambda_2}\right) ,$$

where λ_2 is average reduced flow velocity at throat before additional converging channel. Relationships shown in fig. 2 enable to predict values $\left(\frac{F_{th}}{F_1}\right)_{\text{start}}$ and, hence, $\left(\frac{F_{th}}{F_\infty}\right)_{\text{start}}$ for inlet for full-scale flight conditions in accordance with flow type before starting by tests in wind tunnel.

3. At hypersonic speeds of flight ($M_\infty > 5$) air temperature at entrance and throat of inlet with rear shock wave ($M_{th} < 1$) becomes high enough ($T_{th} \geq 1000\ K$). In this case in designing flow parameters at throat it is necessary to take air thermodynamic properties behaviour into account [3].

There were estimated flow averaged parameters at throats of a number of inlets № 1—4 (fig. 3) at supersonic and subsonic speeds ($M_{th} > 1$, $M_{th} < 1$) for perfect gas with k = 1,4 and real equilibrium air ($k \neq$ const). Averaged parameters were determined on the basis of equations of conservation, momentum and energy for gas mass flowing between sections in free-stream flow and in throat of inlet. Equation of state was given by tables [4] using special computer program. Friction forces were not taken into account.

In fig. 4 there are shown results of computation of total pressure recovery ratio ν_{th} and flow Mach numbers M_{th} at throat of inlet № 2,4 for perfect gas (solid line) and real air (dotted line). Inlet № 2 was with sharp leading edges of cowl. Inlet № 4 was with blunt edges with relative blunt radius $\frac{r_b}{H_0} = \bar{r}_b = 0{,}005$. Computation was run for flight trajectory corresponding with velocity head $\frac{\rho_\infty u_\infty^2}{2} = 7500 \frac{\text{kg}}{\text{m} \cdot \text{sec}^2}$.

Relationships $\nu_{th}(M_\infty)$ for perfect gas and real air are close between themselves both at $\bar{r}_b = 0$ and at $\bar{r}_b > 0$. Flow Mach numbers M_{th} at throat are larger in the case of real air especially at $M_\infty > 10$. In fig. 5 there is shown ratio of pressures $\frac{P_{th}}{P_{1,4}}$ (solid lines) and ratio of temperatures $\frac{T_{th}}{T_{1,4}}$ (dotted line) for real and perfect gas at $M_{th} > 1$. Parameters of perfect gas and real equilibrium air are denoted by subscripts "1,4","r","th".

In increasing Mach number M_∞ and final angle of decelerating wedge θ_K ahead of entrance section ratios $\frac{P_{th}}{P_{1,4}}$ and $\frac{T_{th}}{T_{1,4}}$ decrease. At M_∞ = const and θ_K = const we approximately have:

$$\frac{P_{th}}{P_{1,4}} \approx \frac{T_{th}}{T_{1,4}}.$$

At $M_{th} > 1$, as shown by parametric computations of hypersonic inlets over a range of Mach numbers $M_\infty \le 15$, angles $\theta_K \le 25°—28°$, there hold true approximate relationships such as:

$(\nu_{th})_r \approx (\nu_{th})_{1,4}$ accurate up to 2 — 5 %;

for flow speed at throat $(u_{th})_r \approx (u_{th})_{1,4}$ accurate up to 0,5%.

Provided $(\nu_{th})_r \approx (\nu_{th})_{1,4}$ for total pressure ratios we have:

$$\frac{(P_{0th})_r}{(P_{0\infty})_r} = \frac{(P_{0th})_{1,4}}{(P_{0\infty})_{1,4}}; \frac{(P_{0th})_r}{(P_{0th})_{1,4}} = \frac{\left(\dfrac{P_\infty}{P_{0\infty}}\right)_{1,4}}{\left(\dfrac{P_\infty}{P_{0\infty}}\right)_r}$$

or

$$\frac{(P_{0th})_r}{(P_{0th})_{1,4}} \approx \frac{1}{K(M_\infty, T_\infty)}, \quad \text{where } K = \frac{\left(\dfrac{P_\infty}{P_{0\infty}}\right)_r}{\left(\dfrac{P_\infty}{P_{0\infty}}\right)_{1,4}}$$

Function $K(M_\infty, T_\infty)$ is practically independent of free stream pressure P_∞ up to altitude H = 50 km.

At regime with flow subsonic speed at throat downstream of rear shock wave ($M_{th} < 1$), as shown by computations, there is approximately satisfied relationship: $(P_{0th})_r \approx (P_{0th})_{1,4}$ accurate up to 3 — 5 %. With provision for this formula for total pressure recovery ratio we have:

$$\frac{(\nu_{th})_r}{(\nu_{th})_{1,4}} = \frac{(P_{0th})_r \cdot (P_{0\infty})_{1,4}}{(P_{0\infty})_r \cdot (P_{0th})_{1,4}} = \frac{\left(\dfrac{P_\infty}{P_{0\infty}}\right)_r}{\left(\dfrac{P_\infty}{P_{0\infty}}\right)_{1,4}} = K(M_\infty, T_\infty)$$

In fig. 5 there is shown relationship $\dfrac{(\nu_{th})_r}{(\nu_{th})_{1,4}} = K(M_\infty, T_\infty)$ at $M_{th} < 1$ for flight trajectory corresponding with velocity head 7500 and 3750 $\frac{\text{kg}}{\text{m}\cdot\text{sec}^2}$ (inlet № 2,3). Estimated points are integral with estimated curve practically. Ratio $\frac{(\nu_{th})_r}{(\nu_{th})_{1,4}}$ is independent of inlet geometry. Derived formula $(P_{0th})_r \approx (P_{0th})_{1,4}$ holds true for normal shock wave in free stream also.

To characterize efficiency of air compression process in inlet there are used both total pressure recovery ratio $\nu = \frac{P_0^I}{P_0}$ and kinetic energy recovery ratio η, which is concerned with gas entropy S in a unique fashion. For perfect gas (k = const) coefficient η is related to coefficient ν by formula:

$$\nu = \frac{P(\lambda_\infty)}{P(\lambda_\infty\sqrt{\eta})}, \quad \text{where } P(\lambda) = \left(1 - \frac{k-1}{k+1}\lambda^2\right)^{\frac{k}{k-1}}.$$

Coefficient η for jet engine inlet ($M_{th} < 1$) slightly varies over a wide range of Mach numbers M_∞ unlike coefficient ν and slightly depends on air thermodynamic properties behaviour at high Mach numbers $M_\infty \leq 10$ and $M_{th} < 1$. So it is convenient to use coefficient η in recalculating results of inlet model tests in wind tunnel for full-scale flight conditions. When $(P_{0th})_r \approx (P_{0th})_{1,4}$ we have $\eta_r \approx \eta_{1,4}$. For regime $M_{th} > 1$ and $\frac{(P_{oth})_r}{(P_{0th})_{1,4}} = \frac{1}{K}$ we obtain $\eta_r = \eta_{1,4}$.

The work was performed thanks to financial support of RFFI. Code: 95—01—00444a.

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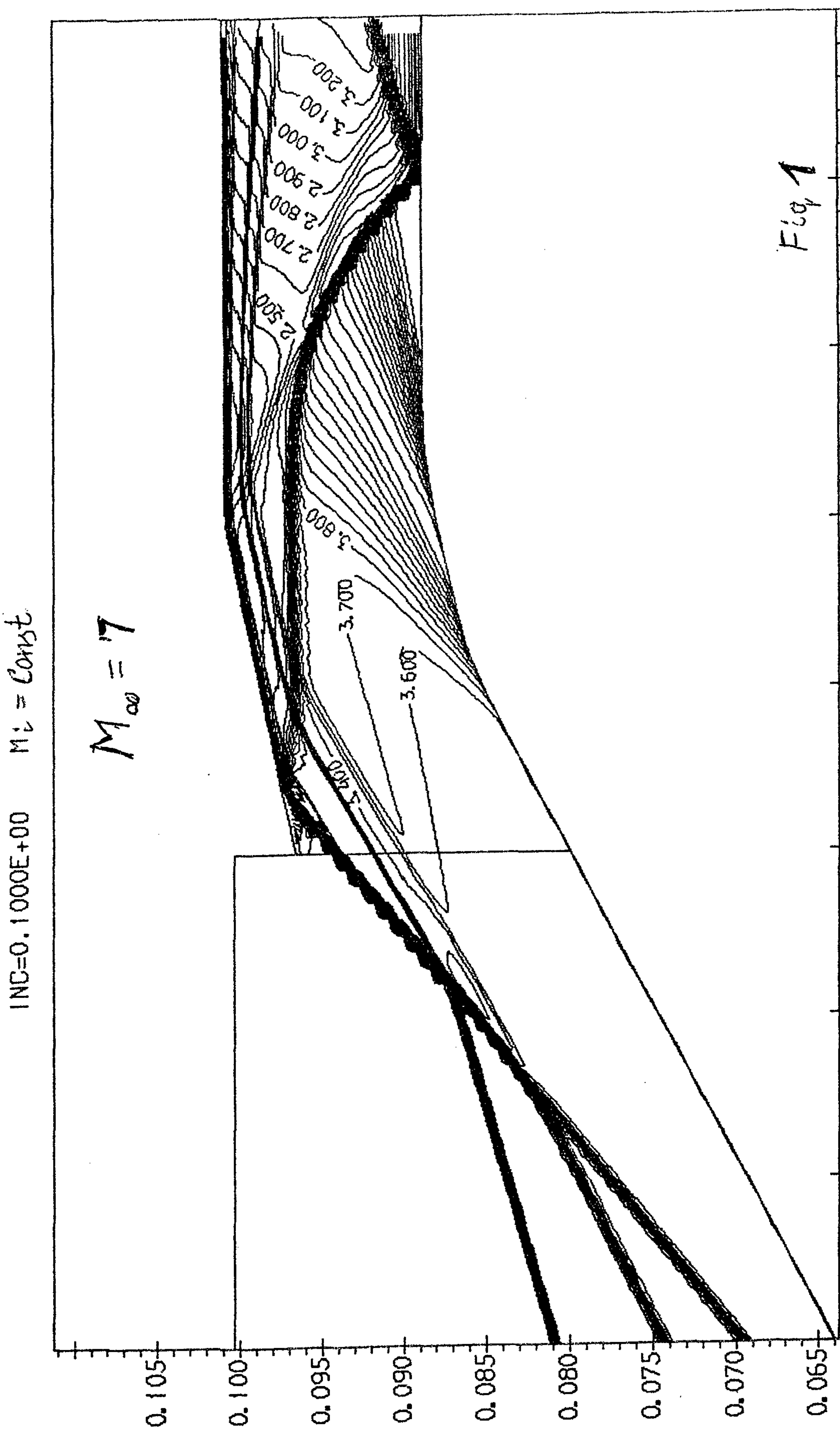
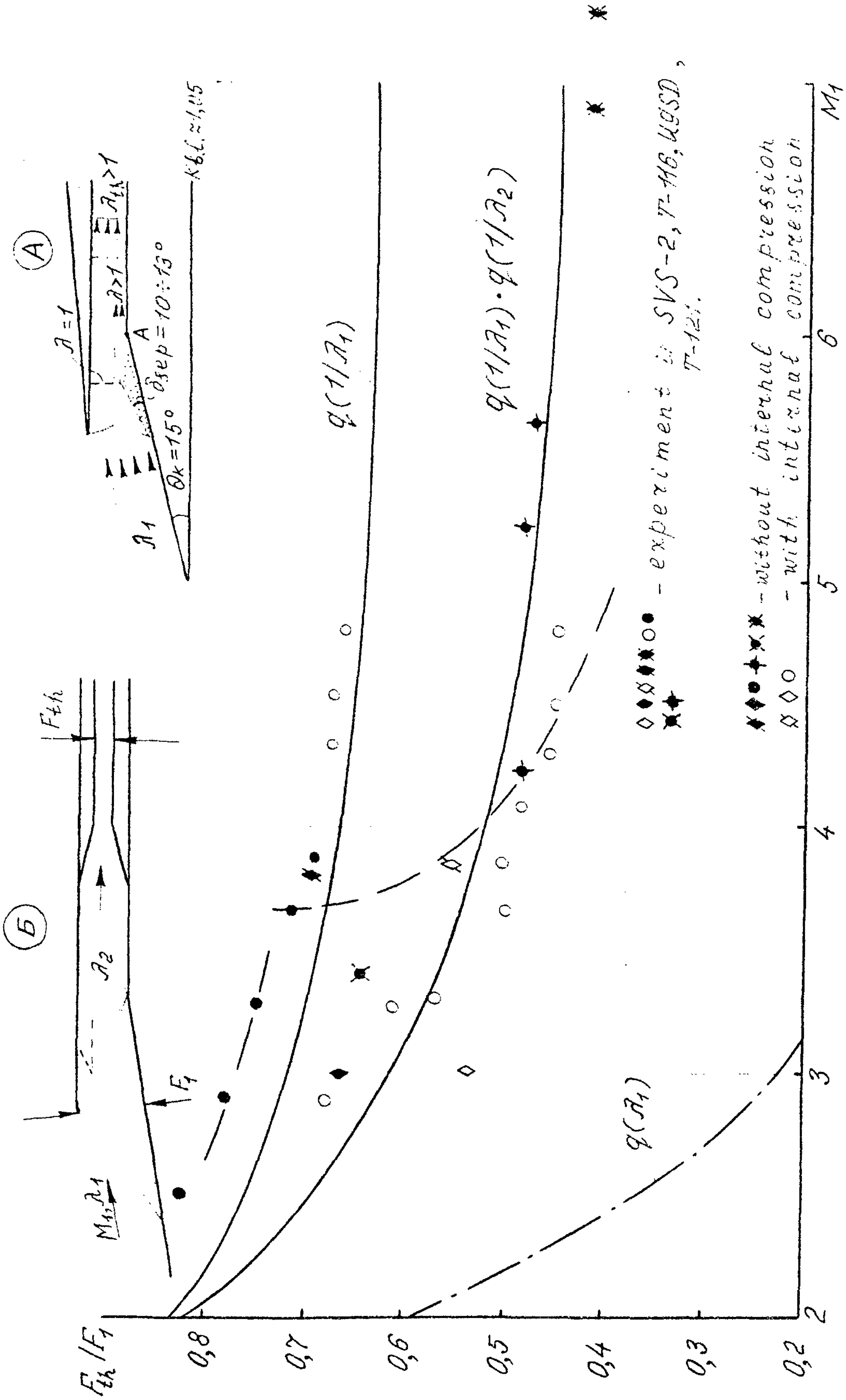
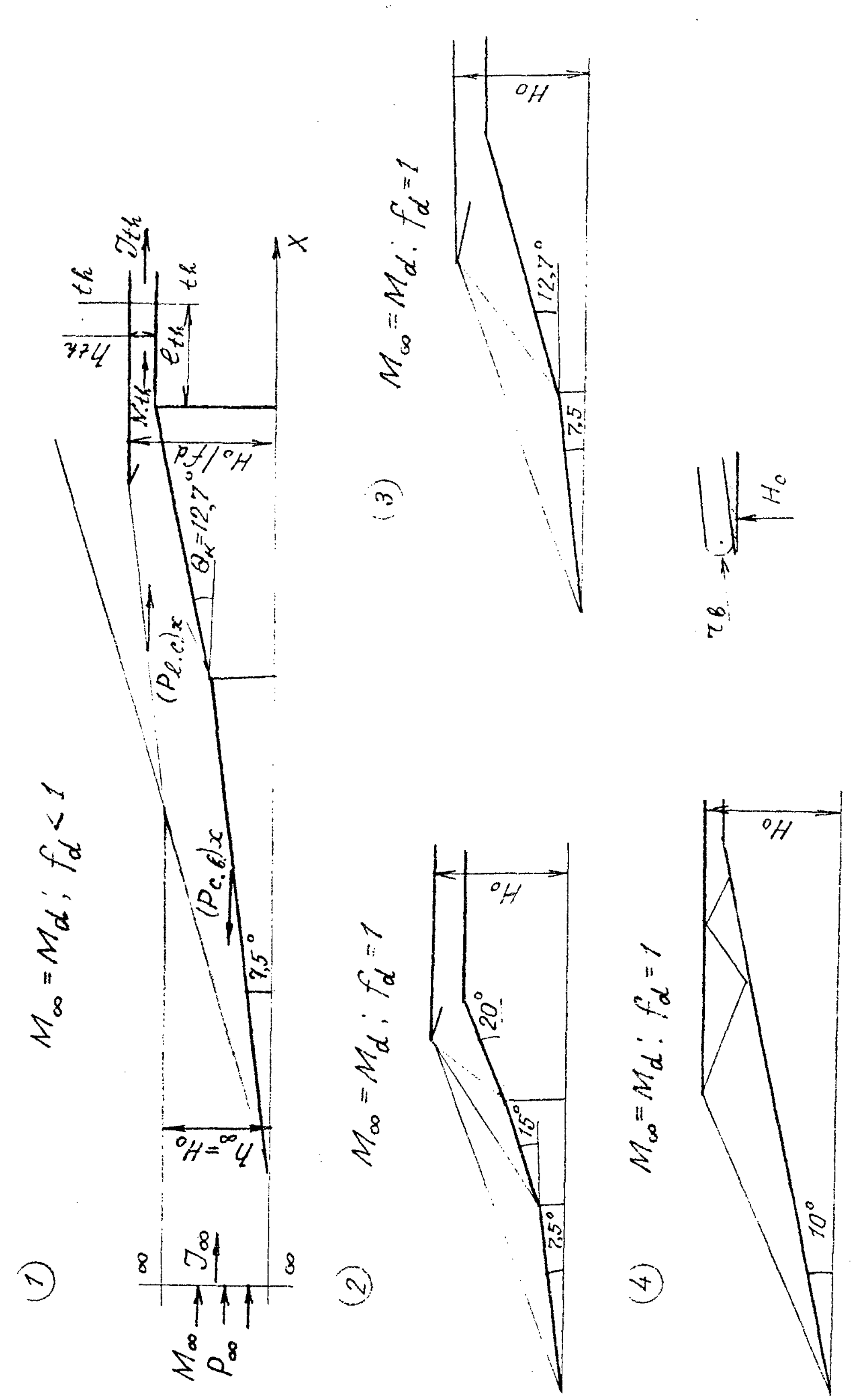

INC=0.1000E+00
Mi = Const
M∞ = 7
3.200
3.100
3.000
2.900
2.800
2.700
2.500
3.800
3.700
3.600
3.400
0.280
0.290
0.300
0.310
0.320
0.330
0.340
0.350
0.065
0.070
0.075
0.080
0.085
0.090
0.095
0.100
0.105
Fig 1



Б
A
λ=1
θk=15°
δsep=10÷13°
Fth
F1
M1, λ1
λ2
q(1/λ1)
q(1/λ1)·q(1/λ2)
q(λ1)
Fth/F1
M1
- experiment in SVS-2, T-116,
T-121
- without internal compression
- with internal compression

Fig. 2



1
$M_\infty = M_d$; $f_d < 1$
M_∞
P_∞
J_∞
$h_\infty = H_0$
7,5°
$(P_{c.b})_x$
$(P_{l.c})_x$
$\theta_k = 12,7°$
H_0/f_d
h_{th}
e_{th}
J_{th}
th
X
2
$M_\infty = M_d$; $f_d = 1$
7,5°
15°
20°
H_0
3
$M_\infty = M_d$; $f_d = 1$
7,5
12,7°
H_0
4
$M_\infty = M_d$; $f_d = 1$
10°
H_0
H_c
r_b

Fig. 3

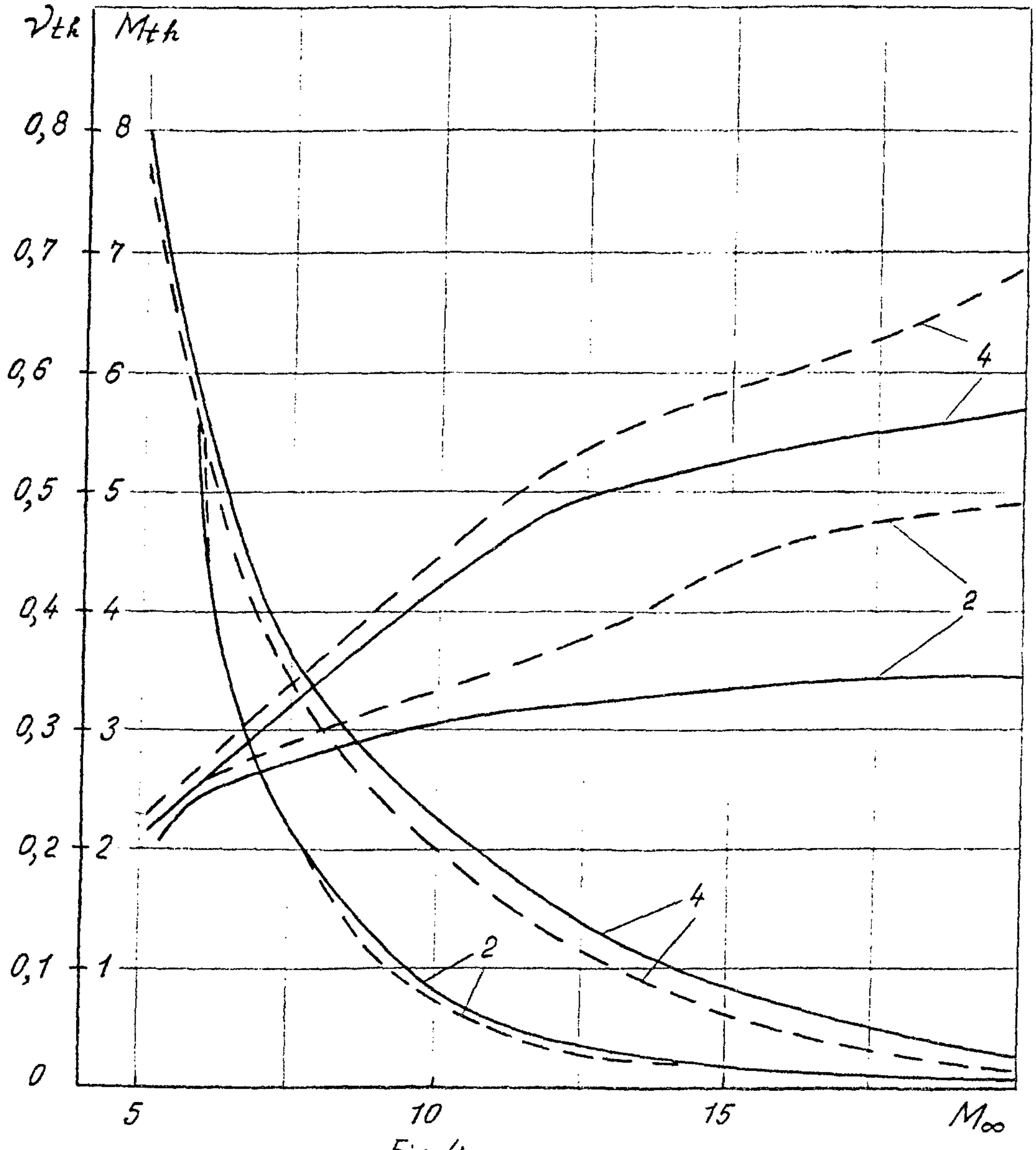


Fig. 4

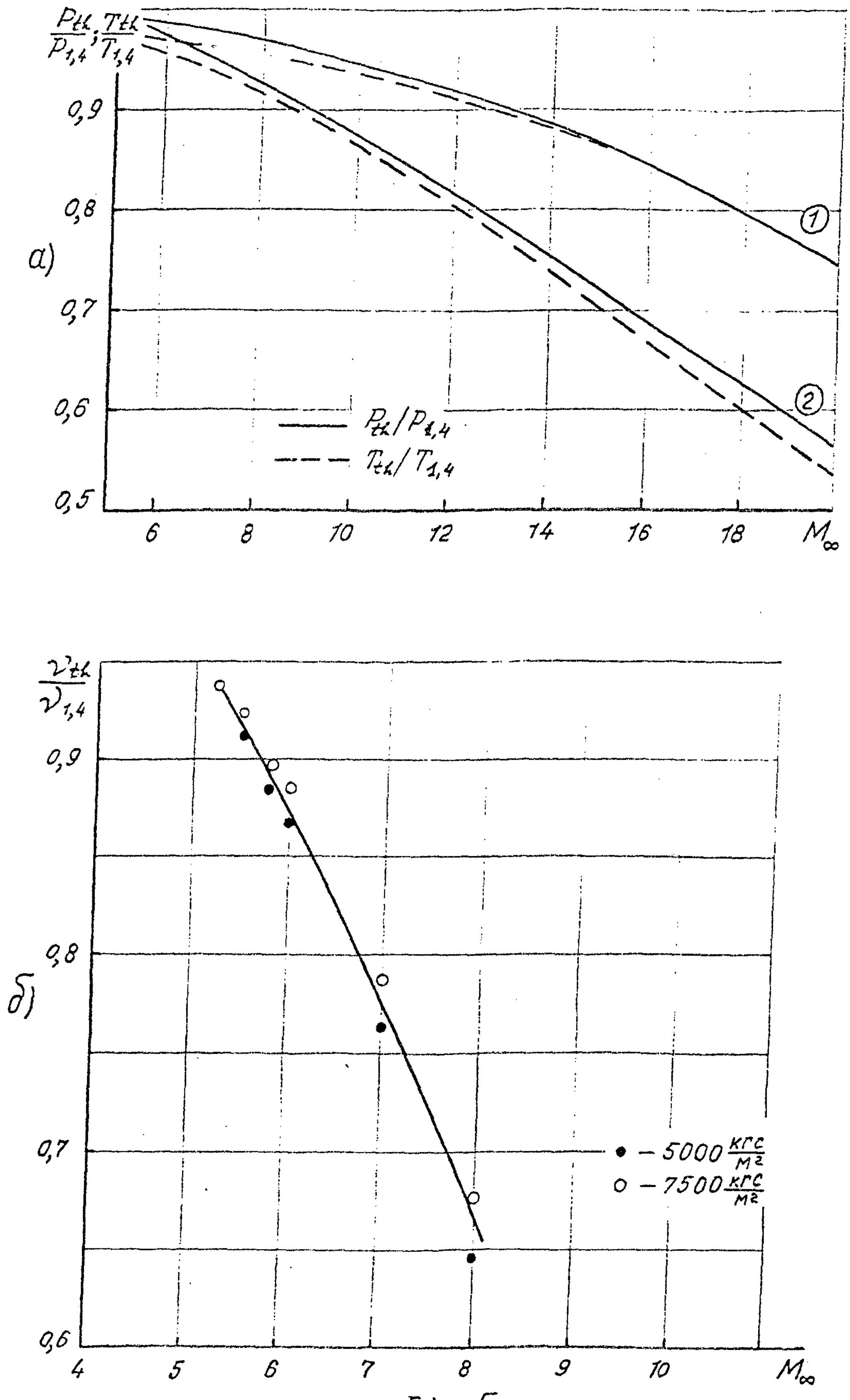

a)
$P_{th}/P_{1,4}$, $T_{th}/T_{1,4}$
0,9
0,8
0,7
0,6
0,5
6
8
10
12
14
16
18
M_∞
1
2
$P_{th}/P_{1,4}$
$T_{th}/T_{1,4}$
б)
$\nu_{th}/\nu_{1,4}$
0,9
0,8
0,7
0,6
4
5
6
7
8
9
10
M_∞
• – 5000 кгс/м²
○ – 7500 кгс/м²

Fig. 5

TEST COMPLEX OF T-101 WIND TUNNEL FOR DETERMINATION OF INTEGRAL AIRPLANE CHARACTERISTICS

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When certifying an airplane in accordance with the Airworthiness regulations, the declared airplane flight characteristics, in particular, L/D-ratio, stability and controllability etc., should be verified. A high accuracy of these characteristics can be attained in wind tunnel model tests, provided the aeromechanical similarity criteria are satisfied.

Among wind tunnel tests investigations of integral aerodynamic airplane characteristics on real airplanes or large-scale models with operating propulsion system models (PSM) are most close to flight conditions in similarity criteria since exactly in these tests the criteria of geometric, kinematic and dynamic similarities can be fulfilled most fully.

The report considers the problems of model investigations.

A great number of the above investigations are accomplished at take-off/landing flight regimes which prove to be most dangerous and complicated in terms of airframe configurations.

The main type of the experiment under consideration suggests the determination of airplane polars for different airframe configurations and control surfaces.

The main features of the experiment are well known:

1. Take-off engine thrust force is several times as large as airplane drag force. Therefore, drag force is defined as a small quantity equal to the difference of two great quantities, namely, engine thrust force and integral axial force acting on the model and measured by a wind tunnel balance.
2. For L/D-ratio to be increased, the propulsion-airframe interference drag should be minimized. This drag is also a small quantity determined by the difference of the same great quantities. Because of this, the integral axial force measured by the balance and the engine thrust force should be determined with a high quality.

Accordingly, the test complex intended to determine the integral aerodynamic airplane characteristics should include:

— a large wind tunnel featuring a high flow quality in the test section and having a high-accuracy balance;

— a large-scal airplane model with operating propulsion system models which satisfies the aeromechanical similarity criteria and which is equipped with adequate measuring means.

The experience shows that the most challenging problem in developing a test model is to make a propulsion system model meeting the above requirements.

The propulsion system model represents an operating propulsion system of an airplane model which is constructed to suit the criteria of the aeromechanical similarity to a real airplane propulsion system in terms of geometric and aerodynamic characteristics and equipped with adequate measurement and data transmission devices.

Based on the similarity theory, it is possible to write a set of equalities to determine the main geometric sizes and jet momentum of the propulsion system model:

$$\begin{aligned} K &= L_n / L_m \\ V_m &= V_n \\ N_m &= K \cdot N_n \\ X_m &= X_n / K^2 \\ R_m &= R_n / K^2 \end{aligned} \tag{1}$$

where K is the geometric scale coefficient, L is the respective geometric size, V is the flight speed (test section flow), N is the propulsor rotational speed, X is the propulsion thrust force, R is the reaction nozzle thrust force. Index m refers to the model and index n to the real propulsion system.

From (1) it follows that the specific power produced in a volume unit of the model engine nacelle is more than that in a real propulsion system by a factor of K. It is impossible to achieve this specific power by means of proportional reduction in geometric sizes of a real propulsion system.

The experience gained shows that a propulsion system model satisfying the similarity criteria can be constructed based on a gas-turbine-powered set with an external working fluid generator (EWFG) in which compression and heat input are provided outside the engine nacelle.

Fig. 1 shows the calculated dependencies which can be used to choose optimal parameters of the PSM working cycle based on the characteristics of available sources of compressed air, heat, safety requirements etc. Having specified reasonable total temperature T_c^* in the PSM throat jet, it is possible to determine, based on the plots presented, the total pressure P^*_{OT}, the total temperature T^*_{OT} at the turbine entry, the available energy h_s, the air mass flow rate G_T and the turbine stage number Z such that the similarity criteria of the PSM jets to the airplane propulsion system jets are satisfied at given values of the ratio of the nozzle trust force R_c to the propulsor thrust force X.

In 1988, the modernization of the T-101 wind tunnel of TsAGI was started in order to install a working fluid generator for testing large-scale models with operating propulsion system models. In 1991, the modernization was finished. The schematic of the external working fluid generator is shown in fig. 2.

The external working fluid generator includes a compressor, a pressure regulator, an air electric heater, a flowmeter and flexible pipe-lines connecting the generator part not to be measured to that to be measured with the model on the balance.

Maximum air mass flow provided by the external working fluid generator is $G = 10$ kg/s, maximum air temperature is $T_o = 600$ K, maximum air pressure is $P_o = 6$ bar.

The influence of the pipe-lines on the balance is accounted for by corrections obtained experimentally.

At present, the test complex of the T-101 wind tunnel is operating with the IL-114 airplane model having two PSM with prop-fans.

The schematic of the PSM-114 is given in fig. 3, and the technical characteristics of the test complex in fig. 4.

The PSM-114 simulator is equipped with the data acquisition system shown schematically in fig. 5. Measured quantities and errors are summarized in table 1.

Fig. 6 presents measured forces and moments acting on the prop-fan in the tests aimed at determining airplane polars at angles of attack α ranging from -10 to +24 degrees for flight velocity (test section flow velocity) of $V = 40$ m/s, rotational prop-fan speed of $N = 1500$ rpm and $N = 2100$ rpm at Reynolds numbers of Re = 1,620,000 and Re = 2,150,000, respectively, when the relative blade radius is $r = 0.75$.

The plots reveal a good coincidence of the curves on the right and left simulators. Forces X and torques M_x are minimum at $\alpha = 0$, which is in agreement with the results of other investigations. Lateral forces Z increase with the angle of attack, they are in the same direction since the prop-fans of both PSM rotate in one direction.

Fig. 7 shows coefficients B of the load on the swept prop-fan area, thrust forces R_c of the reaction nozzles and amplitudes A of alternating stresses in the blade material.

Conclusions

1. The test complex has been constructed in the T-101 wind tunnel to determine integral airplane characteristics at take-off/landing flight regimes using large-scale models with operating propulsion system models.

2. The airplane propulsion system is simulated by an air-turbine engine with a power turbine inside the engine nacelle and with a working fluid generator outside the engine nacelle.

3. The propulsion system model with a prop-fan has been developed for the IL-114 airplane model with two propulsion systems on the wing to meet the criteria of aeromechanical similarity to a real propulsion system. The PSM is equipped with an efficient data acquisition complex comprising a six-component rotary prop-fan dynamometer and telemetric devices.

4. The tests have been carried out to determine polars of the IL-114 airplane using a large-scale model with operating propulsion system models at take-off/landing regimes.

5. The T-101 wind tunnel test results for integral aerodynamic characteristics can be used to accelerate the airplane certification in accordance with the Airworthiness regulations and to increase the safety of flying tests.

Reference

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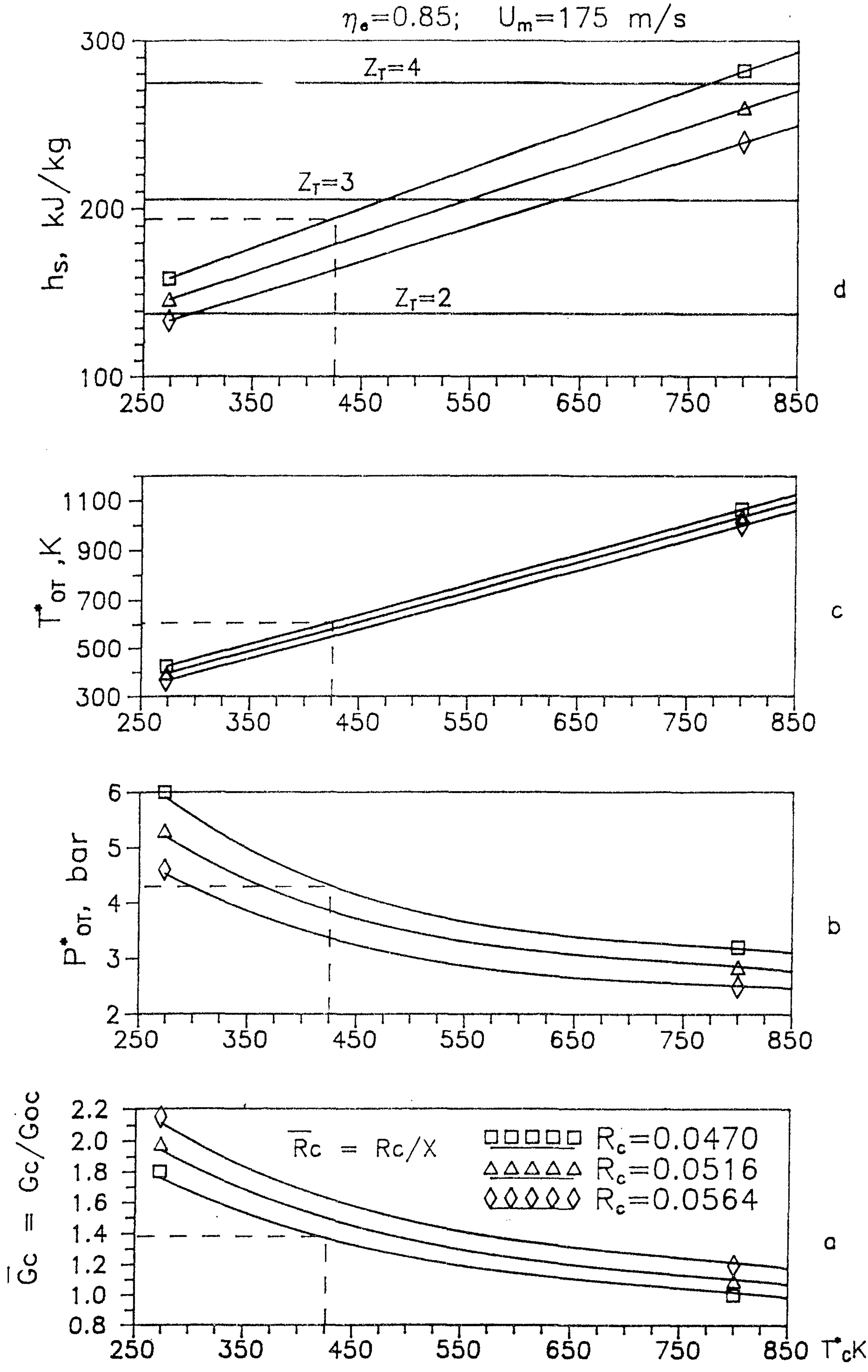


η_e=0.85; U_m=175 m/s
Z_T=4
Z_T=3
Z_T=2
hs, kJ/kg
d
T^*_{OT} ,K
c
P^*_{OT}, bar
b
$\bar{G}c$ = Gc/Goc
$\overline{Rc}$ = Rc/X
R_c=0.0470
R_c=0.0516
R_c=0.0564
a
T^*_cK

Fig. 1.

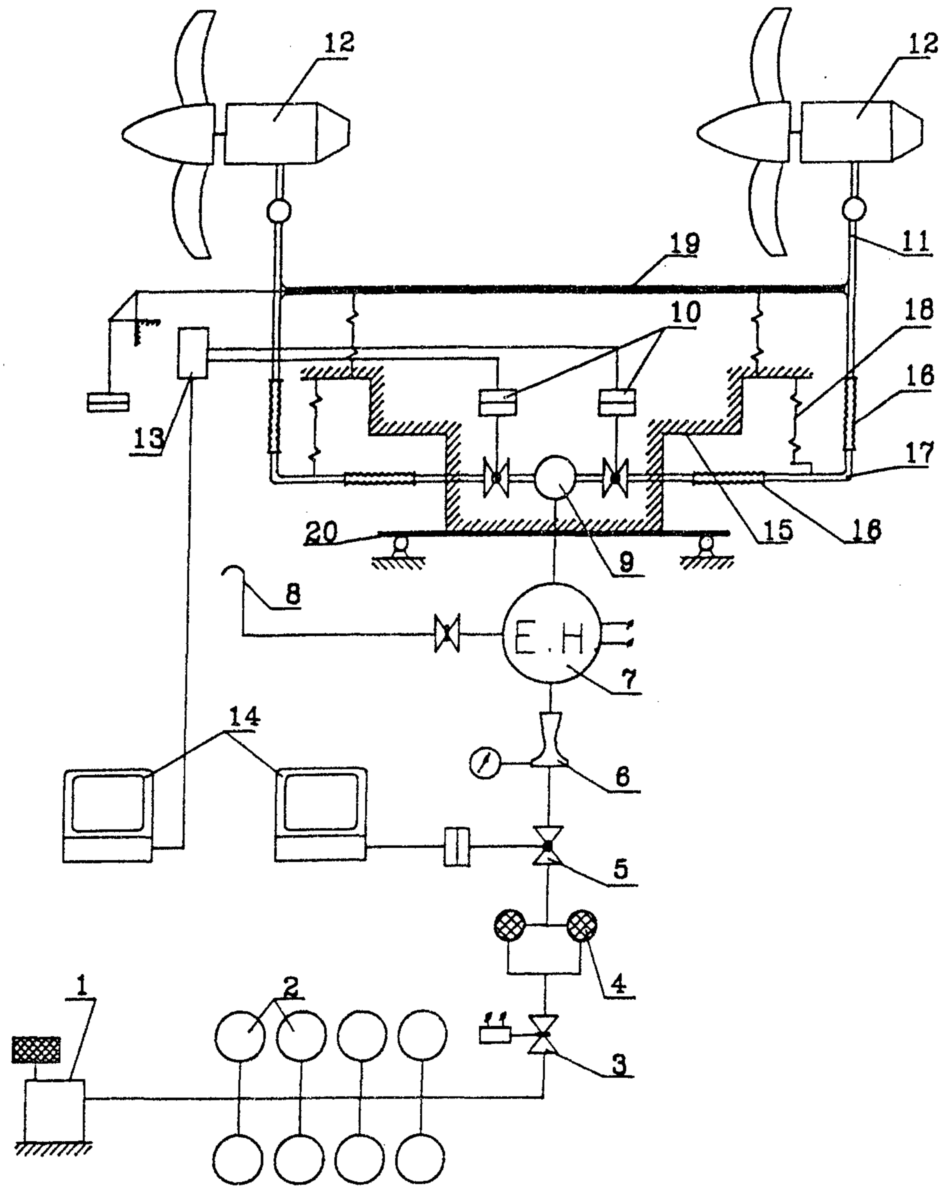


Fig.2. Schematic of hot compressed air suply to PSM of IL-114 airplane model in T-101 wind tunnel

1-compressors; 2-gas holders; 3-gate; 4-filters; 5-pressure regulator; 6-flowmeter; 7-electric heater; 8-by-pass; 9-tee-joint with a sphere; 10-pressure regulators; 11-load-carrying strut pipe-lines; 12-PSM; 13-hydraulic actuator; 14-processor; 15-boundaries of system part not to be measured(ground); 16-flexible pipe-lines; 17-bend; 18-flexible vertical rod; 19-weighted balance platform(6D); 20-rotatable balance support.

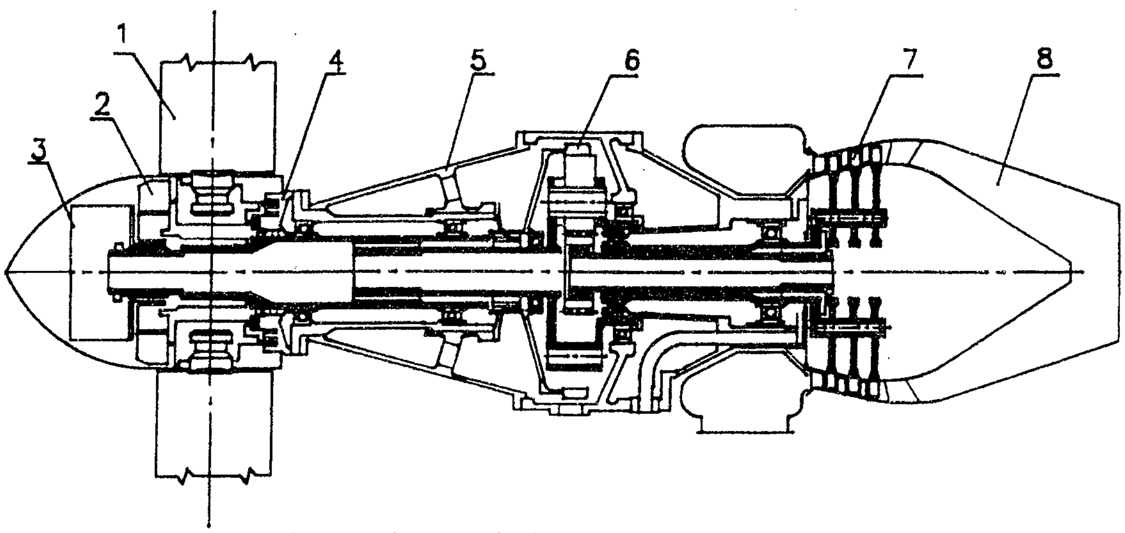


Fig.3. Schematic of MSU-114

1-propfan model, 2-sixs-component (6D) rotating load gauge, 3-telemetry block, 4-fiber-optic devices, 5-casing, 6-planetary reduction gear, 7-three-stage turbine 8-exhaust diffuser.

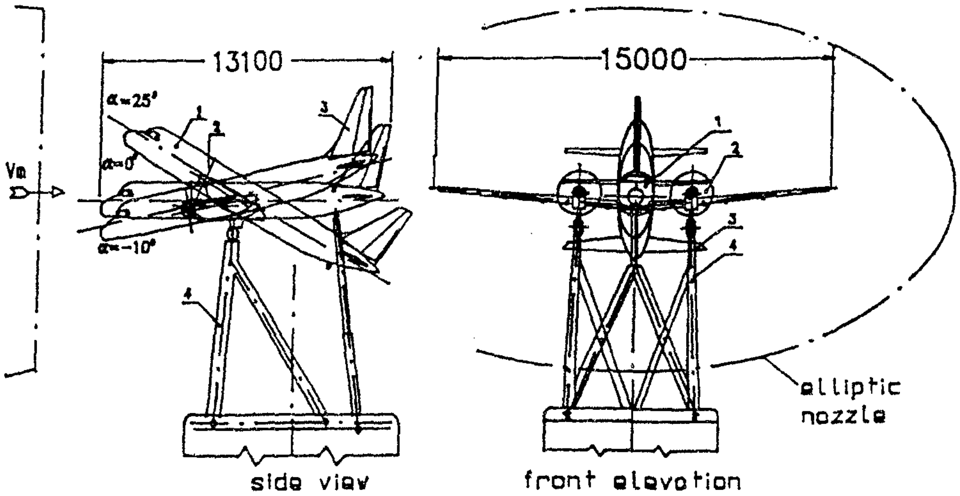


Fig.4. IL-114 airplane model on the balance of T-101 wind tunnel

1-airplane model, 2-propfan, 3-tail unit, 4-balance strut

Technical data of the test complex

size of elliptic nozzle of T-101	24×14 m
flow speed	$\leqslant$ 50 m/s
angle of attack range	$-10°\ldots+25°$
angle of sideslip range	$-20°\ldots+20°$
model wing span	$L_v = 15$ m
simulator propfan diameter	D = 1.8 m
propfan rotational speed	n=0...2400 rpm
propfan thrust	X=-1000...+7750 N
exhaust nozzle thrust	R_c=0...385 N
equivalent power of simulator	$N_{equ} \leqslant 482$ kW

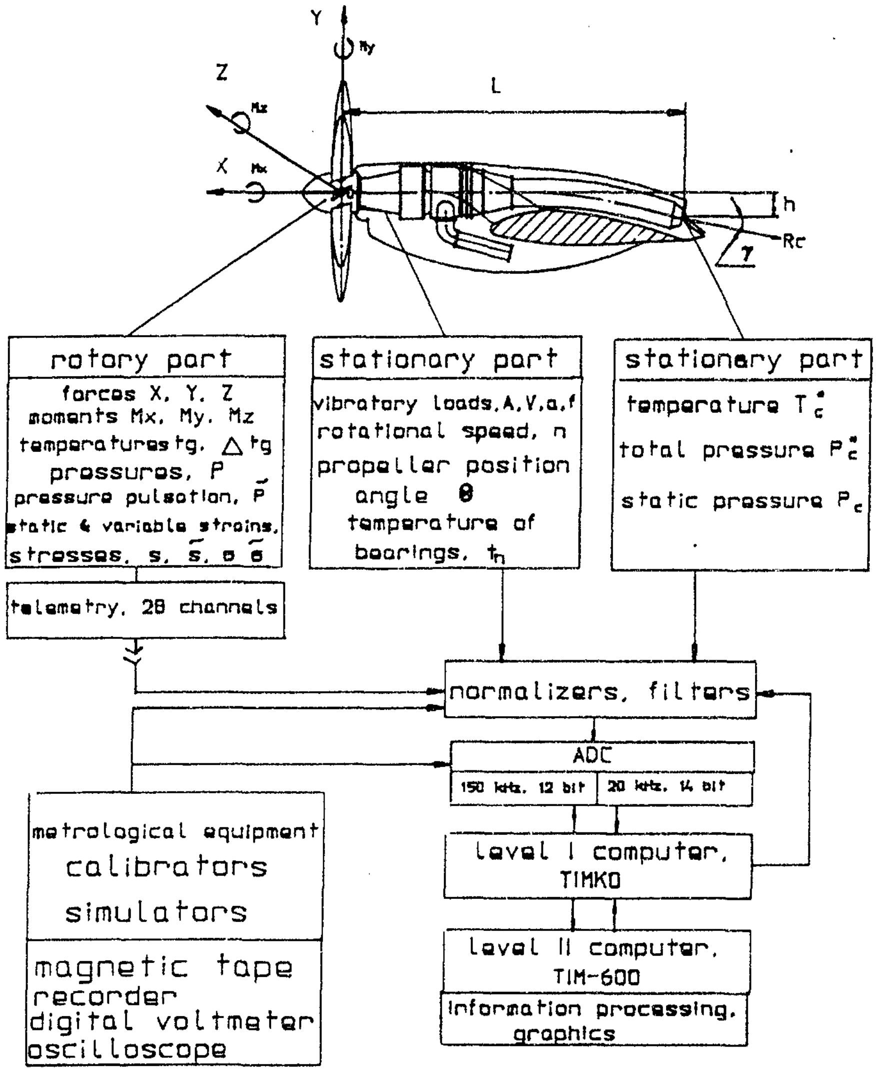


Fig. 5. Measuring system scheme

Table 1

value measd.	number of channels	range capability	frequency range Hz	error (standard deviation), %
X	1	-4000...+8000 N	< 1	< 0.5
Mx	1	-900...+1900 N*m	< 1	< 0.6
Y, Z	1	-2000...+3000 N	≤ 400	< 2
My, Mz	1	-1700...+1700 N*m	≤ 400	< 2
tg	1	0...70° C		< 1
△ tg	1	0...16° C		< 1
P	< 14	0...0.35 bar	< 0.25	< 1
$\tilde{P}$	< 16	0...0.35 bar	≤ 10000	< 5
P_c^*	13	0...0.35 bar	< 0.25	< 0.5
P_c	4	0...0.05 bar	< 0.25	< 2
T_c^*	8	270...550° K		< 1
σ, $\tilde{\sigma}$	< 29	0...$20 \cdot 10^3$ [illegible]	≤ 10000	< 3, < 5
n	1	0...2500 min^{-1}	0...50	< 0.1
TETA	1	0...6.28 rad		< 0.7
t_n	4	0..200° C		< 1.5
A, V, a, f	4	V ≤ 50 mm/s	≤ 1000	< 10

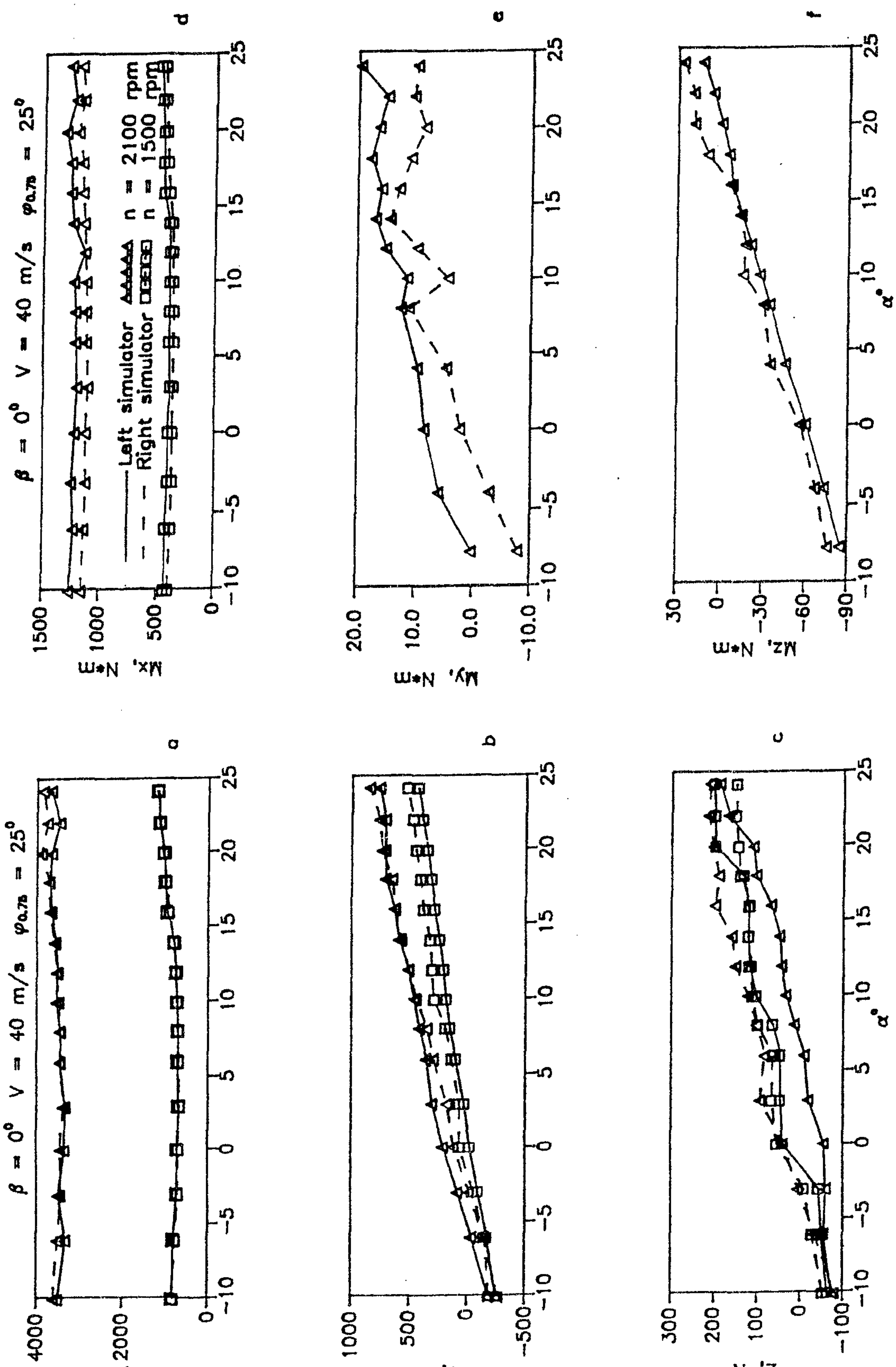
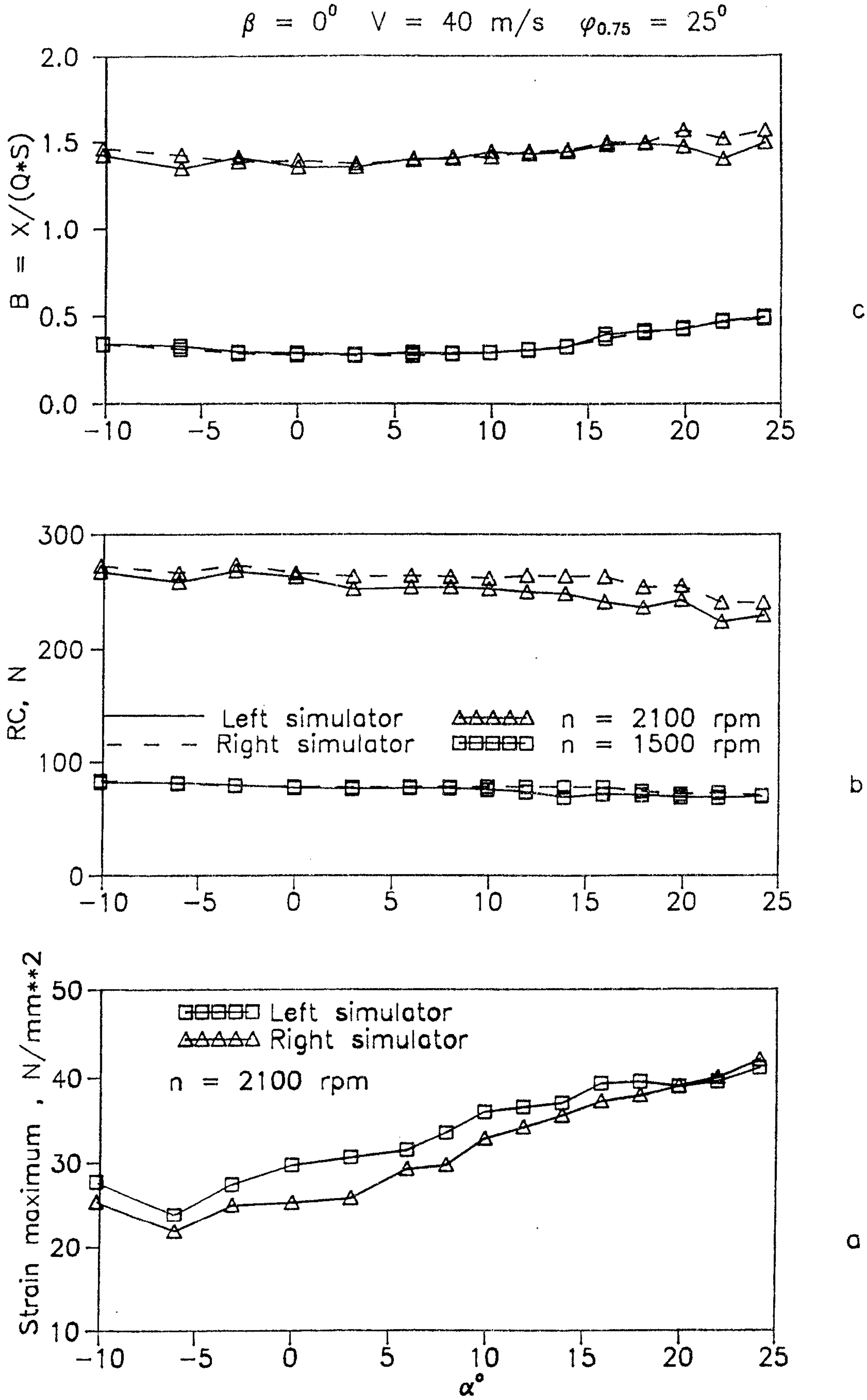
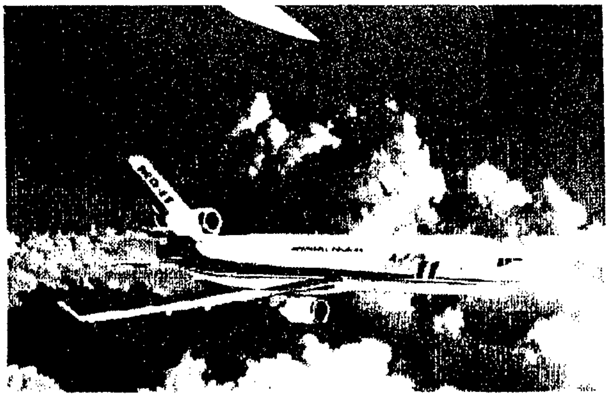


Fig.6. Aerodynamic forces and moments acting on the propfan of the MSU–114.



β = 0° V = 40 m/s φ0.75 = 25°
B = X/(Q*S)
2.0
1.5
1.0
0.5
0.0
−10
−5
0
5
10
15
20
25
c
RC, N
300
200
100
0
Left simulator
Right simulator
n = 2100 rpm
n = 1500 rpm
b
Strain maximum , N/mm**2
50
40
30
20
10
Left simulator
Right simulator
n = 2100 rpm
α°
a

Fig. 7.



MD-11
Drag Reduction Case Study - Pylon Fillet

Presented to:

International Symposium on Experimental Facilities and Aircraft Certification
Central Aerohydrodynamic Institute (TsAGI)
Zhukovski, Russia

August 22 - 24, 1995

Presented by:

W. W. Rickard, General Manager
Aerodynamics and Acoustics

R. D. Gregg, Widebody Program Manager
Aerodynamics and Acoustics

MCDONNELL DOUGLAS

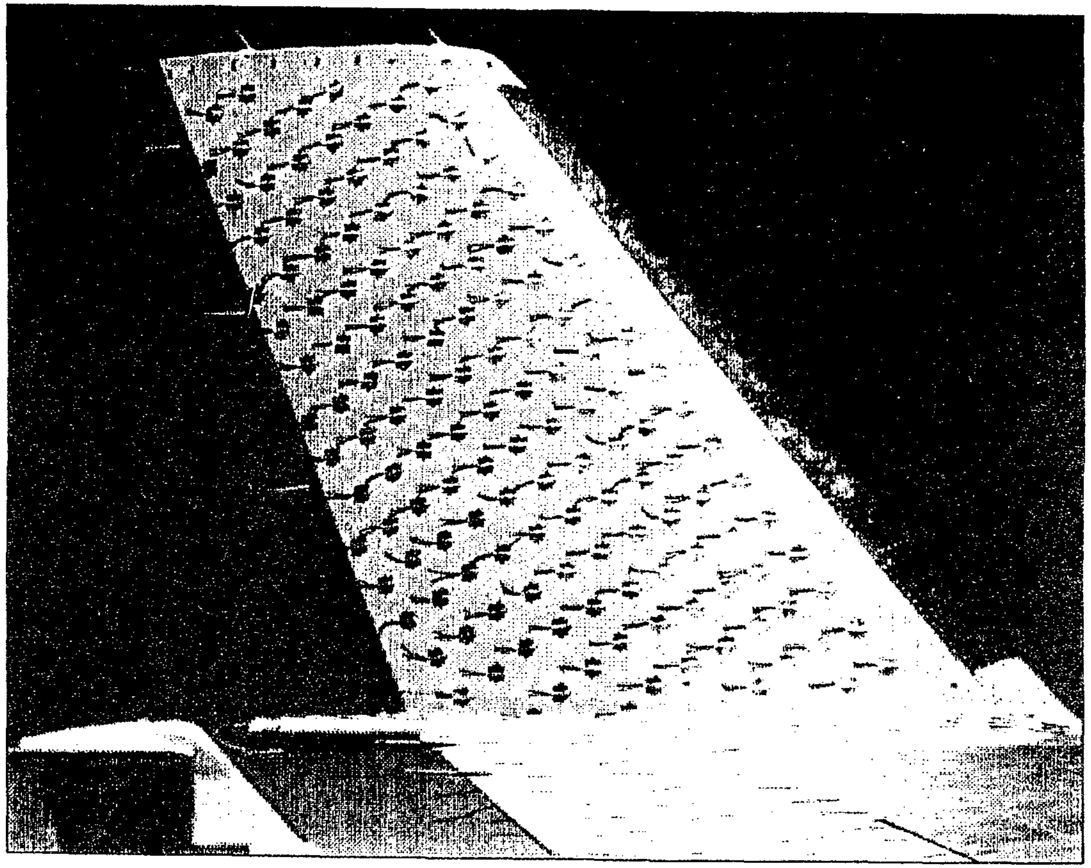
Overview

- MDC Made a Commitment to Significantly Reduce Drag of the MD-11
- Flight Test Flow Visualization was Done to Identify Possible Areas of Improvement
- Small Area of "Lazy" Flow Found on Outboard Side of Wing Engine Pylon - Possibly Separated Flow
- CFD Analysis Was Done to Verify Ability of Code to Duplicate Flight Observation - Showed Some Separation
- Pylon Fillet Designed Using CFD - Virtually Eliminated Flow Disturbance
- Measured Drag Reduction by "Before" and "After" Flight Test
- Actual Drag Reduction of 0.8% Validated by Airlines
- Pylon Fillet Now on Virtually All MD-11s

Flow Visualization

- MDC Has Long, Successful History of Flow Visualization Using Tufts
- For This Application, Used 3 Inch Lengths of 0.125 Inch Parachute Cord Spaced 3 to 6 Inches Apart
- Covered "Hot" Spots (Regions of Suspected Flow Problems) With Tufts
- Viewed and Photographed (Still and Video) Tuft Behavior From Chase Plane at Various Mach/Weight/Altitude Conditions
- Found Area of "Lazy" Tufts (Not Steadily Streaming Aft) on Outboard Side of Wing Pylons

Example of Flow Disturbance Indicated by "Lazy" Tufts

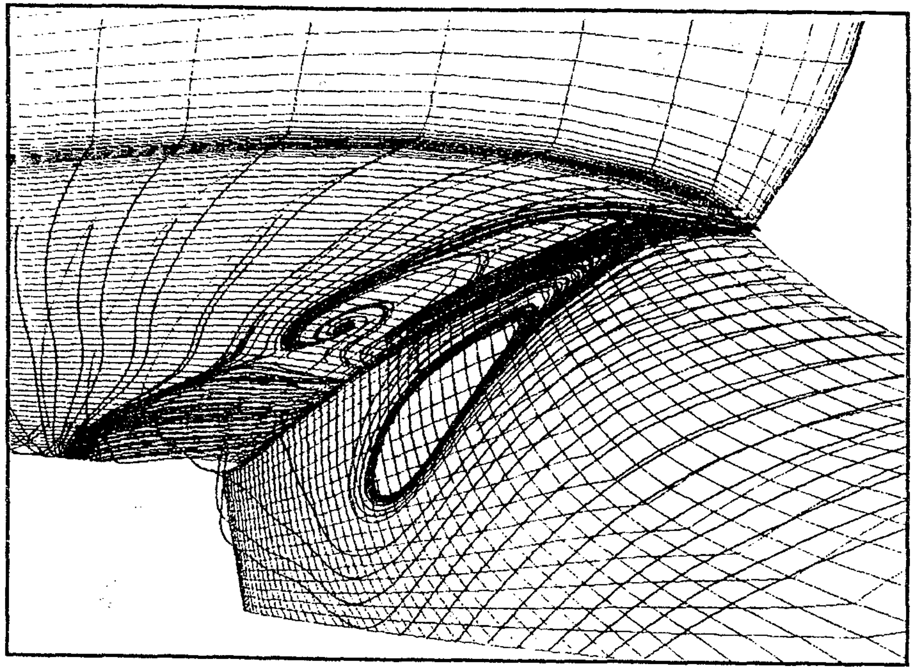


CFD Verification of Problem

- Three Dimensional Model of Wing, Fuselage, Nacelle, and Pylon Was Constructed
- CFD Grid Was Created
- Total of 1.2 Million Nodes Required
- Flow Computed Using NASTD, A Navier-Stokes Code
- Streamline Traces for Near Surface Flow Show Same Flow Flow Disturbance as Flight Test – Validates Ability of Code to Simulate This Flow Disturbance
- Drag Penalty for Flow Disturbance Estimated to be 0.5 - 1%

NASTD

CFD Streamline Tracing Showing Area of Disturbed Flow (No Fillet)

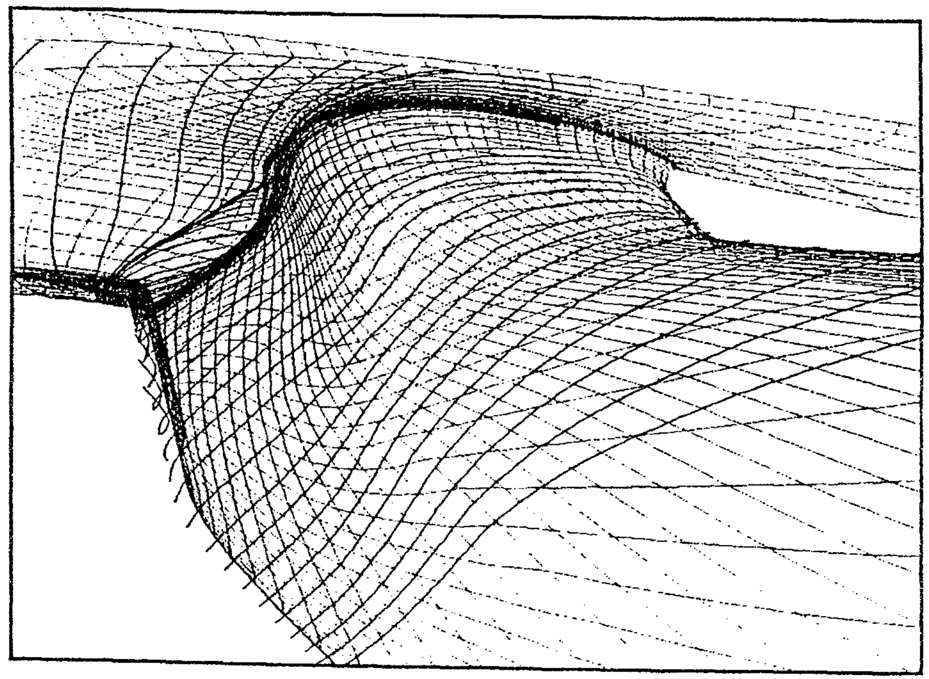


Fillet Design

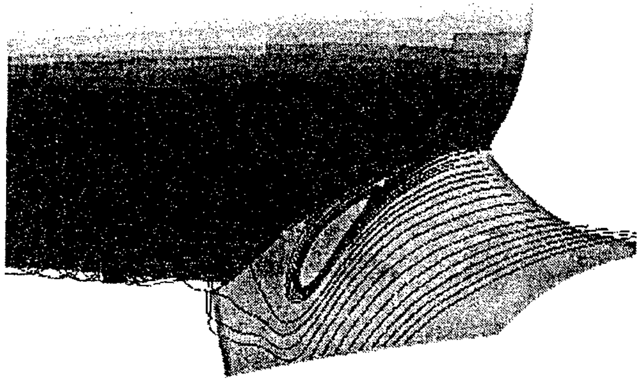
- Compute Flow Using Euler CFD Code AIRPLANE
- Evaluate CFD Solution for Regions of Large Adverse Gradients
- Perturb Geometry at Wing/Pylon Intersection to Reduce Surface Curvature in Region of Adverse Gradients
- Fair Perturbed Section Along Pylon to Create Complete 3-D Definition
- Evaluate New Geometry Using AIRPLANE
- Iterate Process Until AIRPLANE Solution Indicates Adverse Gradients Eliminated
- Assess Candidate Modification Using Navier - Stokes Code NASTD

NASTD

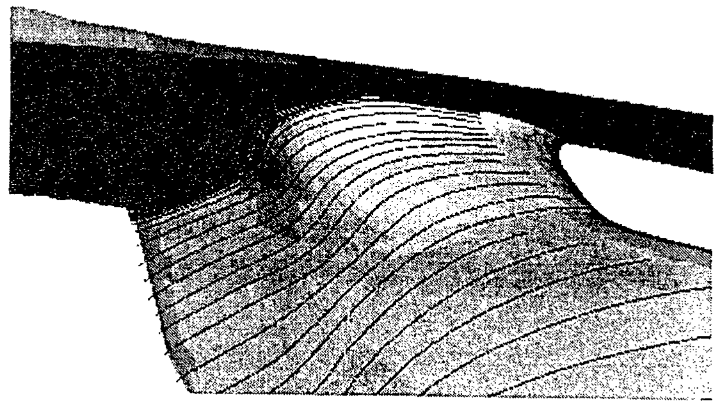
CFD Streamline Tracing Showing Area of Disturbed Flow Eliminated in Presence of Fillet



Comparison of Flow Features Without and With Fillet

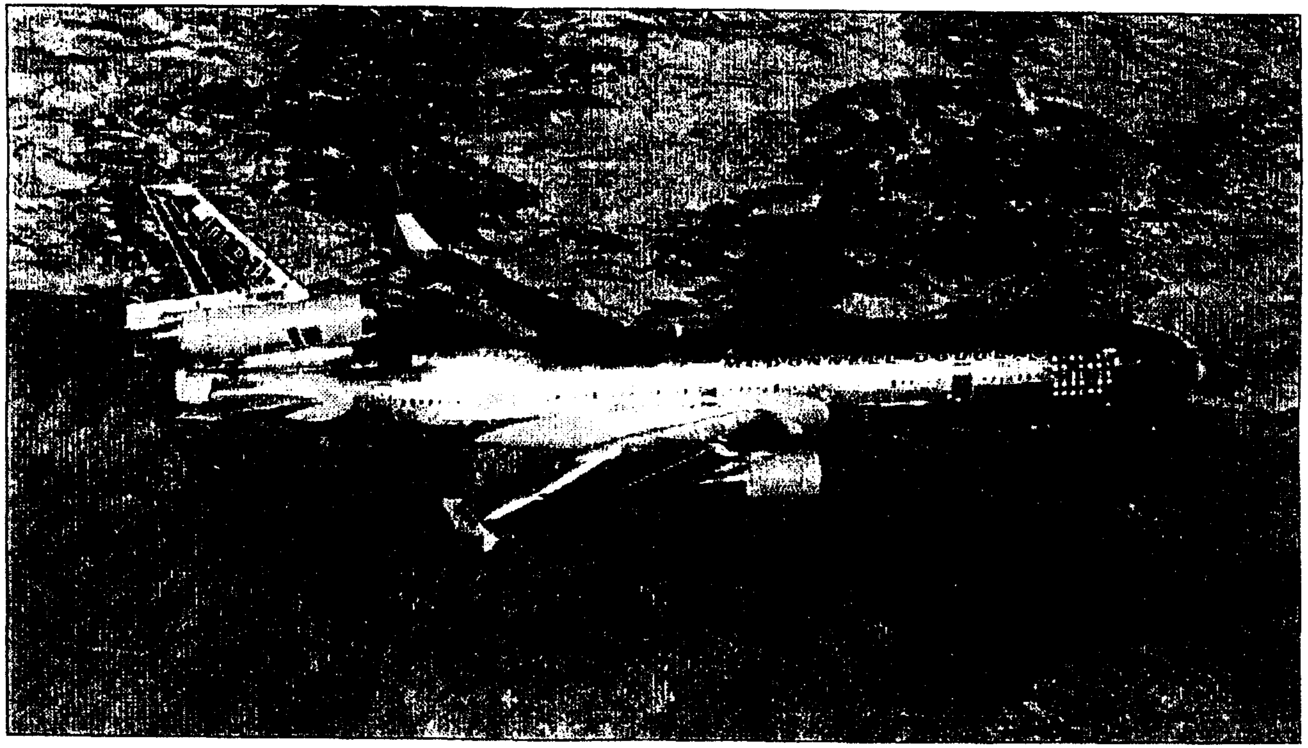


BASELINE – NO FILLET

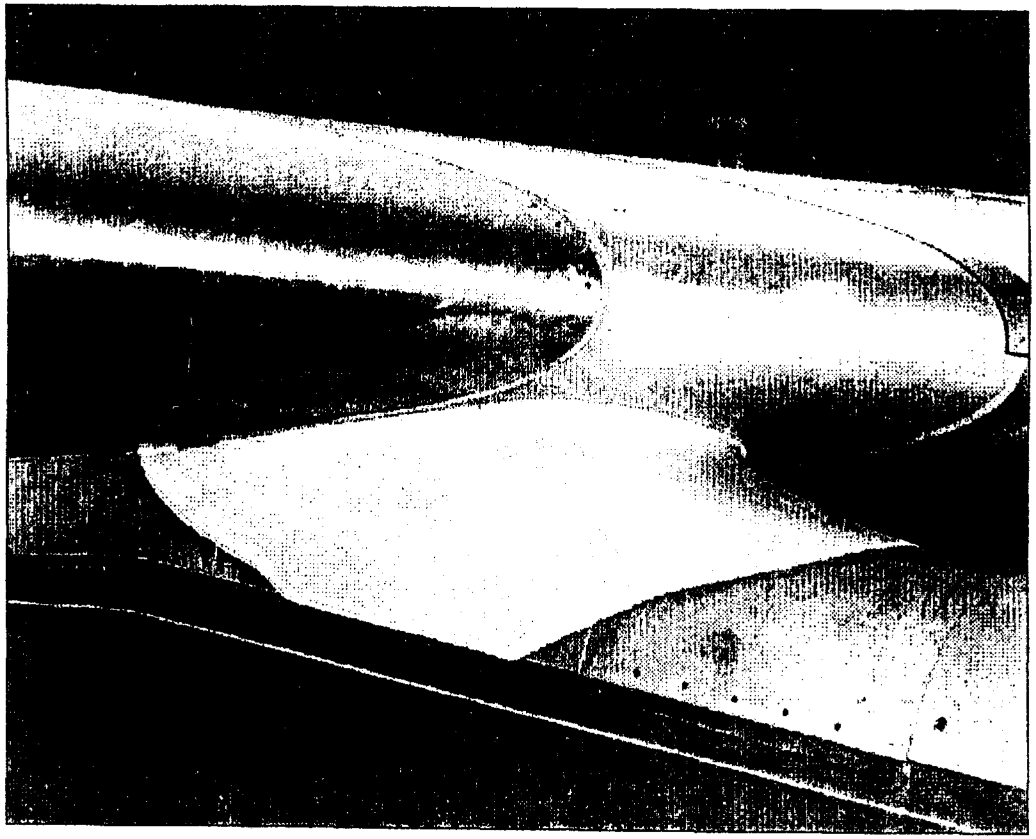


WITH FILLET

The MD-11 Cruise Performance Improvement



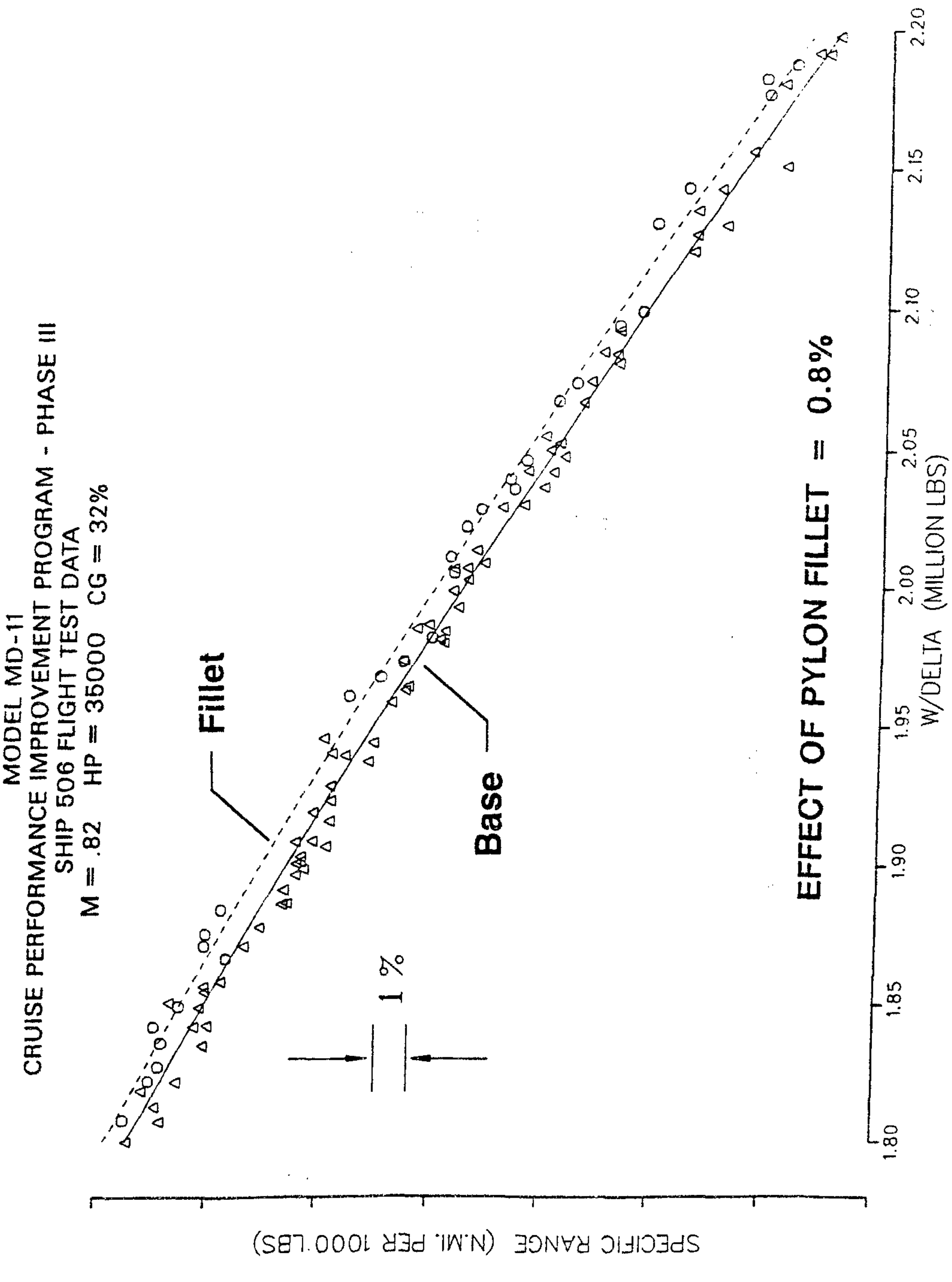
Pylon Fillet
Drag Reduction Program



MD-11 Cruise Performance Improvement Program

Ship 506 Flight Test Summary Log

Flight #	Flight Time (Hr:Min)	# Data Points	Test Configuration
10	10:15	37	Baseline (CPIP I + II)
12	7:40	44	Baseline (CPIP I + II)
13	7:00	31	Baseline (CPIP I + II)
17	11:00	47	Baseline + Pylon Fillets
4 Flights	35:55	159	


MODEL MD-11
CRUISE PERFORMANCE IMPROVEMENT PROGRAM - PHASE III
SHIP 506 FLIGHT TEST DATA
M = .82 HP = 35000 CG = 32%
Fillet
Base
1 %
EFFECT OF PYLON FILLET = 0.8%
1.80
1.85
1.90
1.95
2.00
2.05
2.10
2.15
2.20
W/DELTA (MILLION LBS)
SPECIFIC RANGE (N.MI. PER 1000 LBS)

Summary

- Flight Test Flow Visualization Identified Wing / Engine Pylon as Region of Separated Flow
- CFD Analysis Duplicated Flight Observation and Provided a Tool to Design Pylon Modifications
- Final Pylon Modification Developed Using CFD Virtually Eliminated Separated Flow Region
- Drag Reduction of 0.8% Measured in Back-to-Back Flight Test of Pylon Fillet
- Airline Experience Validates Pylon Fillet Benefit

OPTIMIZATION OF AGRICULTURAL AEROPLANE HANDLING QUALITIES

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For an agricultural aeroplane the specific flight conditions are typical: repeated flight conditions at an extreme low altitude with the need for the constant altitude and flight direction control when repeatedly making steep turns and terrain following. It is extremely exhausting for pilots to perform these regimes. Because of this, not only the special pilot training but also the provision of these (optimal) controlability characteristics as well as such control methods are necessary which would increase in flight safety, extremely unload the pilot and the aeroplane control would be less exhausting for him.

In this connection, of practical importance is the consideration of the following problems concerning the stability, controllability and safety of agricultural aeroplanes:

— determination of safe minimum flight altitude/speed in a low-altitude flight,

— evaluation of permitting stability/controllability characteristics and center-of-gravity position variation as well as evaluation of characteristics of an aeroplane when instantly releasing the whole of amount of chemicals,

— working out of recommendations on improvement in flight safety, provision of satisfactory stability/controllability characteristics and improvement of aeroplane piloting methods when studying measures on decrease in the load on a pilot.

The objective of this paper is the consideration of the problem on optimal longitudinal/lateral control characteristics of an agricultural aeroplane, development of methods for calculation of this characteristics and elaborating the requirements for elevator/ailerons hinge moments.

As controllability characteristics such generally accepted parameters are considered here as follows:

X_{n_z} — displacement of the longitudinal control lever per unit of g-load factor;

F_{n_z} — consumption of longitudinal control forces per unit of load factor, (command gradient);

X_{ap} — displacement of lateral control lever per unit of roll rate;

F_{ap} — consumption of lateral control forces per unit of roll rate.

Experimental investigations on determining optimal characteristics of the agricultural aeroplane controllability were carried out on a PC-101 simulator of TsAGI (without a motion system) when using a TV visual system with the participation of operator-pilots having the great professional experience of the light aeroplane flight and the long service at simulators.

The pilot task is to make a lateral S-maneuver 50 m altitude descent, flare-out and flight on runway (from its threshold) at a low altitude being chosen at pilot's discretion. The task was assumed to be completed after making an initial phase of the aeroplane entry into 1.5 to 2.0 g-load turn.

As a criterion of evaluating handling qualities the PR evaluation of an aeroplane by a pilot (by the Cooper-Harper pilot rating) was used. Equations of aeroplane motion written down within generalized parameters as increments to the initial trim condition of the flight were modeled in the ordinary form (at $V = const.$). As the initial trim condition a straight-and-level flight with wings level and without roll and slip was assumed. Control stick forces were modeled (at $F_0 = 0$) as follows:

$$F = F^X + F_{fr}$$

where F^X is the stick force gradient and F_{fr} is the dry friction level.

The controllability characteristics (the derivatives of $X_{n_z}, F_{n_z}, X_{ap}, F_{ap}$) were varied at the expense of gear ratios K and stick force gradients. On the basis of experimental results an optimum controllability characteristic range was determined by the best pilot rating (by min PR).

The methods of calculating the optimum controllability characteristics of an aeroplane and the required hinge moments of control surfaces are based on known A- and Z- criteria.

In accordance with the A-criterion the values of optimum controllability characteristics for different dynamic parameters of an aeroplane can be determined by the calculation under conditions:

$$|W(j\omega_*)|^{-1} = A \qquad (1)$$

where $|W(j\omega_*)|$ is the value of amplitude-frequency characteristics of an aeroplane at a distinctive control frequency ω_*. A is the constant depending on the feel system characteristics and type of a control stick and on the piloting task in the general case.

As an amplitude-frequency characteristic $|W(j\omega_*)|$ the linear combination of the following amplitude-frequency characteristics of transfer functions the linear combination of the following amplitude-frequency characteristics of transfer functions was assumed:

— for a longitudinal control channel;

$$|W(j\omega)| = |W_{n_z}(j\omega)| + \frac{V_0}{g}|W_q(j\omega)|$$

— for a lateral control channel,

$$|W(j\omega)| = |W_p(j\omega)| + \chi|W_r(j\omega)|$$

— where V_0 and χ are weight factors.

Under this presentation of amplitude-frequency characteristics A-criterion can be presented in form which is suitable for calculating the optimum derivatives x and x as follows:

— for a longitudinal control channel;

$$X_{n_z}^{opt} = -A_e\omega_0^2 \frac{1 + \frac{V_0}{gn_{za}}\sqrt{\omega_{*e} + \left(\frac{n_{za}g}{V}\right)^2}}{\sqrt{(\omega_0^2 - \omega_{*e}^2)^2 + (2\xi\omega_0\omega_{*e})}}$$

— for a lateral control channel,

$$X_p^{opt} = -A_a \frac{1 + \frac{g}{V}\chi\frac{1}{\omega_{*a}}}{\sqrt{T_R^2\omega_{*a}^2 + 1}}$$

According to the z-criterion the optimum controllability characteristics of an aeroplane for various stick feel system characteristics can be calculated from relationships (at $F_0 = F^x = 0$):

$$\overline{X}_i^{opt} = \frac{F^x(F_* - F_{fr}) + (1 + cF^x)(X_* - cF_{fr})}{\left[(F^x)^2 + K(1 + cF^c)^2\right]X_*}$$

$$\overline{F}_i^{opt} = \overline{X}_i^{opt}F^x \text{ at } i = n_z, p$$

where, $\overline{X}_i^{opt} = \frac{X_i^{opt}}{X_{*i}^{opt}}$, X_{*i}^{opt} is the optimum value of the derivative with no artificial feel of the control stick, i.e. at $F^x = F_{fr} = 0$, F_* and X_* are desired levels of control force/displacement when a pilot evaluates handling qualities.

Expressions (2), (3), (4) make it possible to calculate optimum values of derivatives $X_{n_z}^{opt}, F_{n_z}^{opt}, X_p^{opt}, F_p^{opt}$ depending on dynamic parameters of an aeroplane, its flight regimes and stick feel system characteristics. But for making these calculations the constants A_e, A_a, ω_{*e},

ω_{*a}, F_*, X_* entering into the expressions must be known. The constants mentioned above were found from condition that the derivatives $x_{n_z}^{opt}, x_{ap}^{opt}$ calculated in terms of A- and Z-criteria agree best with the derivatives obtained in this experiment.

The results of the experiment by evaluating the optimum control characteristics of an aeroplane and the results of the calculated characteristics in terms of A- and Z-criteria are given in Figs. 1-3. Fig. 1 shows the optimum derivatives $X_{n_z}^{opt}$ depending on the aeroplane g-load factor sensitivity to angle of attack n_{z_α}n variation, and Fig. 2 shows the derivatives depending on the force gradient of the wheel F^x. Fig. 3 shows values $\dot{p}_{X_a}^{opt}$ of the optimum roll acceleration sensitivity of the aeroplane to control wheel movement depending on the roll time constant T_r. These Figs. show ranges of controllability characteristics $X_{n_z}^{opt}, \dot{p}_{X_a}^{opt}$ (optimal in pilot's opinion) and the characteristics calculated in terms of A- and Z-criteria.

The experimental and calculated data given in Figs. 1-3 confirm the known regularity of variation in mentioned controllability characteristics with flight regimes, dynamic characteristics of an aeroplane and a wheel force gradient. It is seen also that the optimum control characteristics calculated in terms of A- and Z-criteria agree well with experimental data.

By this means the calculation of optimum longitudinal/lateral control characteristics of an aeroplane would be appropriate for making as follows:

1. For given flight conditions ($H, V, \overline{X}_T, G$, moments of inertia) and at known characteristics of the control system one calculates:
   - dynamic parameters $\omega_0, n_{z\alpha}, T_r$ of an aeroplane;
   - pitch/roll force gradients F^x.
2. From plots $A_e(F_e^x), A_a(F_a^x)$ (these relationships were obtained experimentally and by calculation in terms of A- and Z-criteria) given in Figs.4 and 6 the constants (A) and (A) are determined depending on a type of the control lever, its force feel gradient and friction value in the control system.
3. From plots given in Figs. 5 and 7 (the plots were obtained by calculation of expressions (2) and (3) — i.e. conditions of A-criterion) the values of parameters $\left(X_{n_z}^{opt}/A_e\right)_0$ and

$\left(\dot{P}^{opt}_{X_a} A_a\right)_0$ are found depending on flight velocity and dynamic parameters of an aeroplane.

4. The optimum controllability characteristics are determined as

$$X^{opt}_{n_z} = (A_e)_0 \left(X^{opt}_{n_z} / A_e\right)_0 \quad ,and \quad F^{opt}_{n_z} = X^{opt}_{n_z} F^x_e$$

$$X^{opt}_{ap} = (A_a)_0 \Big/ \left(\dot{P}^{opt}_{X_a} A_a\right)_0 T_r, \quad and \quad F^{opt}_{ap} = X^{opt}_{ap} F^x_a$$

In this case the optimum force gradients of the wheel/stick in the experiment were as follows:

$F^x_{e_{opt}} = 0.08—0.26$ kg/mm and $F^x_{a_{opt}} = 0.07—0.22$ kg/mm.

The optimum relation of the longitudinal force gradient to the lateral force gradient was as follows:

$F^x_{e_{opt}} / F^x_{a_{opt}} = 1.3—2.0$ both for the wheel and for the stick.

For determining the required hinge moments of elevators/ailerons we proceed as follows:

1. From Figs. 5 and 7 we find the values of parameters $\left(X^{opt}_{n_z} / A_e\right)_0$ and from given V_0, ω_0, n_{z_a}, T_r we find $\left(\dot{P}^{opt}_{X_a} A_a\right)_0$.
2. We calculate the required values of constants $(A_e)_0$ and $(A_a)_0$ from expressions:

$$(A_e)_0 = X_{n_{zAero}} \left(X^{opt}_{n_z} / A_e\right)^{-1}_0 and (A_a)_0 = \frac{1}{\dot{P}_{X_{aAero}} \left(\dot{P}^{opt}_{X_a} A_a\right)}$$

3. From plots in Figs. 4 and 6 from found values of $(A_e)_0$ and $(A_a)_0$, control lever type and friction values in the control system we determine the required force gradients F^x_e and F^x_a in longitudinal and lateral channels.
4. From known formulae we calculate the required elevator/ailerons hinge moments providing the required force gradients.

The calculations which are related to the choice of elevator/ailerons hinge moment coefficients are made for regimes of the main use of an agricultural aeroplane. Initially it is essential that the calculation must be made without regard to control system springs, i.e. we should seek to provide the controllability characteristics to be near-optimal only at the expense of control hinge moments.

A correlation of results of calculating optimum controllability characteristics of an agricultural aeroplane with FAR-23 and MIL-F-8785c standards for the longitudinal/lateral control effectiveness and for control lever forces has shown that the calculations of the longitudinal/lateral controllability of an agricultural aeroplane with proposed methods make it possible to fulfil FAR-23 and MIL-F-8785c standards for light civil aeroplanes. In this case the calculated curves are within the level I of handling qualities.

In conclusion it may be said that the proposed methods are applicable for calculating the optimum controllability characteristics and required hinge moments of controls not only for an agricultural aeroplane but also for light civil aeroplanes because of similarity of tasks for piloting the aeroplanes of the same class.

The proposed methods are most simple and attractive for the use in the computer-aided design.

These methods are universal in that one can use them for calculating optimum controllability characteristics practically for any aeroplane class providing the constants $A_e\left(F_e^x, F_{fr}, \ldots\right)$ and $A_a\left(F_a^x, F_{fr}, \ldots\right)$ are known; the constants depend on the task of piloting the aeroplanes as well as on the feel system characteristics and type of the control lever.

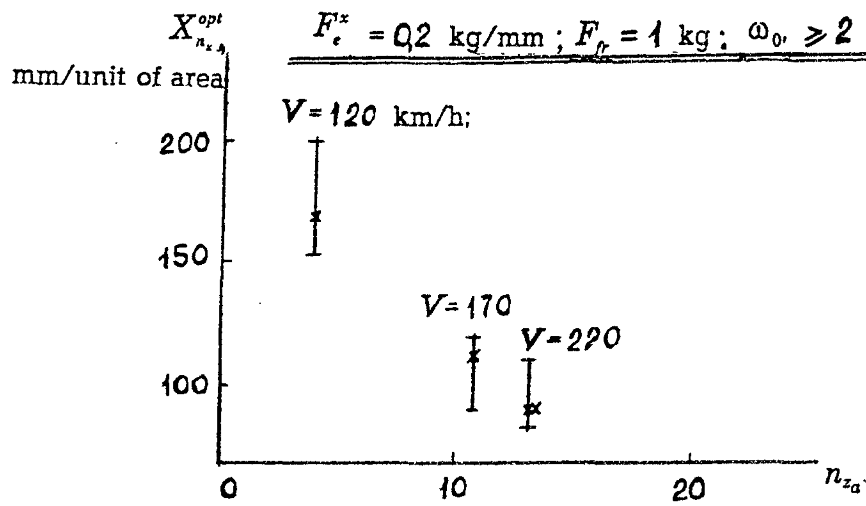

Fig. 1

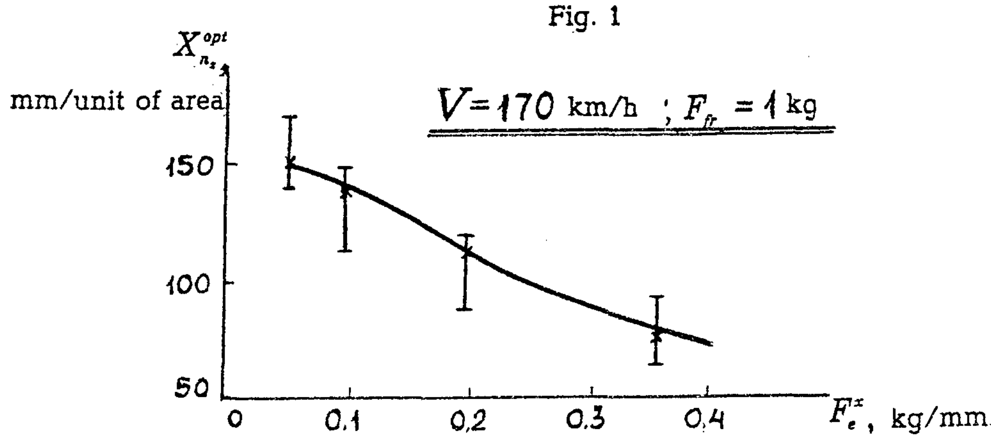

Fig. 2

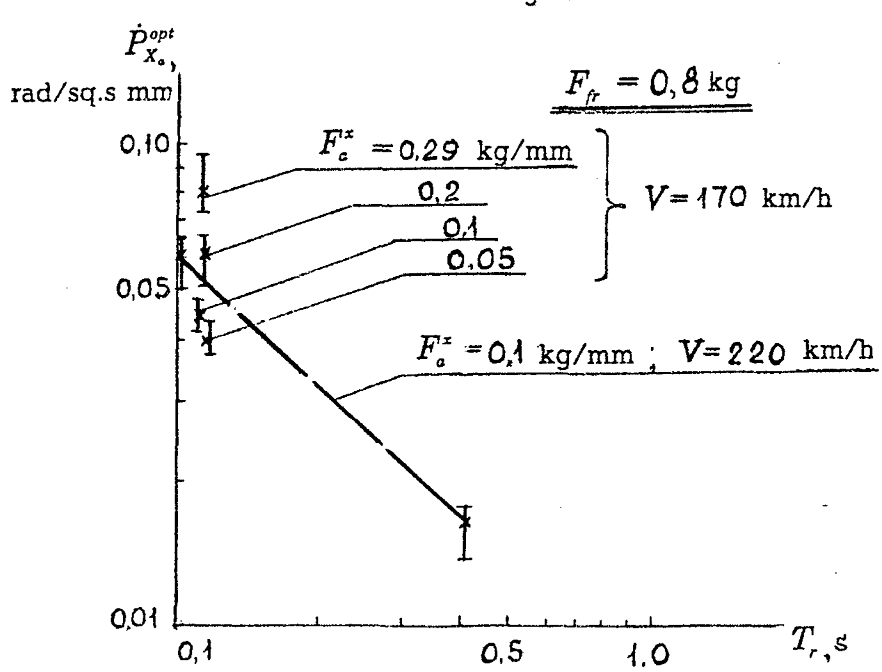

Fig. 3

x — A- and Z- criteria

I — range of optimum values of x opt from the experiment

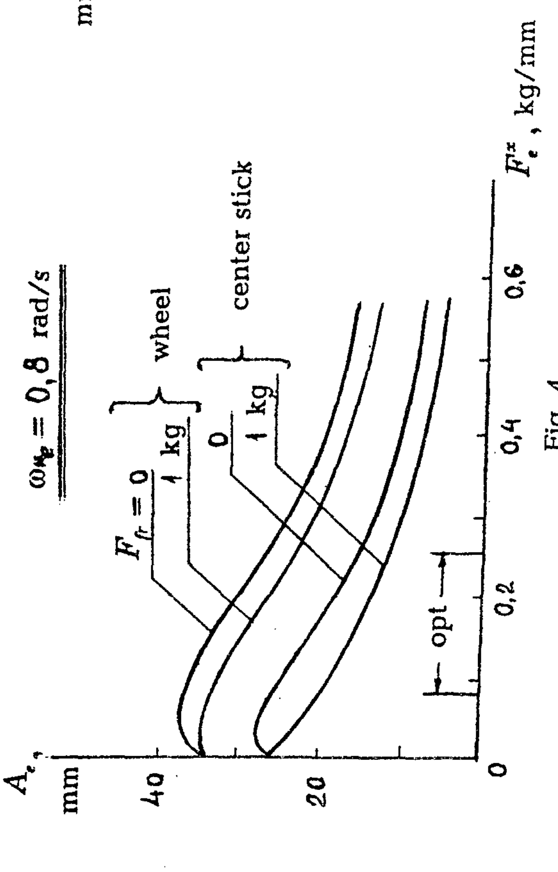

Fig. 4

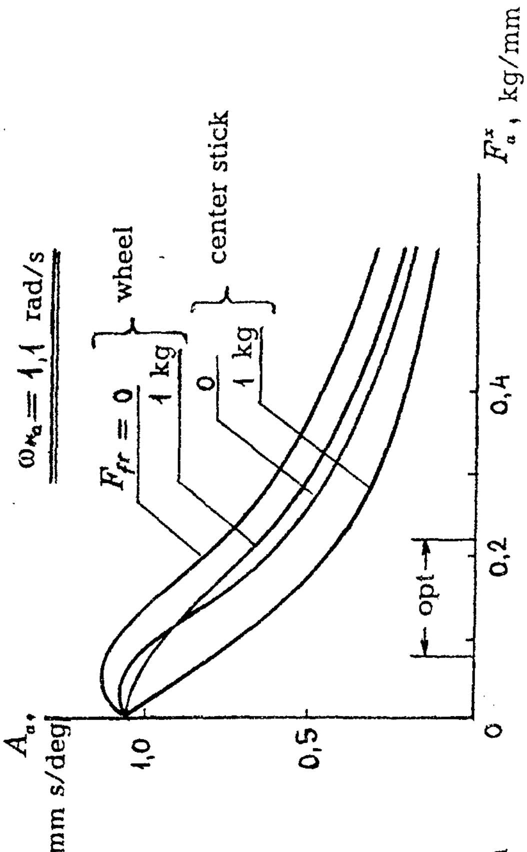

Fig. 5

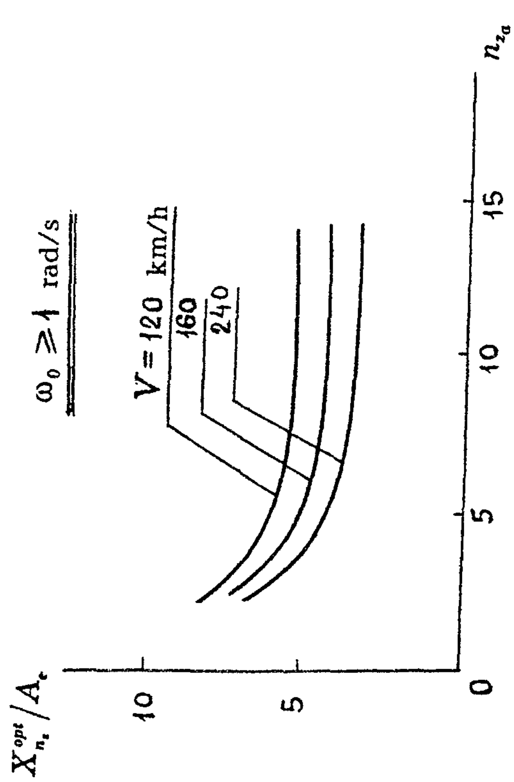

Fig. 6

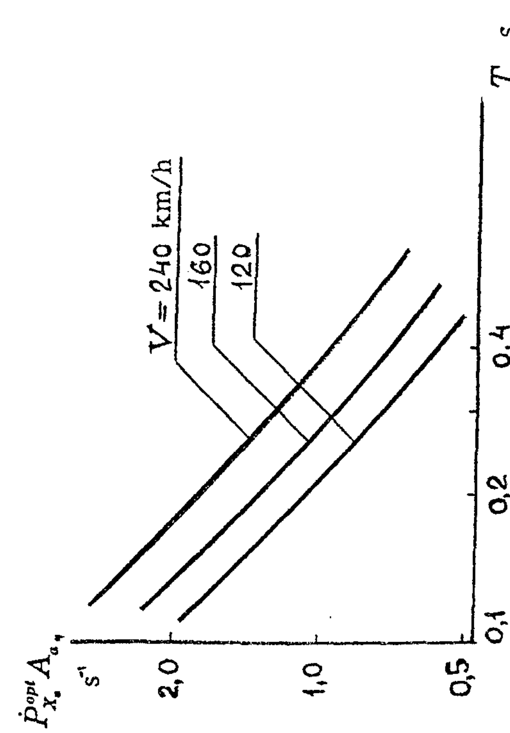

Fig. 7

TECHNIQUE ASPECTS OF FLIGHT SIMULATION

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TsAGI, Zhukovsky, Russia

Introduction

As it is known, flight simulators are used through all stages of aircraft development: from fundamental researches to the investigations connected with airplane exploitation and maintain. Since a great number of controllability characteristics and variety of possible failures in control system are typical for the modern aircraft, it is impossible to investigate all of them in flight tests only. Therefore, customer and manufacturer have to use to a large extent the flight simulators for the certification as well.

Nevertheless, flight simulators potentialities are not determined yet in full measure. In many respects the wider using of flight simulators for the certification is restrained by the necessity to solve a number of technique problems of flight simulation.

The main problem is a problem of conformity between simulation results and flight test results. In other words, it is a problem of how an accuracy of results depends on flight simulator characteristics. No one of ground simulators permits us to reproduce flight as such, either on cost reason or in essence. Therefore, it is important to answer to this question for both to validate the requirements for flight simulators and to know where we may use simple simulators, and where complicated simulators and flight tests are needed.

The second problem is a problem of processing the results obtained in simulator experiments. The problem is associated with the fact that the piloting has complicated, nonstationary and stochastic nature.

It is impossible to consider the entire variety of these problems in one paper. So, only some our results and directions of our further work will be presented below.

1. VISUAL CUEING SIMULATION

At present the basic mean of visual cueing simulation has become the systems with computer generated image. The main problem of systems of such a type is formatting a wide field-of-view image which would have high resolution and high dynamic properties.

Therefore, it is necessary to know how these characteristics affect piloting and what requirements they must meet.

Our investigations have shown (fig. 1) [ref. 1] that field of view influences greatly the roll mode and piloting along linear degrees of freedom when image angular dimensions are just up to 45 in horizontal and vertical planes. Therefore, the image angular dimensions of about 45 would be sufficient for simulation of almost all piloting tasks excluding those associated with searching a target.

A visual system time delay inserts additional phase and amplitude distortions into modeled aircraft control loop. From different Standards it is known that control system time delay must not exceed 0.1 sec to hold aircraft controllability on a high level (Level 1). So, a visual system time delay must be much less than 0.1 sec. The results we obtained cause us to anticipate that the visual time delay must not exceed 0.03—0.05 sec.

An image resolution can cause a considerable influence on piloting as well. In fig. 2 there is presented the probability of detecting the aircraft attitude variation against image resolution. Based on these results one could consider the image resolution value to be no more than 0.05.

Having regard to this consideration a special low-cost visual system with computer generated image has been developed in TsAGI. The system consists of three windows with collimation systems. The system creates an image with angular dimensions of about 45°; resolution and time delay are close to the values mentioned above.

2. MOTION CUES SIMULATION

It is one of most important scientific problems of modern flight simulators. Flight simulators in their technical aspects have almost attained perfection. There are expensive, of million dollars, systems, which transfer simulator platform by several meters. However, the question of how much the expenditure is justified, is not clear yet. To answer to this and other problems of motion cueing simulation it is necessary to answer to the following questions.

First of all, it is necessary to know a role of motion cues in different piloting tasks. The examples presented in fig. 3 show that influence of motion cues can be significant. For example, motion cueing leads to decreasing a roll stabilization error (roughly in 1.5—2 times), touchdown vertical velocity and an altitude stabilization error at hovering.

The role of motion cues is particular considerable while modeling PIO phenomenon both of low-frequency [ref. 2] and high-frequency types, as it is presented in fig. 4. It is seen from here that high-frequency oscillations which arise at some characteristics of a manipulator, roll mode time constant or aircraft elasticity, can be reproduced in moving-base simulator only. This type of oscillations can not be reproduced in fixed-base simulator in principle.

At the same time, in some cases motion cues do not cause any influence on piloting. Obviously, if motion cues do not exceed pilot thresholds, they may be not modeled. For example, angular rates of civil aircraft, if there is no failure, are usually less than sensitivity thresholds presented in fig. 5. Therefore, while modeling ordinary regimes, angular accelerations may be not reproduced. It appears also that pitch accelerations do not affect piloting practically, since perception of the accelerations is hampered by *g*-loads acting simultaneously with the pitch accelerations.

Thus, motion cues influence depends on many factors. In some cases it can be considerable, in others it is negligible. It depends on piloting task, aircraft dynamic performance, level and frequency composition of accelerations. Based on wide experimental researches we developed several rules which can be used to estimate the necessity of motion cues simulation in every specific case [ref. 1]. However, the problem of effect of motion cues on piloting needs its further investigation.

To solve this and a number of other problems it is necessary to study regularities of motion cues perception. At present well-studied are the regularities of perception of motion cues along separate degrees of freedom. However, at piloting a pilot experiences the accelerations acting along different degrees of freedom simultaneously. As our results show, the accelerations acting along a certain degrees of freedom affect greatly pilot's sensitivity to accelerations along other degrees of freedom:
where
— angular acceleration threshold at g-load equal to 1;
— rms of linear accelerations.
For example, because of *g*-loads with the level of 0.05, sensitivity thresholds of angular motion increase in 2.5 times in comparison with the case where *g*-loads are absent.

We have also determined pilot's differential thresholds to the normal acceleration (fig. 5). It appears that their values are about 10—15 % to the normal acceleration. However, these data were obtained against the backgrounds of very small *g*-load.

In our further researches we would like to investigate the influence of large accelerations on angular motion perception and pilot's sensitivity to acceleration variation. The results obtained in these investigations would allow us to answer to the question of necessity to simulate motion cues (pitch, roll accelerations) against back-grounds of large linear accelerations. This point is very important since in conditions of a ground flight simulator we can not reproduce large accelerations and, so, we have no the right to reproduce the high-frequency component only. On a centrifuge we can reproduce large accelerations but can not reproduce properly the high-frequency accelerations.

Simulation quality depends in many respects on platform motion control laws. There are different principles based on which the control laws are forming. These principles are expounded in many scientific works including our works. However, the potentialities of the methods are not sufficiently studied yet. Usually, selection of control laws are based on the previous experience, by way of "trial and error". We do not know enough in theory to solve rigorously the problem of optimization of control laws having regard to simulator characteristics and investigation tasks.

Our investigations show that large errors in motion cues reproducing can distort the simulation results to a greater extent then the absence of platform motion. This brought us up to special experiments to learn what requirements the motion system must meet (i.e., the number of degrees of freedom, platform displacement, dynamic performance).

3. SIMULATION OF MANIPULATOR LOADING SYSTEM

A task of manipulator loading system simulation can be solved rather completely with a help of electrohydraulic loading systems. The systems of such a type are widely used in modern flight simulators. We have an experience in development and manufacture of such systems to order of different firms including the foreign ones.

Our loading system is considerably cheaper than foreign analogies. Nevertheless, electrohydraulic loading system is a complicated and expensive device. For low-cost systems the cheaper loading systems are needed, that leads to deterioration of accuracy and completeness of forces to be reproduced. We carried out the extensive investigations of influence of all basic loading characteristics on aircraft controllability: force gradient, breakout force, loading damping, — for various types of manipulators — a wheel, side and center stick, mini wheel.

4. RESULTS PROCESSING PROCEDURE

Value and trustworthiness of the results obtained on flight simulators depend considerably on procedure of their processing. In certification investigations there are used the objective controllability parameters which are stated in Standards. However, since criteria of controllability are not absolute yet, aircraft handling qualities are estimated finally on the basis of pilot opinion analyzing. To analyze uniformly a pilot opinion and to compare opinions of different pilots, pilot rating scales are used. We use mainly Cooper-Harper scale. Pilot ratings contain often more information than objective estimation of piloting process, since pilot has in mind all factors affecting the controllability, including piloting precision and his own mental and physiological workload.

Pilot ratings obtained in the experiments inevitably have a certain range of variation. In this connection a number of questions arises: what pilot rating we should consider to be a true rating, how the true rating depends on a number of replicate experiments, how much pilots we need in order to obtain the true rating, and so on. We developed and validated a procedure of pilot ratings processing, which answers to these questions [ref. 3]. We carried out extensive experimental investigations to determine the range of pilot ratings variation, determined the laws of pilot ratings probability distribution (fig. 6) (it is the binomial law) and developed a procedure of determination a pilot rating confidence interval and a needed number of experiments.

Conclusion

In summary it may be said that flight simulators efficiency is reduced considerably because of a set of technique problems of simulation which should be solved. Expenses needed for these purposes, in our opinion, are much less than expenses on flight simulator design and improvement. But these expenses would bring to us no less effect than expenses on improvement of technical characteristics of flight simulators.

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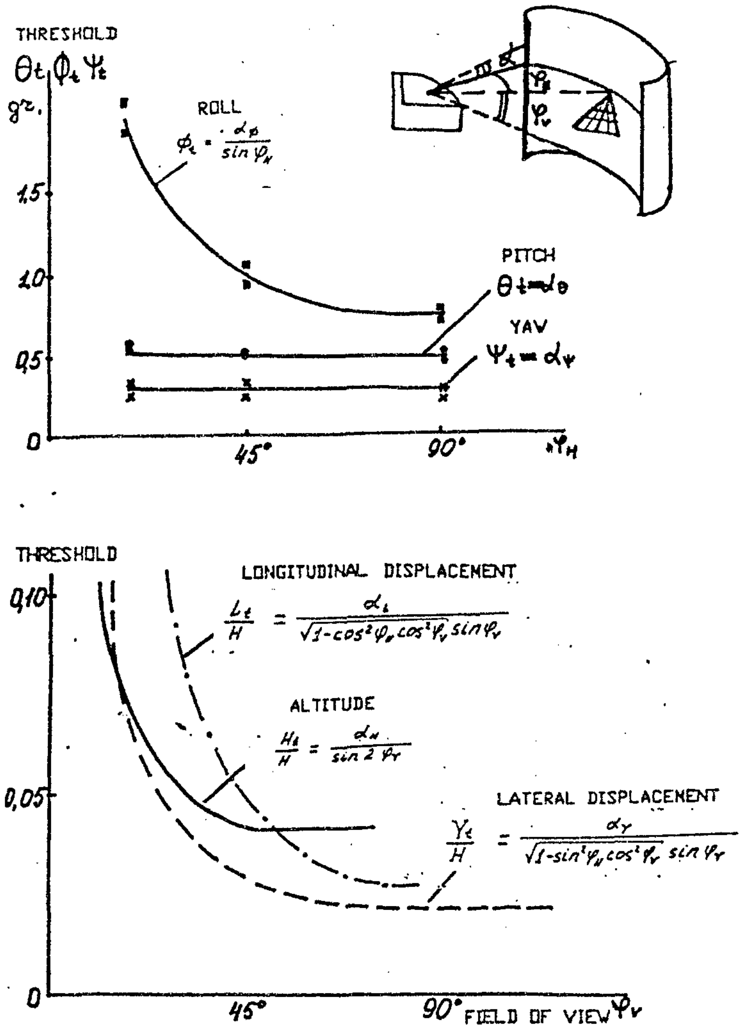

Fig.1 Field of view affecting motion perception.

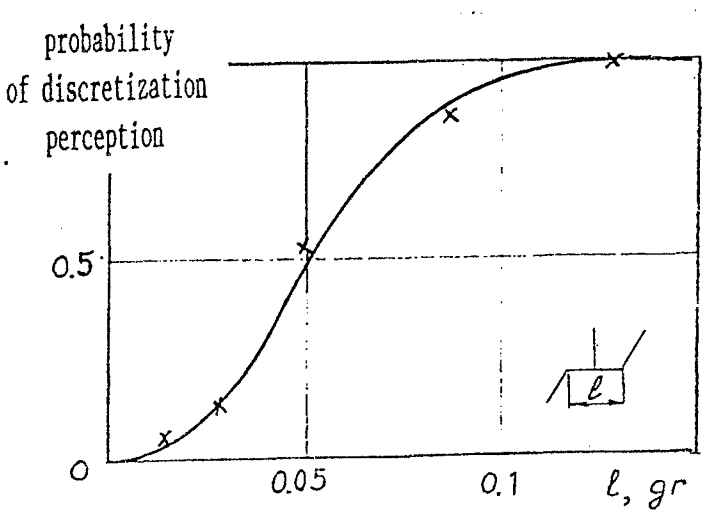

Fig.2 Detecting the aircraft attitude variation against image resolution.

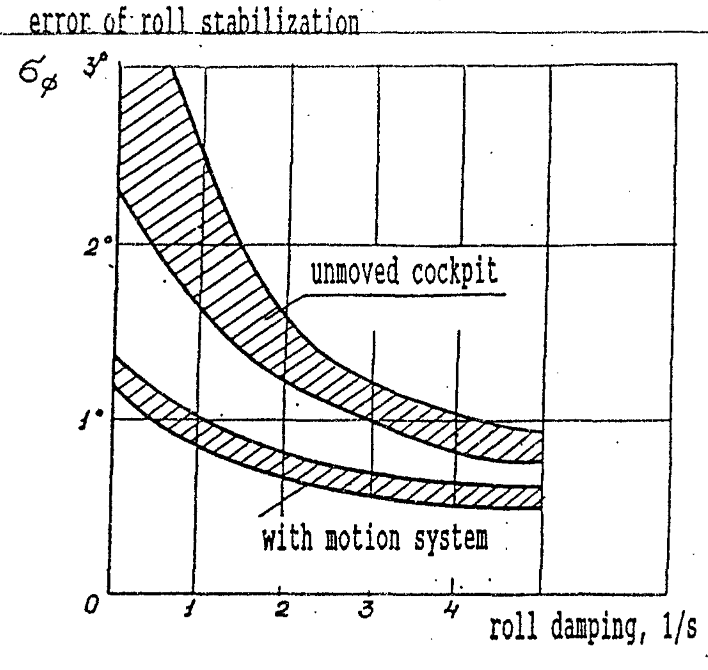

Fig.3 Influence of motion cues on piloting.

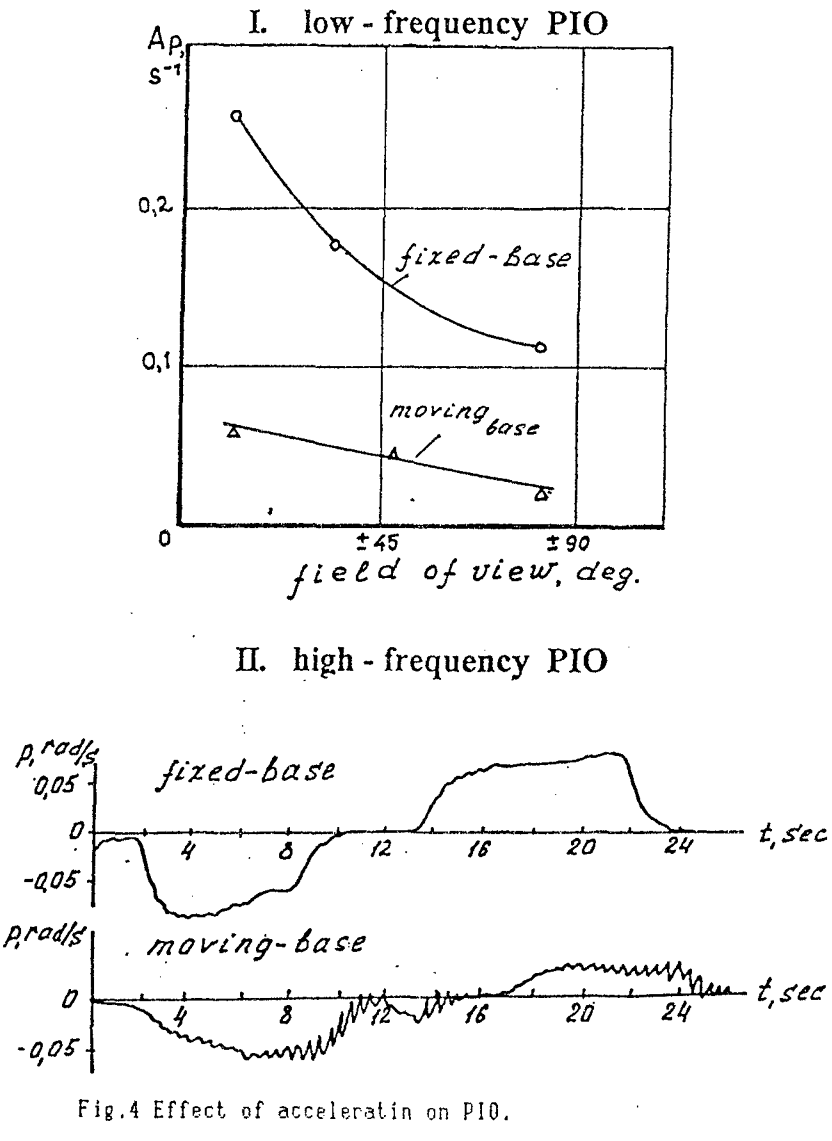

Fig.4 Effect of acceleratin on PIO.

absolute thresholds

	RATE OF ANGULAR MOTION, deg/s:		
	ROLL	PITCH	YAW
SCARCELY MARKED MOTION	2.7	1.5	0.6
MOTION DIRECTION IS DIFFERENT	4.5	2.2	0.9

differential thresholds

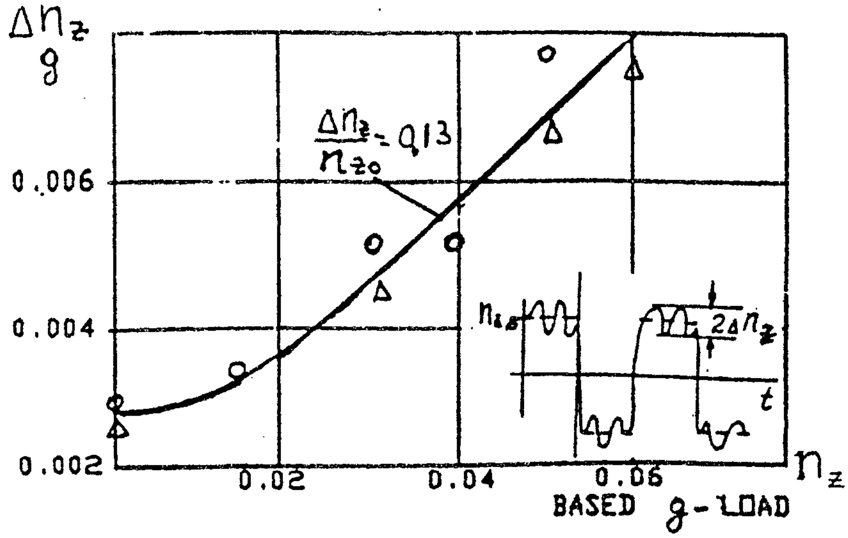

Fig.5 Motion perception thresholds.

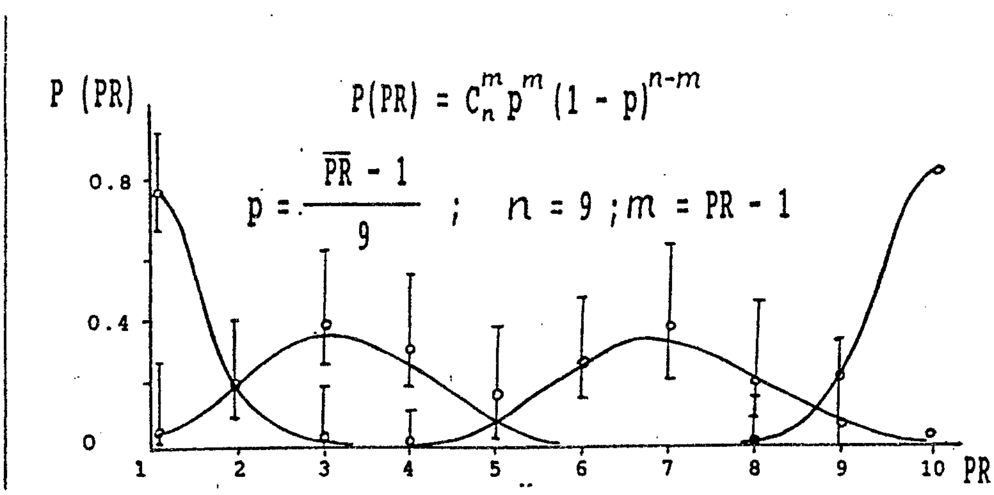

Fig.6 Pilot rating distribution law

SIMULATION OF MiG-21 (BIS) FLIGHT DYNAMICS

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Abstract- The paper presents mathematical model for MiG-21 vehicle dynamics. The model is trimmed in different flight conditions. The vehicle response in steady state flight conditions and to control actuations is simulated. The model is linearised and modal decomposition is applied to demonstrate the natural modes of MiG-21. Important transfer functions for the vehicle dynamics are identified and analysed. The model and simulation exercises are validated by flight test data.

NOMENCLATURE

A	Slope of drag polar.
C_X, C_Y, C_Z	Drag, side force and lift coeff.
C_{Xbase}	Base component of drag coeff.
C_{Xi}	Induced drag coefficient.
C_{Xstab}	Drag coeff. due to stabiliser deflection.
$C_{Y\beta}$	Side force coeff. due to side slip.
C_{Yrud}	Side force coeff. due to rudder deflection.
$C_{Z\alpha}$	Lift coeff. due to angle of attack.
C_{Zstab}	Lift coeff. due to stabiliser deflection.
m_X, m_Y, m_Z	Roll, pitch and yaw moment coeff.
$m_{X\beta}$	Roll moment coeff.due to side slip.
m_{Xail}	Roll moment coeff. due to aileron deflection.
m_{Xrud}	Roll moment coeff. due to rudder deflection.
$m_{Y\alpha\phi_o}$	Pitch moment coeff. at zero roll angle due to angle of attack.
$m_{Y\alpha_o}$	Pitch moment coeff. at zero roll angle and at zero angle of attack.
m_{Ystab}	Pitch moment coeff. due to stabiliser deflection.
$m_{Z\beta}$	Yaw moment coeff. due to side slip.
m_{Zail}	Yaw moment coeff. due to aileron deflection.

m_{Zrud}	Yaw moment coeff. due to rudder deflection.
stab,ail,rud	Stabiliser, aileron and rudder deflection angle.
X,Y,Z	Drag, side force and lift acting on the aircraft.
L,M,N	Total roll, pitch and yaw moment of the aircraft.
m	Mass of the aircraft.
I_{XX}, I_{YY}, I_{ZZ}	Inertia moments.
M	Mach no.
T	Net thrust.
T_{XW}, T_{yW}, T_{zW}	X, Y and Z components of net thrust in Wind Axes reference frame (F_W).
V	True air speed.
X_E, Y_E, Z_E	Aircraft displacement in Earth fixed reference (F_E).

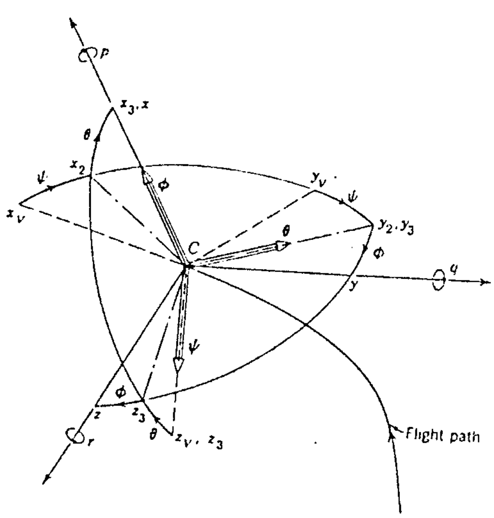

Fig. 1. Euler angles.

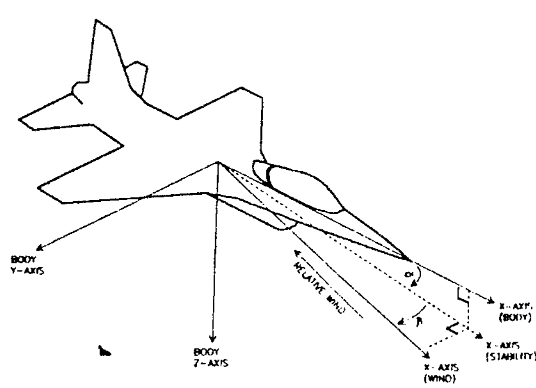

Fig. 2. Aircraft fixed reference frames and aerodynamic angles.

I. INTRODUCTION

The application of simulation to Avionics Systems has traditionally concentrated on the development of flight control systems and of flight deck simulators for the training of pilots. It is often required to study the interaction between elements of Avionics Systems, Vehicle Dynamics and Outside World. Simulation provides a useful and cost-effective method for such study. Aircraft model plays vital role in Avionics System Simulation;in fact, simulation is one of the most important model based studies. Aircraft Model is a mathematical description of an aircraft flight dynamics based on solution of set of simultaneous non-linear differential equations.

When discussing a modelling of an aircraft in relation to its use in avionics simulations, it is usually a requirement to embed a representation of an aircraft flight dynamics within total control system. In this way total aircraft response to pilot inputs is modelled. The study of Aircraft Control System and its representation in simulation is a vast topic. The modest aim of the present study is to model MiG-21(BIS) rigid body dynamics, simulate its open loop response during non-terminal steady state flight phases and demonstrate the natural modes for this aircraft. The mathematical model and simulation exercises are validated by flight test data.

II. MODEL DEVELOPMENT

The total exercise of mathematical model development for MiG-21(BIS) aircraft is based on the broad assumptions that an aircraft is a rigid body with six degree of freedom, aerodynamic flow field is quasi-stationary, gravitational acceleration (g) remains constant throughout the flight envelope and Environment of an aircraft satisfies the standard atmospheric conditions and it is at rest relative to Earth.

The total take-off weight of this aircraft is assumed at 8799Kg; this includes fuel capacity and weight of armament stores. The mass of the aircraft has been modelled as a function of specific fuel consumption of R-25 engine. Inertia Moments are modelled as linear functions of fuel burn-out. The cross product of inertia moments are neglected as MiG-21 is a symmetric vehicle about its longitudinal axis.

Atmosphere for the aircraft has been modelled based on [1]. Troposphere of Indian Standard Atmosphere (ISA) has been used for the model development. By characterising temperature distribution, pressure and density at any altitude are computed.

R-25 engine fitted on MiG-21 aircraft is a turbojet engine with an afterburner. Dimensional Curves for ISA conditions have been used for Engine Model. Net Thrust and Specific Fuel Consumption (sfc) for various altitudes and mach numbers corresponding to different engine ratings have been computed. Here, our main interest is in steady state performance of the engine.The steady state values of net thrust and sfc are taken from series of look-up tables by linear interpolation between two break points.

The aerodynamic model has to produce aerodynamic forces and moments acting on the aircraft. Nondimensional aerodynamic coefficients used for this model development were available from the wind tunnel tests, carried out earlier. Wing Reference Area (S), Mean Aerodynamic Chord Length ($\bar{C}$) and Wing Span (L) are taken as $23m^2$, 4.002m and 7.15m respectively. Force Coefficients are given in the Wind Axes System and Moment Coefficients are given in Body Axes System Aircraft cetre of gravity (c.g.) is assumed to be fixed at $\bar{X}_{c.g.} = 0.32\bar{C}$ in Body Axes reference frame. The coefficients are modelled for angle of attack ranging between -5^{o} to 25^{o}, stabiliser deflection -16^{o} to 8^{o}, aileron deflection between -20^{o} to 20^{o} and rudder deflection between -25^{o} to 25^{o}. The coefficients are modelled as function of mach number. The entire speed range has been divided into subsonic, transonic and supersonic speeds. The component build up for the coefficients has been done as shown below:

$$C_X = C_{Xbase}(M) + C_{Xi}(A, C_Z) + C_{Xstab}(stab, C_Z) \quad \text{(a)}$$

$$C_Y = C_{Y\beta}(M) * \beta + C_{Yrud}(M) * rud \quad \text{(b)}$$

$$C_Z = C_{z\alpha}(\alpha, M) + C_{Zstab}(M) * stab \quad \text{(c)}$$

$$m_X = m_{X\beta}(\beta, \alpha) + m_{Xail}(ail, \alpha) + m_{Xrud}(rud) \quad \text{(d)}$$

$$m_Y = m_{Y\alpha\phi_o} + m_{Ystab}(M, stab) \quad \text{(e)}$$

$$\text{where } m_{Y\alpha\phi_o} = m_{Y\alpha_o}(M) + m_{Y\alpha}(M, \alpha)$$

$$m_Z = m_{z\beta}(\alpha, \beta) + m_{Zrud}(M, rud) + m_{zail}(M, ail) \quad \text{(f)}$$

Total roll, pitch and yaw moments are computed considering the damping effects.

The equations of motion for MiG-21 dynamics have been developed in the most general form of implicit state equations $f(\dot{X}, X, U) = 0$. The state variables identified for this purpose are:

(V, α, β) : Give translation of a vehicle relative to Earth fixed reference frame (F_E).

(p, q, r) : Give rotation of a vehicle relative to F_E .

(x_E, y_E, y_E) : Give position of a vehicle relative to F_E .

$(\phi\ \theta\ \psi)$: Give angular orientation of vehicle relative to Vehicle carried reference frame (F_V).

$(\phi_W\ \theta_W\ \psi_W)$: Give angular orientation of Wind Axes.

$\theta w = \gamma$ = angle of climb.

ψw = heading of the flight path.

$(p_W\ q_W\ r_W)$: Give angular velocity of Wind Axes.

The control vector { stab, ail, rud, net thrust (T) } has been considered for this model development. The approach followed for this model development has been discussed thoroughly in [7]. MiG-21 model equations based on the assumption that Earth is a stationary plane in inertial space are as follows:

$$p_W = p \cos\alpha \cos\beta + (q - \dot{\alpha}) \sin\beta + r \sin\alpha \cos\beta \tag{1}$$

$$q_W = -(1/m V) (T_{ZW} g - Z + m g \cos\theta_W \cos\phi_W) \tag{2}$$

$$r_W = (1/m V) (T_{YW} g - Y + m g \cos\theta_W \sin\phi_W) \tag{3}$$

$$\dot{p} = (1/I_{XX}) (L + (I_{YY} - I_{ZZ}) q r) \tag{4}$$

$$\dot{q} = (1/I_{YY}) (M + (I_{ZZ} - I_{XX}) r p) \tag{5}$$

$$\dot{r} = (1/I_{ZZ}) (N + (I_{XX} - I_{YY}) p q) \tag{6}$$

$$\dot{V} = (1/m) (T_{XW} g - X - m g \sin\theta_W) \tag{7}$$

$$\dot{\alpha} = q - q_W/\cos\beta - p \cos\alpha \tan\beta - r \sin\alpha \tan\beta \tag{8}$$

$$\dot{\beta} = r_W + p \sin\alpha - r \cos\alpha \tag{9}$$

$$\dot{\phi} = p + \sin\phi \tan\theta\, q + \cos\phi \tan\theta\, r \tag{10}$$

$$\dot{\theta} = \cos\phi\, q - \sin\phi\, r \tag{11}$$

$$\dot{\psi} = \sin\phi\, q/\cos\theta + \cos\phi\, r/\cos\theta \tag{12}$$

$$\dot{\phi}_W = p_W + \sin\phi_W \tan\theta_W\, q_W + \cos\phi_W \tan\theta_W\, r_W \tag{13}$$

$$\dot{\theta}_W = \cos\phi_W\, q_W - \sin\phi_W\, r_W \tag{14}$$

$$\dot{\psi}_W = \sin\phi_W\, q_W/\cos\theta_W + \cos\phi_W\, r_W/\cos\theta_W \tag{15}$$

$$\dot{x}_E = V \cos\theta_W \cos\psi_W \tag{16}$$

$$\dot{y}_E = V \cos\theta_W \sin\psi_W \tag{17}$$

$$\dot{z}_E = -V \sin\theta_W \tag{18}$$

III. TRIMMING THE AIRCRAFT MODEL

To trim the aircraft model in the steady state flight condition, the motion derivatives $\dot{V}$, $\dot{\alpha}$, $\dot{\beta}$, $\dot{p}$, $\dot{q}$, $\dot{r}$ should be zero simultaneously. The aircraft model has been trimmed by a SIMPLEX optimisation algorithm after placing different flight path constraints. The *cost function* has been formed by the sum of the squares of the above mentioned derivatives. Appendix-I lists some of the trimmed flight conditions used as initial conditions during simulation.

IV. LINEARISATION

Virtually in all control system analysis and design techniques linear time invariant (LTI) state equations are required. The limitations of algebraic linearisation are discussed in [11]. In this study, MiG-21 mathematical model has been linearised numerically. The numerical algorithm used for linearisation, perturbs state, control and state derivative variables from the steady state condition and evaluates partial derivatives by using adaptive step size.Thus we get LTI state equations in the form :

$$\dot{X} = J1X + J2U + J3\dot{X},\ X \in R^n,\ U \in R^m.$$

The above equation can be further expressed as $\dot{X} = AX + BU$. This is our linear model. MiG-21 nonlinear model has been linearised in wings level flight condition and the same is used for simulation.

V. MODAL DECOMPOSITION

Similarity transformations and eigen analysis provide insight into LTI state equations. $\dot{X} = AX + BU$ can be transformed to $\dot{Z} = (M^{-1}AM)Z + (M^{-1}B)U$ under the nonsingular transformation $X = MZ$. The homogenous solution of this equation contains natural modes of the system. These modes are the characteristic of a particular system. A necessary and sufficient condition for a linear system to be stable is that the real parts of eigenvalues be all negative. In our linear model some states couple back very weakly in level flight condition. The vehicle dynamics is not significantly affected by displacement states (x_e, y_e, z_e), vehicle angular orientation states (ϕ, θ, ψ) and yaw state ψ_v; hence they can be neglected in the analysis. Further in level flight there is clear cut decoupling of lateral and longitudinal dynamics. MIL-F-8785 provides analytical specifications; which are used for analysis of MiG-21 (BIS) modal decomposition. Specifications for class IV aeroplanes (high maneuverability planes) and category B flight phases (non-terminal flight phases normally accomplished using gradual maneuvers without precision tracking) are used for comparison. The results are placed at appendix-II. MiG-21 possess "Level-1" flying qualities during level flying. The same approach can be used for various other flight phases for this aircraft.

VI. TRANSFER FUNCTION ANALYSIS

Transfer function matrix G(s) can be expressed as

$$G(s) = \frac{C\ \mathrm{adj}(\ sI - A\ )\ B + D\ |\ sI - A\ |}{|\ sI - A\ |}$$

The transfer function from jth input to ith output is the ijth element of G(s) and is denoted as $g_{ij}(s)$.

$$g_{ij}(s) = \frac{c_i\ \mathrm{adj}(\ sI - A\ )\ b_j + d_{ij}\ |\ sI - A\ |}{|\ sI - A\ |}$$

The transfer functions developed for wings level flight condition are as follows:

a) thrust to true air speed:

$$\frac{0.0011\ s\ (\ s + 3.9645 \pm 6.6163i\ )}{(\ s + 0.0066 \pm 0.0592i\ )(\ s + 3.9645 \pm 6.6163i)}$$

b) stabiliser to pitch rate:

$$\frac{-3.6691\, s\,(s+0.0106)(s+0.5936)}{(s+0.0066 \pm 0.0592i)(s+3.9645 \pm 6.6163i)}$$

c) rudder to yaw rate:

$$\frac{0.5571\,(s-28.2203)(s+0.8784 \pm 0.4037i)}{(s+22.1294)(s+3.3407 \pm 9.1755i)(s+0.0460)}$$

d) aileron to yaw rate:

$$\frac{6.9009\,(s+2.4912)(s-2.4093)(s+0.5026)}{(s+22.1294)(s+3.3407 \pm 9.1755i)(s+0.0460)}$$

e) rudder to roll rate:

$$\frac{-0.5571\,(s-0.0032)(s+5.2510 \pm 24.2058i)}{(s+22.1294)(s+3.3407 \pm 9.1755i)(s+0.0460)}$$

f) aileron to roll rate:

$$\frac{-6.9009\,(s-0.0033)(s+2.8787 \pm 7.8337i)}{(s+22.1294)(s+3.3407 \pm 9.1755i)(s+0.0460)}$$

In thrust to true air speed transfer function short period poles get cancelled with pair of zeros and transfer function is $\frac{0.0011s}{(s+0.0066 \pm 0.0592i)}$. There is one zero at origin and poles are also quite close to origin. The relative degree is unity, so throttle inputs are initially integrated. In general when the throttle is opened, the extra power input may produce an increase in speed and/or a gain in altitude, and the phugoid mode is associated with the subsequent interchange of potential energy and kinetic energy. This is confirmed by simulation. The stabiliser to pitch rate transfer function has a dc gain of zero (because of the zero at the origin), indicating that a constant stabiliser deflection will not sustain a steady pitch rate.

All the transfer functions of lateral dynamics have r.h.s. zeros. Hence, initial response and final response are in opposite directions. Non minimal plane (NMP) transfer functions do have undesirable effect from human operator point of view. The simulation of response to aileron or rudder pulse confirms the above observations.

VII. SIMULATION EXERCISES

A mathematical model which has dynamic structure in addition to static structure, can be used in a simulation study to generate trajectories of some or all of its variables. This trajectory behaviour can be processed as analysis of or display of the behaviour;ref [9]. The nonlinear model has been used for the simulation of the vehicle response in different flight conditions when controls are kept fixed and for the simulation of vehicle response to control actuation while flying level. The response of the vehicle in level flight is also studied by simulation using the linear model of the aircraft. Simulation run time, output interval for trajectory generation, control pulse duration and magnitude are given in table 1 and table 2. For studying response to control actuation, symmetric doublet pulse (symmetric about the trim setting) has been applied. The trajectories of all the variables are generated. As a visual aid, symbology for "Headig Indicator" and "Gyro Horizon" has been generated. The trajectories of some of the variables have been processed as display and the rest are analysed by plotting the graphs. The simulation programmes are developed in general purpose computer language "C".

Table 1 : Simulation of steady state flight conditions

Flight Condition	Run Time (sec)	Output Interval (sec)
a. Pull-up	15	0.1
b. Wings Level	600	1.00
c. Climb	60	1.00
d. Turn	60	1.00
e. Roll	10	0.1

Table 2 : Simulation of response to control actuation

Control	Simulation Run Time (sec)	Pulse Duration (sec)	Pulse Magnitude
1. Stabiliser	10	1	$\pm 2^{o}$
2. Rudder	15	3	$\pm 2^{o}$
3. Aileron	7	1	$\pm 2^{o}$
4. Throttle	15	3	setting $\equiv$ 3 1

VIII. MODEL VALIDATION AND SIMULATION ANALYSIS

For model validation we have relied on available flight test data. Flight test data suggests, to get maximum range at an altitude of 5000m, true air speed of the aircraft should be 225 m/sec. We are able to trim our aircraft model at the same speed for level flight. The stabiliser deflection for level flight, according to flight test data is -1.65^{o}. The trim setting of stabiliser deflection for aircraft model is -1.51^{o}. In pull-up flight, flight path angle (γ) changes with θ while α remains constant. It is obvious that the constant stabiliser deflection is not able to sustain a steady pitch rate because of NMP zero in stabiliser to pitch rate transfer function. In level flight, phugoid mode is clearly visible. θ and γ change in phase while α remains constant. V and θ have a phase difference of approx. 90^{o}. V, first decreases and then increases as the aircraft climbs the altitude with the thrust assumed constant. Due to nonlinearity in model γ does not settle to zero; but at slightly higher value . The period of phugoid oscillation is 90 sec. for simulation with nonlinear model and it is 104 sec for linear model. This was expected as in our linear model altitude state has not been included. The simulation response with linear model closely agrees with the modal decomposition

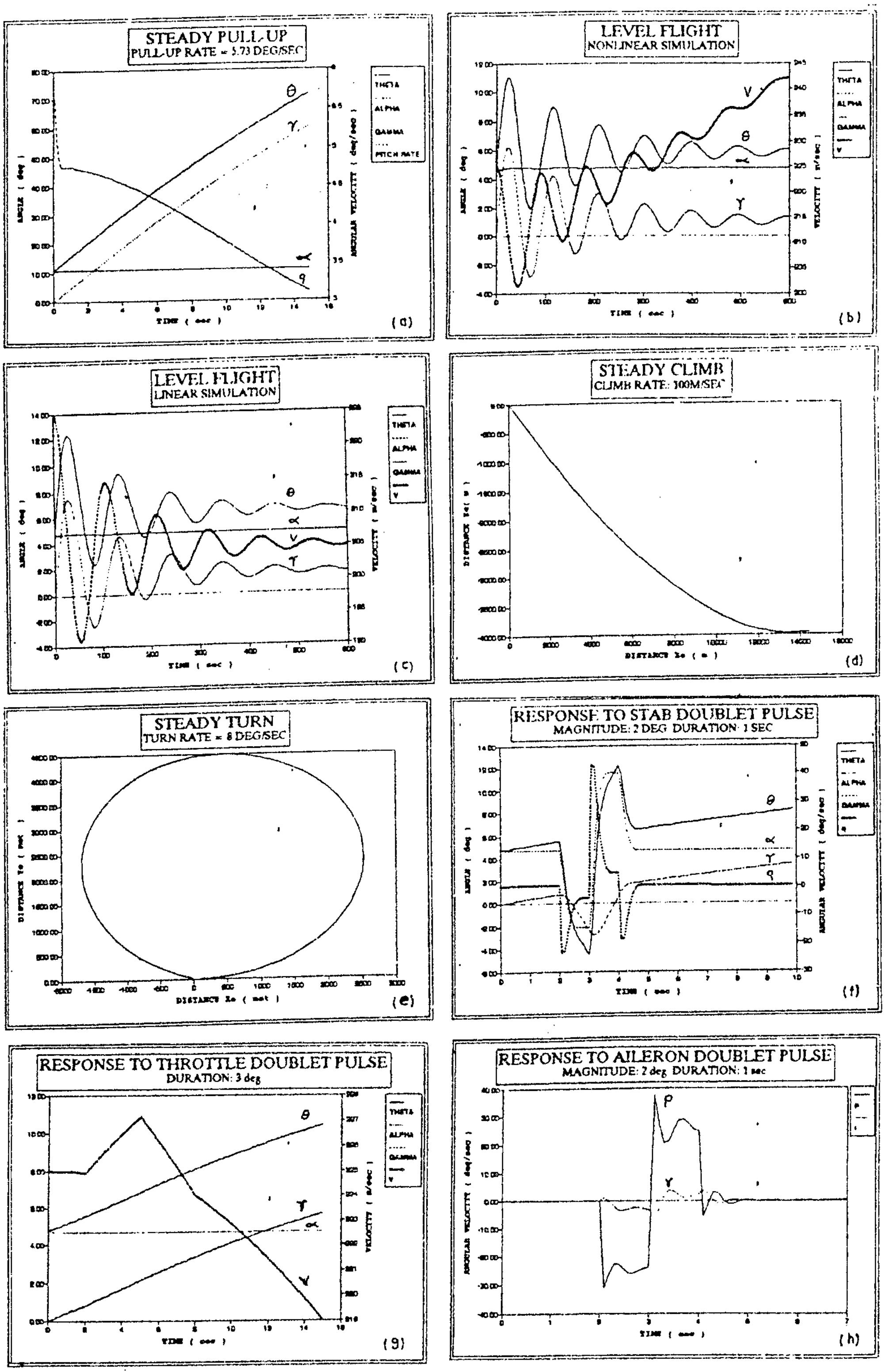

Fig. 3. Simulation graphs.

results. During steady climb the aircraft climbs 3976m in 13500m during a period of 57 sec. Flight test data of the aircraft gives the climb of 4000m (5000m to 9000m) in 20km (15km to 35km) during 50 sec, while climbing to ceiling altitude at full throttle. This, closely matches with our simulation experiment. For steady turn, simulation gives X_{radius} = 2500m and Y_{radius} = 2250m and time to full turn is 60.5 sec. Flight performance of the aircraft gives turn radius = 1700m at 0.7 mach with the aircraft weight = 7500kg at g load equal to 3.5 g. Time required for full turn is 75 sec.

When the stabiliser pulse is applied, short period mode is getting excited. θ and α vary while α slightly ahead of θ in phase. The lateral dynamics remains unchanged. When the throttle setting is changed about the trim setting, Phugoid mode is superimposed on level flight behaviour.

When the rudder pulse (positive) is applied at time = 2 sec, p overshoots in negative direction up to 2.3 sec and then starts increasing in positive direction. This can be explained by rudder to role rate transfer function which has got one zero very near to the origin in r.h. plane. Similarly r overshoots and then undershoots due to a zero in rudder to yaw rate transfer function, which is far inside in r.h . plane. The simulation reveals clearly the effect of competing physical mechanisms. Similarly response to aileron pulse can be analysed. In both the cases, roll angle and yaw angle do not settle back to initial value before the disturbance, but negative heading change with positive bank angle change remains. This clearly brings out the undesirable effect of NMP transfer functions from human operator point of view.

IX. CONCLUSION

Simulation results are in close agreement with the flight test data. Simulation exercises with modal decomposition analysis and transfer function analysis give sufficient insight into MiG-21 (BIS) rigid body dynamics. This work would form an useful input for formulating new control strategy during any avionics/armament update programme on MiG-21 (BIS) aircraft.

APPENDIX - I: TRIMMED FLIGHT CONDITIONS FOR MiG-21 (BIS)

	PULL-UP	LEVEL	CLIMB	TURN	ROLL
CONSTRAINT --->	$\dot{\theta}$ = 0.1 rad/sec		$\dot{z}_e$ = 100 m/sec	$\dot{\psi}$ = 0.14 rad/sec	$\dot{\phi}$ = 0.6 rad/sec
V (m/sec)	225	225	225	225	225
α (rad)	0.188045	0.083579	0.061824	0.12860	-0.00998
β (rad)	0.000126	0.00273	-0.001614	0.006525	0.007384
ϕ (rad)	0	0	0	1.3089	0
θ (rad)	0.188045	0.083579	0.61082	0.13089	0
ψ (rad)	0	0	0	0	0
p (rad/sec)	0	0	0	-0.07	0.6
q (rad/sec)	0.1	0	0	0.115	0
r (rad/sec)	0	0	0	0.1	0
Net Thrust (kg)	4520.077	1379.669	4780.340	2032.607	869.208
stab (deg)	-3.362715	-1.509934	-1.189398	-2.136359	-0.14218
ail (deg)	-0.014271	-0.205160	0.110744	-0.164135	-2.08851
rud (deg)	-0.06726	0.286742	-0.173916	-0.408196	0.03293

Note: a) For level flight $\dot{\phi} = \dot{\theta} = \dot{\psi} = 0$.
b) Altitude = 5000m., for all trim conditions.

APPENDIX - II : MODAL DECOMPOSITION

1. LONGITUDINAL DYNAMICS

(i) Decoupled longitudinal matrix A

$$\begin{array}{cccc} V & \alpha & \theta_W & q \end{array}$$

$$A = \begin{bmatrix} -0.0141 & -15.677 & -9.6635 & 000000; \\ -0.0004 & -0.6290 & 0000000 & 1.0000; \\ 0.0004 & 0.6290 & 0000000 & 000000; \\ -0.0002 & -54.901 & 0000000 & -7.300 \end{bmatrix}$$

(ii) Eigen Values

$-3.9649 \pm 6.6161i$ $\quad\quad$ $-0.0066 \pm 0.0592i$

(iii) Modes

The four states give rise to two complex-conjugate pairs of eigenvalues; which correspond to two stable oscillatory modes.

(a) Short Period Mode ($\lambda = -3.9649 \pm 6.6161i$)

$\omega_d = 6.6161$ rad/sec $\quad$ $T_d = 0.94968$ sec

$\zeta\omega_n = -3.9649$ rad/sec $\quad$ $\zeta = 0.5140$ $\quad$ $\omega_n = 7.7129$ rad/sec

(b) Phugoid Mode ($\lambda = -0.0066 + 0.0592i$)

$\omega_d = 0.0592$ rad/sec $\quad$ $T_d = 106.135$ sec

$\zeta\omega_n = -0.066$ rad/sec $\quad$ $\zeta = 0.1108$ $\omega_n = 0.05956$ rad/sec

(iv) Eigen Vectors

	Short Period	Phugoid
V	$0.1333 \pm 0.2322i$	$1.0000 \pm 0.0000i$
α	$-0.0607 \pm 0.1205i$	$-0.0001 \pm 0.0000i$
θ_W	$-0.0059 \pm 0.0093i$	$-0.0007 \pm 0.0061i$
q	$1.0000 \pm 0.0000i$	$0.0004 \pm 0.0000i$

2. LATERAL DYNAMICS

(i) Decoupled Lateral Matrix A

$$\begin{array}{cccc} \beta & \phi_W & p & r \end{array}$$

$$A = \begin{bmatrix} 0.1065 & 0.0429 & 0.0835 & -0.9955; \\ 0.0082 & -0.0004 & -0.9965 & 0.0835; \\ -380.43 & 0.0000 & -23.022 & -1.9430; \\ 79.794 & 0.0000 & -0.2306 & -5.9410 \end{bmatrix}$$

(ii) Eigen Values

$-3.4165 \pm 9.1390i$ $\quad -22.0760$ $\quad -0.0460$

(iii) Modes

The four states give rise to one complex-conjugate pair of eigenvalues and two real eigenvalues.

(a) Dutch Roll Mode ($\lambda = -3.4165 \pm 9.1390i$)

$\omega_d = 9.139$ rad/sec $\quad T_d = 0.6875$ sec

$\zeta\omega_n = -3.4165$ rad/sec $\quad \zeta = 0.35$ $\quad \omega_n = 9.750$ rad/sec

(b) Roll Subsidence Mode ($\lambda = -22.0760$):

$\tau = 0.00453$ sec

(c) Spiral Mode ($\lambda = -0.0460$)

$\tau = 21.739$ sec

(iv) Eigen Vectors

Dutch Roll	Roll Subsidence	Spiral	
-0.0499 ± 0.0259i	-0.0026	0.0028	β
0.0397 ± 0.0972i	0.0450	1.0000	ϕ_W
1.0000 ± 0.0000i	1.0000	-0.0493	p
-0.3284 ± 9.1390i	0.0273	0.0391	r

3. MIL vis-a-vis MiG-21

MODE	MIL SPECIFICATION	MiG-21(BIS)
1.Phugoid	$\xi_p \geq 0.04$ (Level 1)	0.1108
2.Short Period	$0.30 < \xi_{sp} \leq 2.0$ (Level 1)	0.5140
3.Dutch Roll	$\xi_d \geq 0.08$	0.3420
	$\omega_{nd} \geq 0.4$ rad/sec	9.764 rad/sec
	$\xi_d\ \omega_{nd} \geq 0.15$ rad/sec (Level 1)	3.339 rad/sec
4.Roll Subsidence	$\tau < 1.4$ sec (Level 1)	0.0453 sec
5 Spiral	Minimum doubling time 20 sec (Level 1)	τ =21.739 sec

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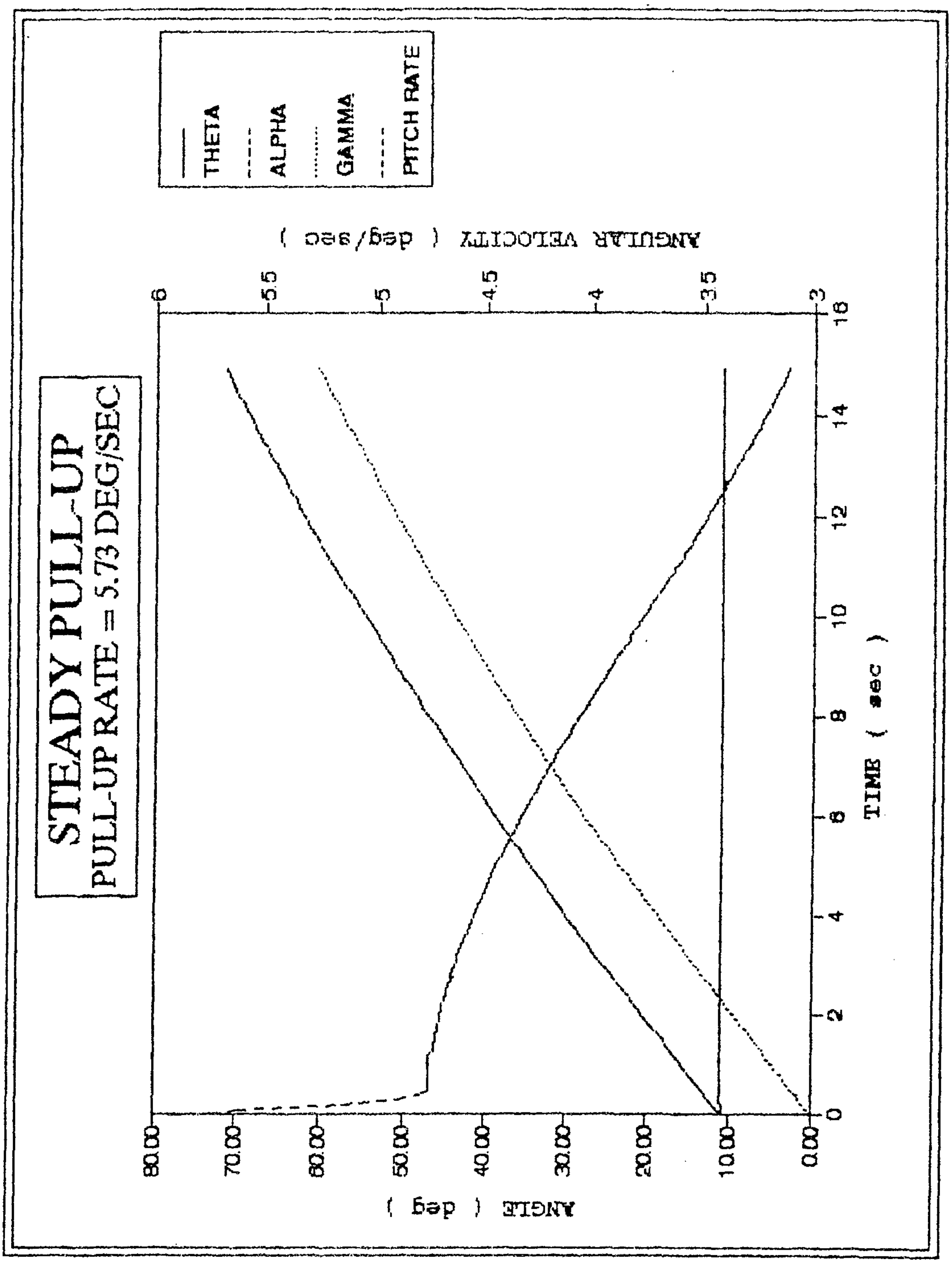
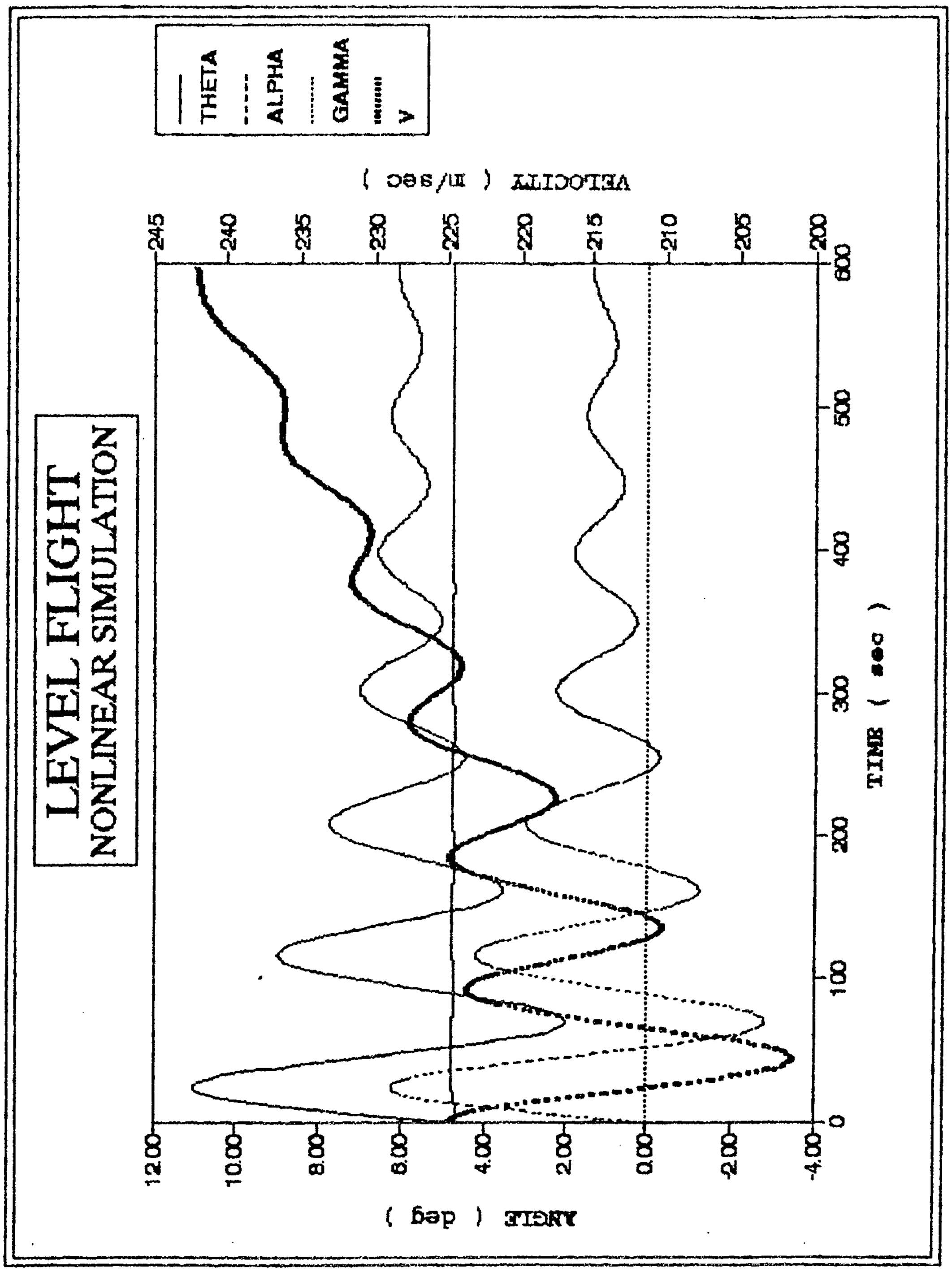
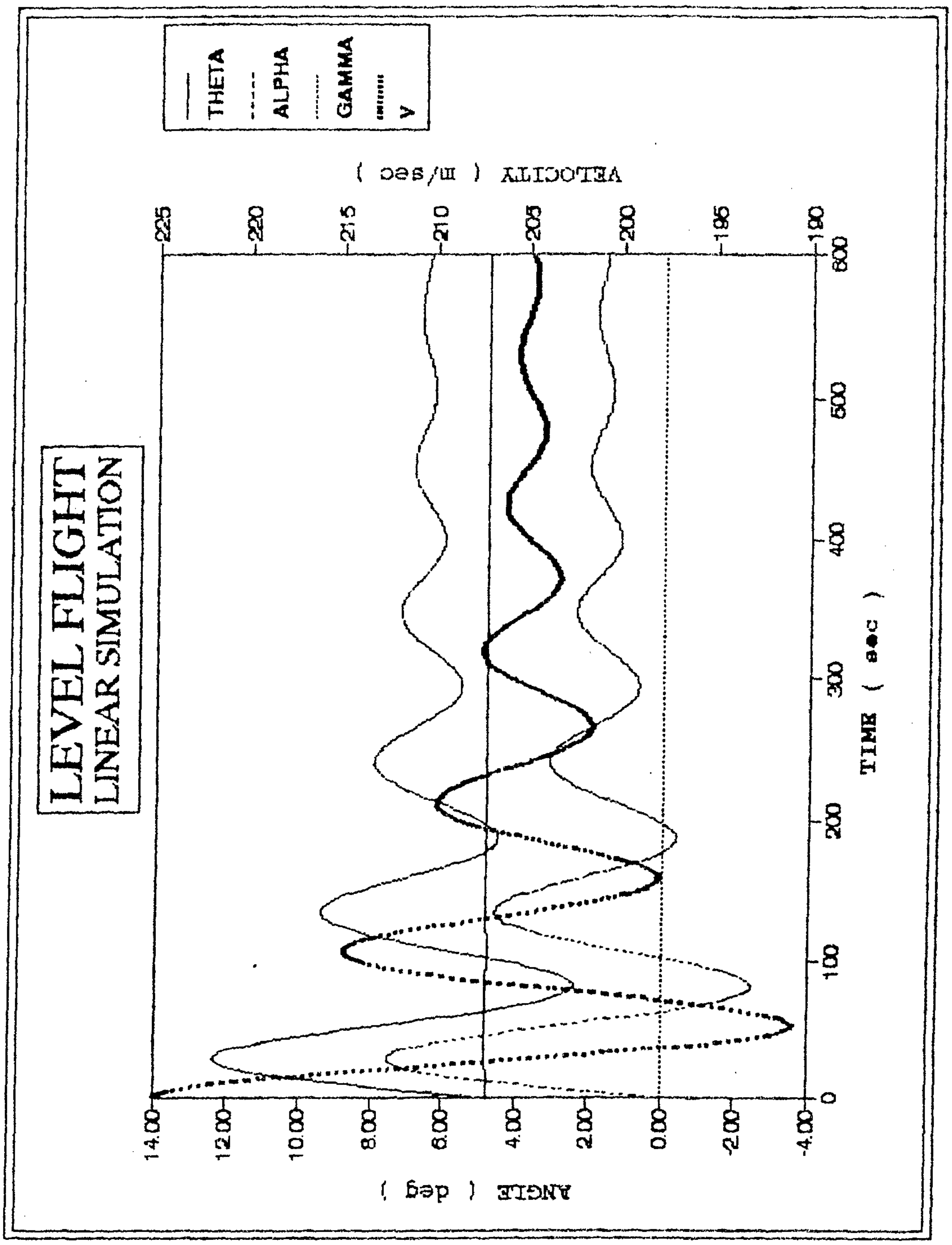
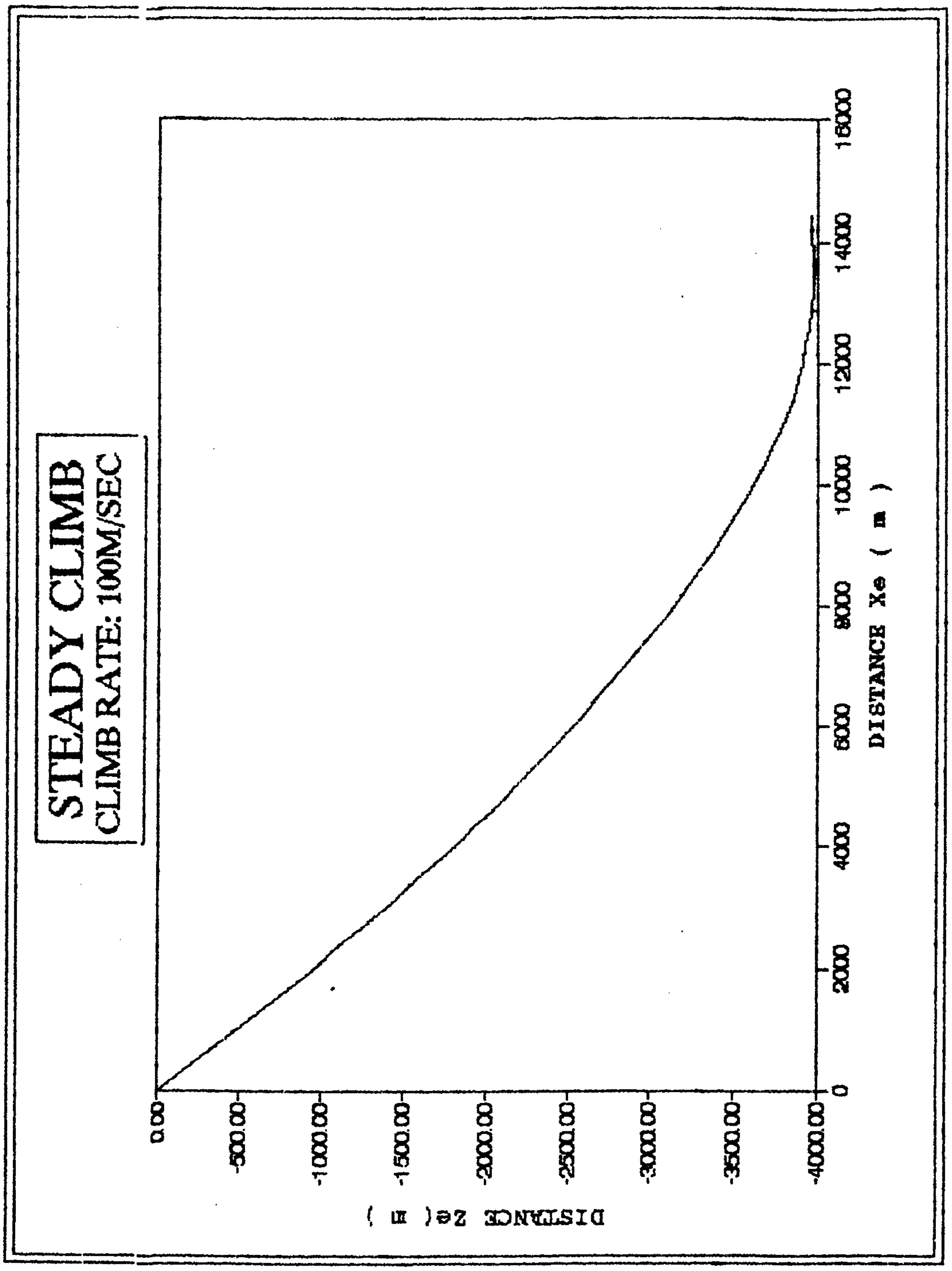
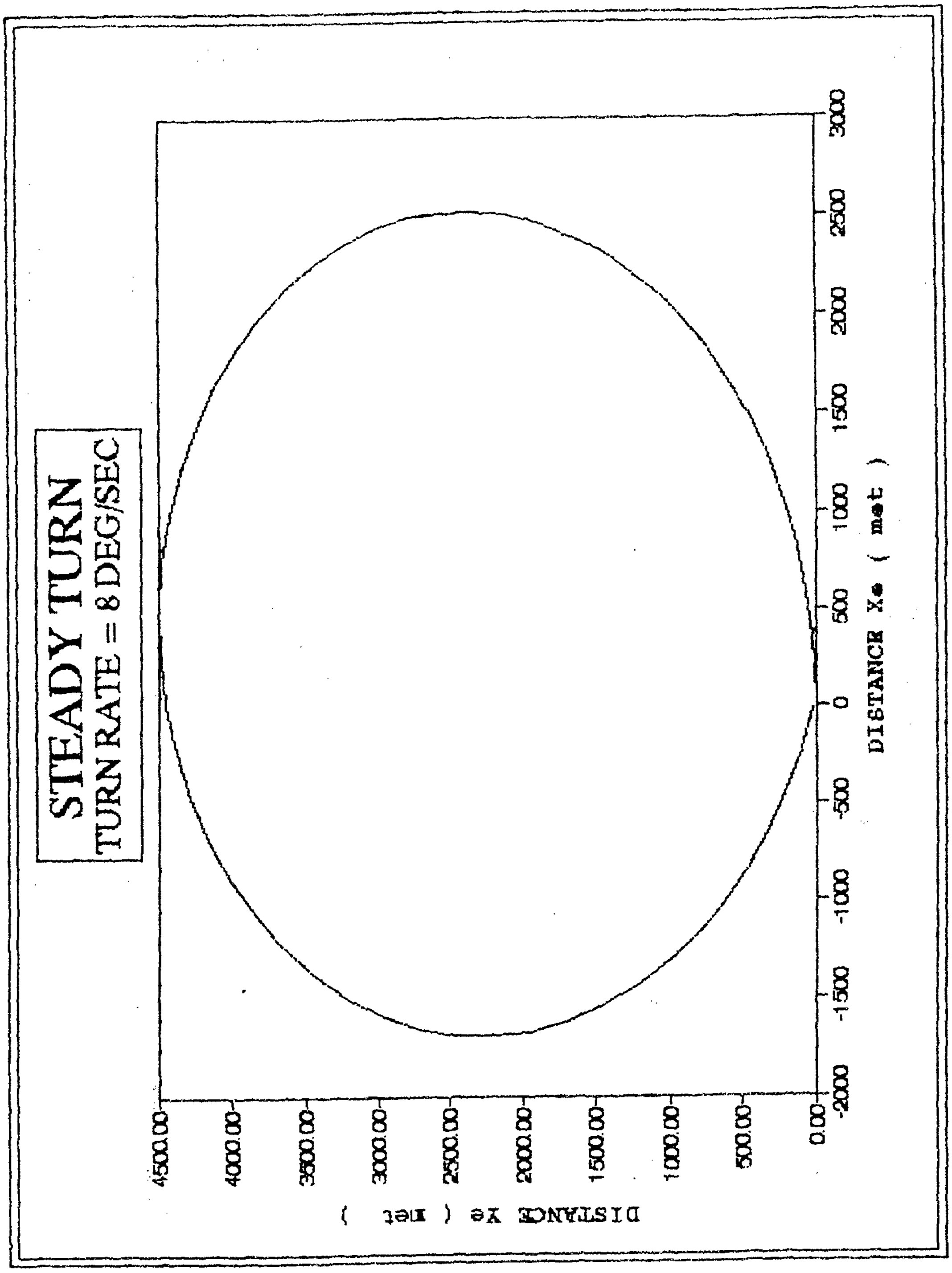
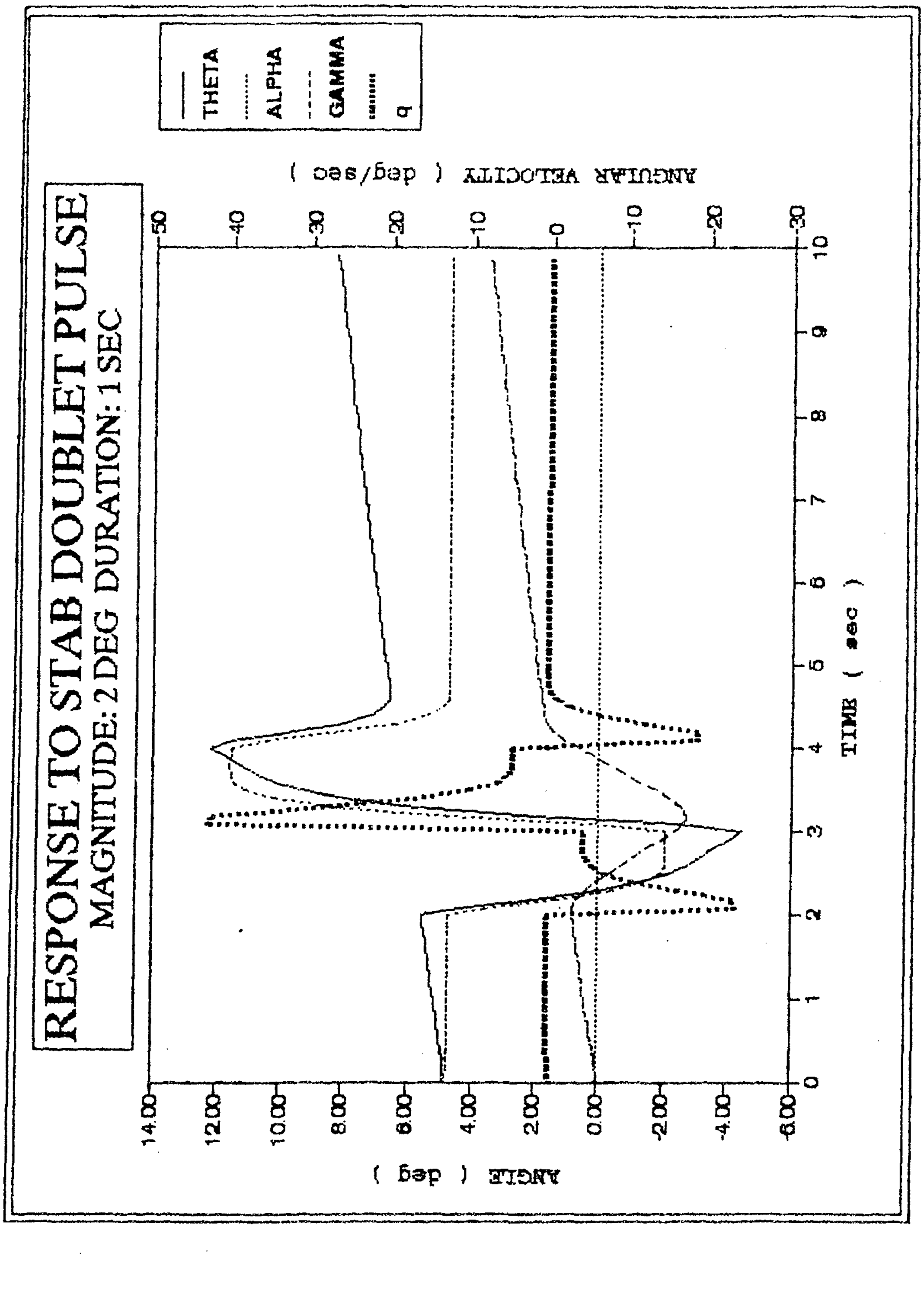

STEADY PULL-UP
PULL-UP RATE = 5.73 DEG/SEC
THETA
ALPHA
GAMMA
PITCH RATE
ANGLE (deg)
80.00
70.00
60.00
50.00
40.00
30.00
20.00
10.00
0.00
ANGULAR VELOCITY (deg/sec)
6
5.5
5
4.5
4
3.5
3
TIME (sec)
0
2
4
6
8
10
12
14
16

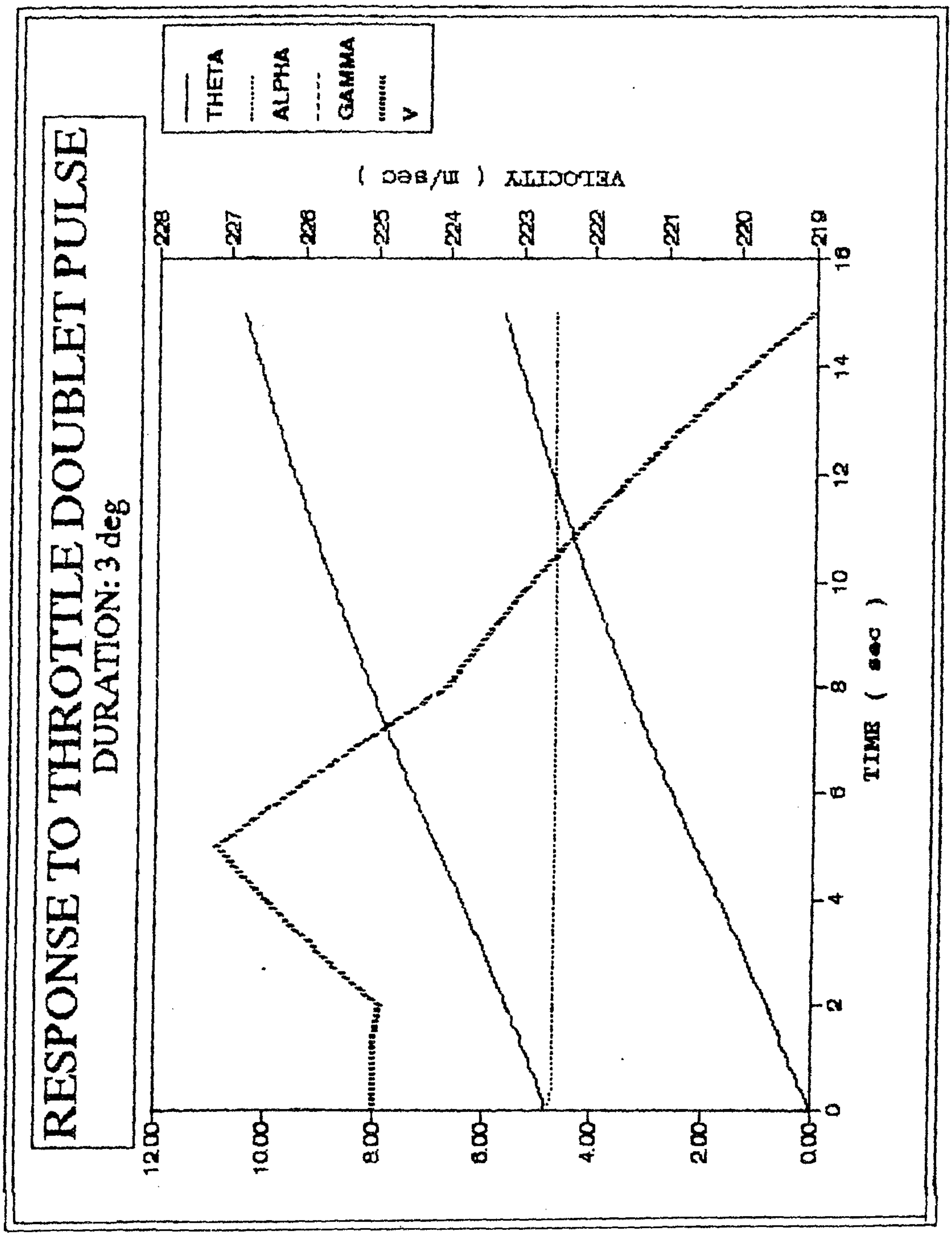

LEVEL FLIGHT
NONLINEAR SIMULATION
THETA
ALPHA
GAMMA
V
ANGLE (deg)
VELOCITY (m/sec)
TIME (sec)
12.00
10.00
8.00
6.00
4.00
2.00
0.00
-2.00
-4.00
245
240
235
230
225
220
215
210
205
200
0
100
200
300
400
500
600

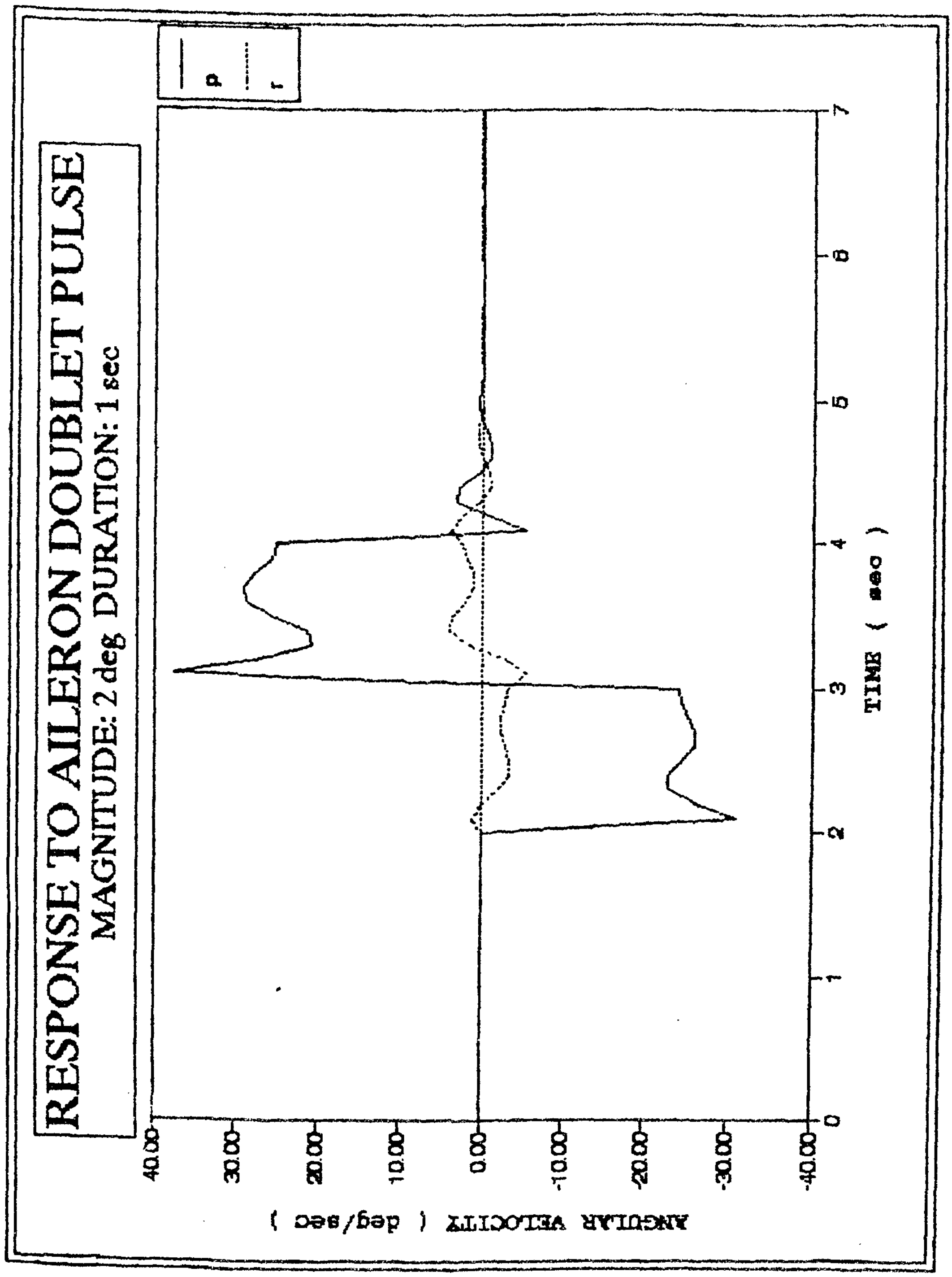

LEVEL FLIGHT
LINEAR SIMULATION
THETA
ALPHA
GAMMA
V
ANGLE (deg)
VELOCITY (m/sec)
TIME (sec)
14.00
12.00
10.00
8.00
6.00
4.00
2.00
0.00
-2.00
-4.00
225
220
215
210
205
200
195
190
0
100
200
300
400
500
600


STEADY CLIMB
CLIMB RATE: 100M/SEC
DISTANCE Xe (m)
0
2000
4000
6000
8000
10000
12000
14000
16000
DISTANCE Ze(m)
0.00
-500.00
-1000.00
-1500.00
-2000.00
-2500.00
-3000.00
-3500.00
-4000.00


STEADY TURN
TURN RATE = 8 DEG/SEC
DISTANCE Xe (met)
DISTANCE Ye (met)
-2000
-1500
-1000
-500
0
500
1000
1500
2000
2500
3000
0.00
500.00
1000.00
1500.00
2000.00
2500.00
3000.00
3500.00
4000.00
4500.00


RESPONSE TO STAB DOUBLET PULSE
MAGNITUDE: 2 DEG DURATION: 1 SEC
THETA
ALPHA
GAMMA
q
ANGLE (deg)
ANGULAR VELOCITY (deg/sec)
TIME (sec)
14.00
12.00
10.00
8.00
6.00
4.00
2.00
0.00
-2.00
-4.00
-6.00
50
40
30
20
10
0
-10
-20
-30
0
1
2
3
4
5
6
7
8
9
10


RESPONSE TO THROTTLE DOUBLET PULSE
DURATION: 3 deg
THETA
ALPHA
GAMMA
V
VELOCITY (m/sec)
228
227
226
225
224
223
222
221
220
219
12.00
10.00
8.00
6.00
4.00
2.00
0.00
0
2
4
6
8
10
12
14
16
TIME (sec)


RESPONSE TO AILERON DOUBLET PULSE
MAGNITUDE: 2 deg DURATION: 1 sec
p
r
40.00
30.00
20.00
10.00
0.00
-10.00
-20.00
-30.00
-40.00
ANGULAR VELOCITY (deg/sec)
0
1
2
3
4
5
6
7
TIME (sec)

THE DEVELOPMENT OF SYSTEM APPROACH TO THE REQUIREMENTS TO THE HANDLING QUALITIES AND PREDICTION OF PILOT-INDUCED-OSCILLATION (PIO) TENDENCY

A. V.Efremov
MAI, Moscow, Russia

The creation of highly augmented aircraft leads to appearance of specific dynamic peculiarities arisen in pilot-vehicle closed-loop system. This circumstances defined the necessity in investigations in development of handling quality criteria combined the parameters of pilot workload and task performance. Analysis demonstrates that such criteria needs in the further modification. Except there is necessary to develop the criteria for the prediction of PIO tendency. This phenomena is a typical for the modern highly augmented aircraft. In [1] there were carried out the investigation included:

— analysis of the variables influenced on the dynamics features of pilot-vehicle system and PIO;
— experimental investigations of the different dynamic configurations with goal to define the parameters for prediction of the handling qualities and PIO tendency;
— development of technique for prediction of PIO tendency by mathematical modeling.

In this paper it is considered the influence of part of variables investigated in [1] and particularly:

1. The controlled element dynamics defined by the piloting task and side effects typical for highly augmented aircraft.
   In particular there were considered the following side effects:
   — the phase delay in flight control system, which can be approximated by the time delay element $e^{-s\tau}$, where τ can reach up to 0.2—0.3 sec;
   — the limitation on maximum control surface rate $\dot{\delta}_{max}$.

In the case when $\dot{\delta} > \dot{\delta}_{max}$ the considerable degradation in dynamics is noticeable (decrease of damping ratio, increase of phase delay and run-to-run variability of controlled element frequency response characteristics). As for piloting tasks there are considered below two such tasks: pitch angle (Θ) and angle of sight $(\varepsilon = \Theta + H/L)$ control tracking tasks.

2. Factors corresponding to the real flight. There are considered the influence of additional task, acceleration arisen in aircraft motion, and interval of permissible error (±d) which can be different for the different piloting tasks. Analysis of the influence of variables was carried out by usage the system approach to the manual control tasks [1] based on careful investigation of pilot and pilot-vehicle control and psycho physiological response characteristics.

Results of experiments

Experimental investigations demonstrated that increase of phase delay increases the resonance peak (r) in closed-loop system. In depend on the controlled element dynamics the frequency of the resonance peak takes place at crossover frequency ω_c (for case of small value of short period frequency) or at frequency corresponding to the amplitude margin $\omega_{\varphi=180^\circ}$. In these cases a pilot induces a considerable lead or complex lag-lead (fig. 1) adaptation to compensate the controlled element dynamics. These compensations cannot be described by well-known crossover model [2], In any case the pilot-ratings (PR) increases when time delay τ increases.

The decrease of $\dot{\delta}_{max}$ leads to decrease of crossover frequency, appearance of droop and disappearance of resonance peak in closed loop system. In that case parameters of closed loop frequency response characteristics don't give any information about PIO tendency what leads to the search of other parameters characterized this tendency. In experiments it was discovered the resonance peak of spectral density of the signal (e_n) defined by remnant $(S_{n_e n_e})$. The value of this resonance peak is higher than level of spectral density of signal defined by input signal $(S_{e_i e_i})$. In that connection it was offered the criteria $\rho = S_{n_e n_e} / S_{e_i e_i}$. In the case when $\rho > 1$, the system has the periodically unstable oscillations.

In comparison with pitch tracking task the angle of sight control tracking task is accompanied by increase of resonance peak up to 2—6 dB, pilot lead compensation and pilot opinion rating up to 2.5 units, appearance of additional low frequency resonance peak, connected with path oscillatory motion. Thus the aircraft evaluated as satisfactory in one piloting task can be evaluated as unsatisfactory in other more complex piloting task. This result allowed to conclude that in pitch tracking task the pilot ratings has to be less then PR= 3.5 (for example PR=1—1.5). In that case the angle of sight tracking task will be not accompanied by considerable deterioration of ratings. The necessary pilot rating in each piloting task can be reached only by corresponding change the controlled element dynamics supplied the best pilot rating in each piloting task. Such approach is considered in [2].

The increase of control channel leads to considerable increase of resonance peak in close-loop system (in 2—3 times) and pilot ratings (up to 2 units). These results leads to the question about the possibility to use the criteria received from consideration of single loop system. In case of the positive solution there is obvious that the requirements received in the single loop system has to correspond to PR < 3.5. Investigation demonstrated that one of the major variable is a permissible range (level) of error (± d) where pilot tries to keep the error signal. For example, for configuration 2.10 investigated in [4] the decrease of "d" from 2.0 sm to 0.5 sm leads to increase of the resonance peak (from 2.1 dB to 8.15 dB) maximum pilot lead compensation $\varphi_{p_{max}}$ (from 0° to 45°) and pilot rating (from 3.5 up to 8.5). Thus the same aircraft can be evaluated by the different way under the different values of "d". Decrease of this variable can lead to appearance of PIO tendency for aircraft not having this phenomena.

These and other results allowed to define the shortcomings of well-known Neal-Smith criteria [3], which doesn't explain this effect, and to develop the criteria allowed to take into consideration a wide range of variables. For these purpose it was carried out a wide range of experimental researches. It was investigated 48 configurations considered in [3,4]. The measurement of pilot and pilot-vehicle system frequency response characteristics and pilot ratings allowed to get the ranges of equal ratings in coordinates r, $\Delta\varphi_p$.

The more difficult thing in process of definition of ranges was the determination of pilot workload parameter $\Delta\varphi_p$ because of the pilot phase had considerable positive or negative values in frequency interval $\omega_c < \omega < \omega_{\varphi=180°}$ for a set of configurations. In that connection it was offered to consider the difference $\Delta\varphi_p = \varphi_p - \varphi_p^e$, where φ_p is a pilot phase for the investigated dynamics W_c , and φ_p^e — is a pilot phase for dynamics W_C^{opt} [2] corresponding to the simplest type of pilot behavior in all frequency ranges. The knowledge of $\Delta\varphi_p(\omega)$ is necessary for definition of positive and negative values $\Delta\varphi_p^+$ and $\Delta\varphi_p^-$.

As a pilot workload parameter $W=(\Delta\varphi_p^+, \Delta\varphi_p^-)$ it has to be chosen the value corresponding to the worse range of ratings. This rule gives the accordance of parameters r and $\Delta\varphi_p$ with PR for all investigated configurations (fig. 2). The measurement of ratings on the PIOR and Cooper-Harper scales and their comparison (fig. 3) allowed to get the ranges of equal PIOR ratings in coordinates r, W (fig. 4) which can be used as a criteria for prediction of PIO tendency.

In [1] it was offered to use the optimal pilot model for calculation of parameters r and W. For this purpose it was carried out the mathematical modeling for the majority of configurations investigated experimentally. The results are shown on fig. 5. There is seen that except two configurations it was received good accordance with experimental results. Such good accordance was supplied by inclusion of variance of stick deflection in cost function and by the development of special procedure for the choice of coefficients from condition of accordance of measured and calculated integral and frequency response characteristics.

References

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4. R.E.Smith. Effects of control system dynamics on fighter approach and landing longitudinal flying qualities v.1 AFFDL-TR—78.

The MAI Pilot Workload Parameter

$$W = \{ \Delta\varphi_p^- \ , \ \Delta\varphi_p^+ \}$$

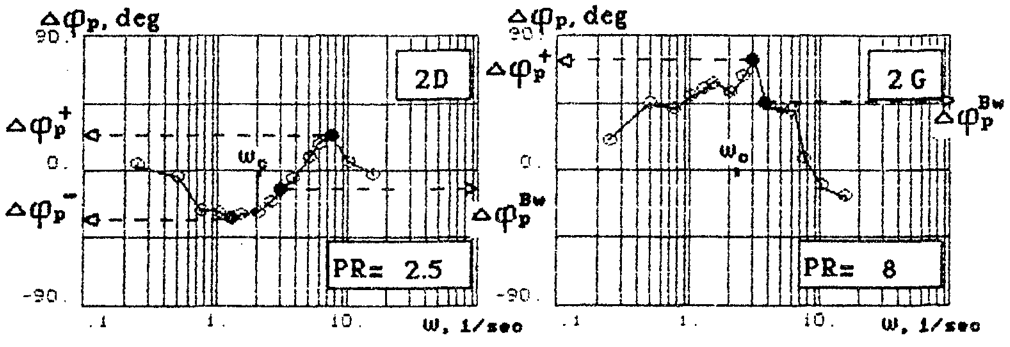


fig. 1

Good correlation of pilot ratings with closed-loop parameters

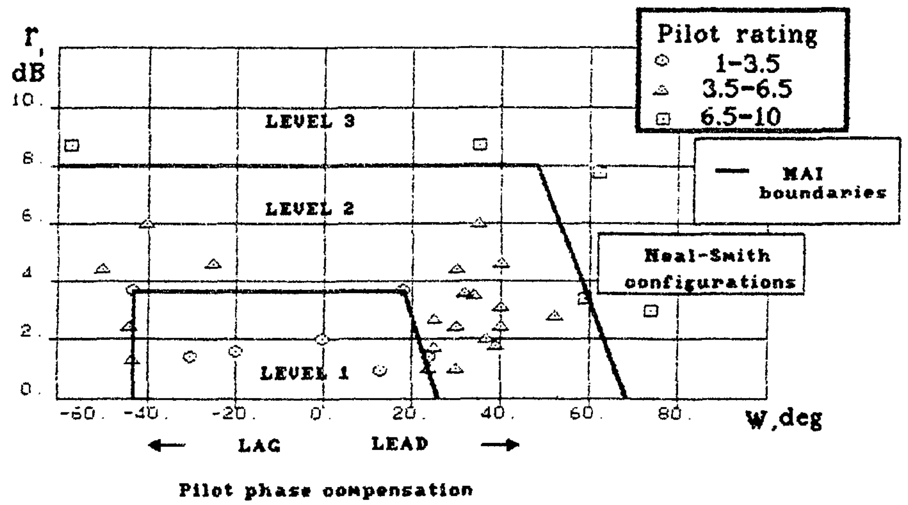


fig. 2

Correlation of PIO ratings with pilot ratings

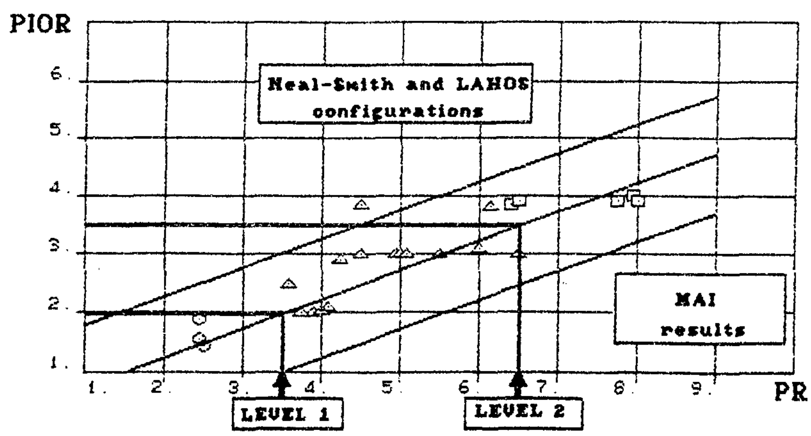
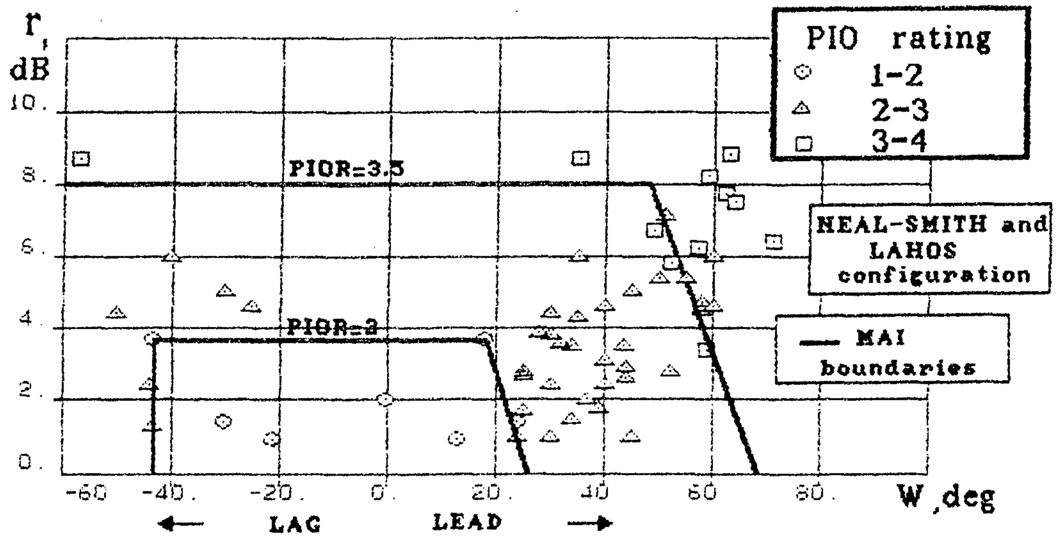


fig. 3

Correlation of PIO ratings with closed-loop parameters



Pilot phase compensation

fig. 4

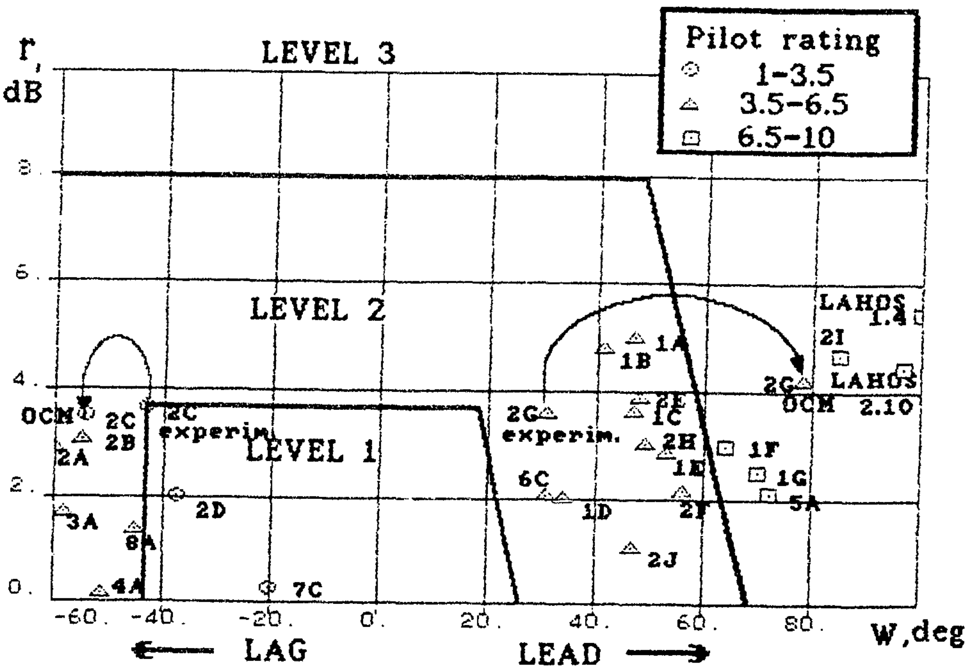


fig. 5

SOME ASPECTS OF DESIGN, CERTIFICATION AND TEST OF ADVANCED AIRPLANE CONTROL AND STABILITY AUGMENTATION SYSTEM

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Soukhoi Aircraft Design Bureau, Moscow, Russia

A.A.Bortsov
Pavlovo Machinery Plant "Voskhod", Pavlovo-on-Oka, Russia

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TsAGI, Zhukovsky, Russia

ABSTRACT

One of the main problems appearing in the advanced airplane design is providing high level of flight safety and control and stability characteristics, especially for the airplane with reduced longitudinal static stability, high efficiency of control surfaces and with electrohydraulic actuators. To achieve these purposes the special design of control system architecture and ground based tests are necessary. Traditional routines of control system refinement can not meet all requirements and evaluate the operation of control system algorithms under the wide range of flight conditions. The great volumes of tests are necessary including the real time hardware tests.

The design concept of the control system and nontraditional experimental techniques are shown in the paper on the base of the advanced supersonic administrative airplane. The experimental control system investigation and refinement is based on the special equipment such as high precision frequency analyzer and other equipment. The certification of control system can be fulfilled successfully as the result of such design and test technique.

INTRODUCTION

The highly augmented flight control system without any doubt is a main onboard system at any modern or advanced airplane. The flight control system provides high level of stability characteristics and flight safety and certification of airplane may be impossible without the certification of flight control system, especially for the airplane with reduced longitudinal static stability margin. So for this class of airplanes the special routines of control system design, ground based tests and experimental investigation and refinement are necessary. The particular feature of research and development process of flight control system is taking into account the certification problem at any stages from the preliminary design till the onboard testing and refinement.

Soukhoi Aircraft Design Bureau (Soukhoi ADB) in cooperation with Honeywell Int., Central Aerohydrodynamic Institute (TsAGI), Pavlovo Machinery Plant "Voskhod" (PMP "Voskhod"), MOOG and some other organizations are developing a project of advanced administrative supersonic airplane S-21 with longitudinal static instability at subsonic speeds (up to 7 — 10 % of MAC) and reduced static stability margin at supersonic speeds (about neutral stability). And the development of S-21 flight control system has demanded attention to all aspects mentioned above and some illustrations of flight control system design and experimental refinement is presented in this paper.

FLIGHT CONTROL SYSTEM ARCHITECTURE

Common view of S-21 airplane is presented in fig. 1. Because of longitudinal instability of airplane the flight control system does not include mechanical control or backup channels in comparison with typical modern transport airplanes and consists of electronic and hydraulic elements only. For this reason and because of the usage a number of electronic command displays in crew cockpit the side stick controllers (left and right hand) are installed. Because of the flight safety requirements in situation of control system failure six independently control surfaces must be mounted at S-21 airplane for longitudinal and roll control: two canard surfaces and four elevons. This configuration provides the controllability of airplane in any control surface fault situation including maximum deflection. The longitudinal efficiency of canard and elevons are approximately equal, so in at worst failure situation longitudinal or roll control efficiency can not be smaller than about 50% of total efficiency. In the rudder control channel conventional solution — one control surface — is used.

An architecture of flight control system (control sticks, flight computers and sensors, redundancy levels and faults' detection principles) was developed by Soukhoi ADB and Honeywell in cooperation and provides high level of reliability, convenience of technical service and possibility of future upgrade. Flight control system block-diagram is presented in fig. 2, 3.

Airplane flight control algorithms were developed by Soukhoi ADB and TsAGI and presented in fig. 4 and 5. These flight control algorithms provide strong limitation of normal load factor and angle of attack (because of longitudinal static instability and absence of airplane longitudinal trim at high angle of attack). In addition to these limitations control system algorithms provide the limitations of maximum pitch angle and bank angle and stabilization of pitch and bank angles, vertical speed, horizontal flight, etc. Lateral-directional control algorithms were developed to provide good coordination in roll with very small side accelerations during roll maneuvers. Another problem to be solved by flight control algorithms is an optimization of airplane trim configuration at all flight regimes. The problem solution is achieved in longitudinal control algorithms by special control signal distribution between canard and elevons, that is a function of flight regime. In failure situations control algorithms must be reconfigured to provide necessary level of controllability.

Control system consists of four independent digital primary flight computers (all flight control tasks are executed in each computer) and analog back-up computer to provide minimal level of controllability and stability of airplane (in the situation of total fault of digital part). Three kinds of processors and software are used in primary flight computer.

The side sticks with controlled loads ("active sticks") developed by Honeywell provide good agreement between pilots and have ergonomic advantages in comparison with side stick without load control, i.e. "passive sticks".

To provide high reliability of information signals needed for control algorithms such as angular rates p, q, r and load factors N_z and N_y (in inertial navigation unit), aerodynamic angles α and β (in air data computer) and control stick deflections all sensors have three or four redundancy levels as presented in fig. 2. Analog back-up computer has its own independent sensors of flight parameters.

CONTROL SYSTEM INVESTIGATION AND REFINEMENT

In addition to conventional test procedures special routines based on the investigation of "airplane — control system" closed loop must be used. The main part of these routines are frequency domain tests of control system and its elements for very small amplitude of input signals. Special requirements on the frequency responses of canard and elevon actuators were developed by Soukhoi ADB, TsAGI and PMP "Voskhod". The typical requirements are shown in fig. 6a for input signal amplitude 0.1% of maximum control signal. The requirements are boundaries at the amplitude — frequency and phase — frequency planes in the airplane rigid body frequency region and guarantee the amplitudes of self induced oscillations with amplitudes no more than 0.02 g for normal load factor N_z and 0.1 degree for pitch angle. Such requirements led to actuator construction shown in fig. 3. The next steps are necessary to satisfy requirements mentioned above:

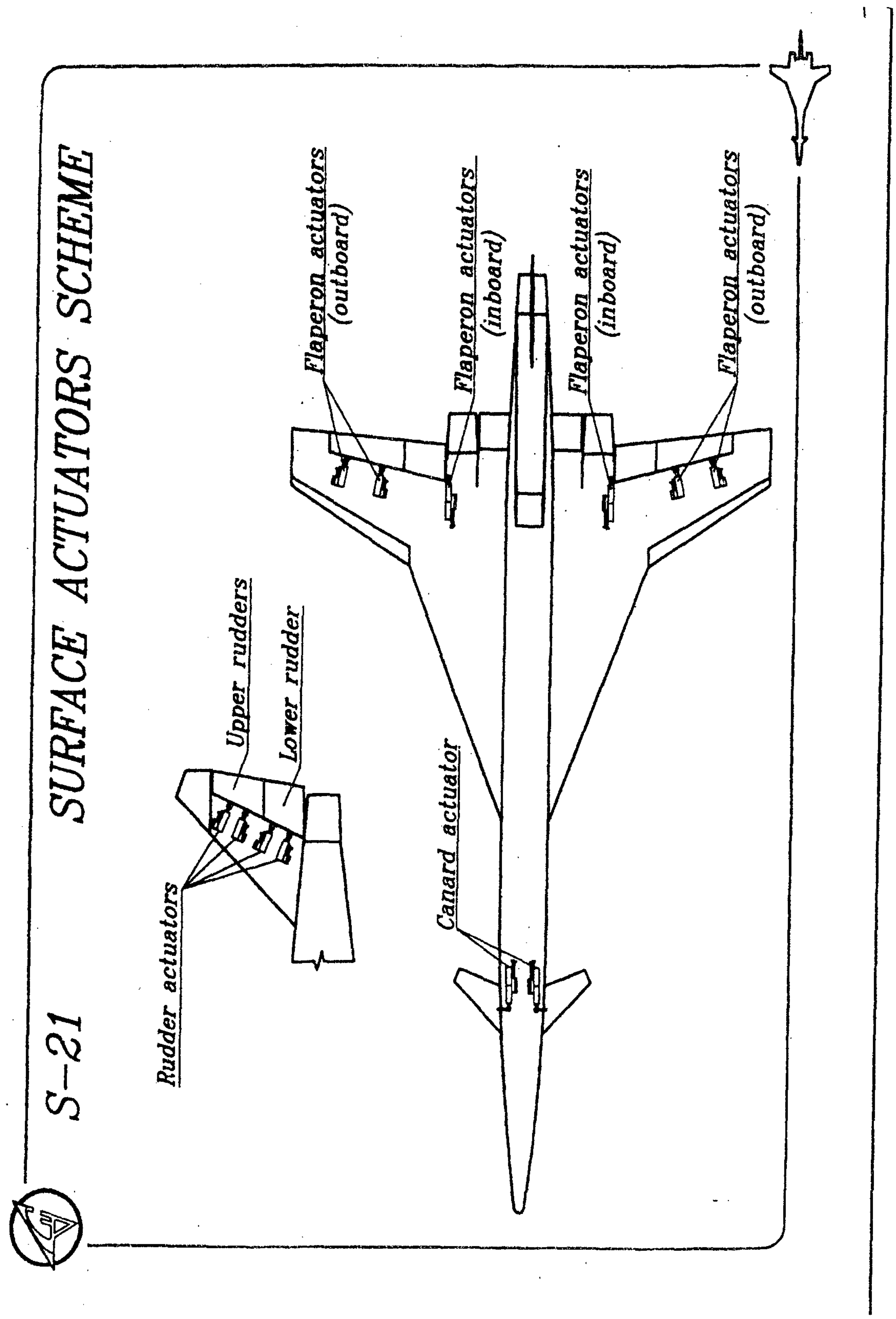
- dead zone of actuators must be about zero,
- sensitivities of actuators control elements must be enough to provide the negligible distortions produced by nonlinearities.

Ground based investigation and refinement of flight control system is a necessary stage to meet all requirements and to finish control system certification successfully, especially for S-21 type of airplane. The list of tests, testbeds and equipment and main goals of the tests is presented in fig. 7. The equipment used during control system tests must be suitable to investigate the control system and its elements at small signal amplitudes. To investigate the control system of S-21 type airplane the special equipment such as high precision frequency analyzer with dynamic range 1:2500 (for 10 V signal) is used. Typical frequency responses are presented in fig. 6b.

On the base of such experimental data the nonlinear mathematical model of actuator is developed and it is in a very good agreement with experimental results. Then this model is used to predict the airplane — control system closed loop dynamics, stability margins and so on. After such tests of all control system elements the total control system real time simulation is carrying out using the control system hardware and actuators. The comparison between calculated and experimental closed loop characteristics provide a good criterion of the reliability of control system tests.

To carry out tests mentioned above special certified equipment are needed. The equipment must include frequency analyzer with experimental data storage system and recording system. As an example of such equipment the SIEL-4200 frequency analyzer + IBM PC compatible computer can be used. This equipment enables the designer to solve all tasks during control system experimental investigation and refinement and have the next technical parameters:

— Frequency range: 0.01 Hz — 8 kHz
— Stability: no worse then 0.001%
— Test signal amplitude: 0 — 9.99 V
— Number of measuring channels: 2 in parallel
— Number of inputs: 2
— Input type: differential Rinp = 1Mohm
— Maximal input voltage: 200 V
— Resolution: 16 bits 4 decades
— **Re** and **Im** measuring accuracy: 0.5%
— Integration period: 1 — 128 cycles
— Time delay: 0 — 127 sec


S-21
SURFACE ACTUATORS SCHEME
Rudder actuators
Upper rudders
Lower rudder
Canard actuator
Flaperon actuators (outboard)
Flaperon actuators (inboard)
Flaperon actuators (inboard)
Flaperon actuators (outboard)

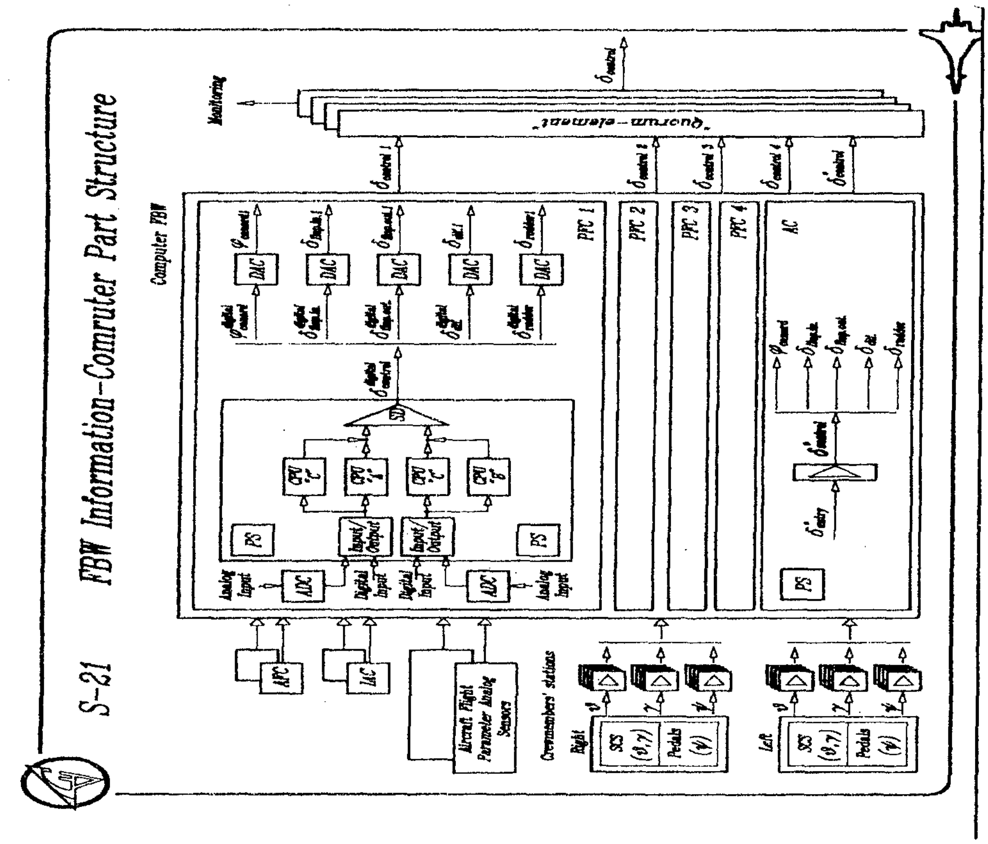

S-21
FBW Information-Comruter Part Structure

Fig. 2

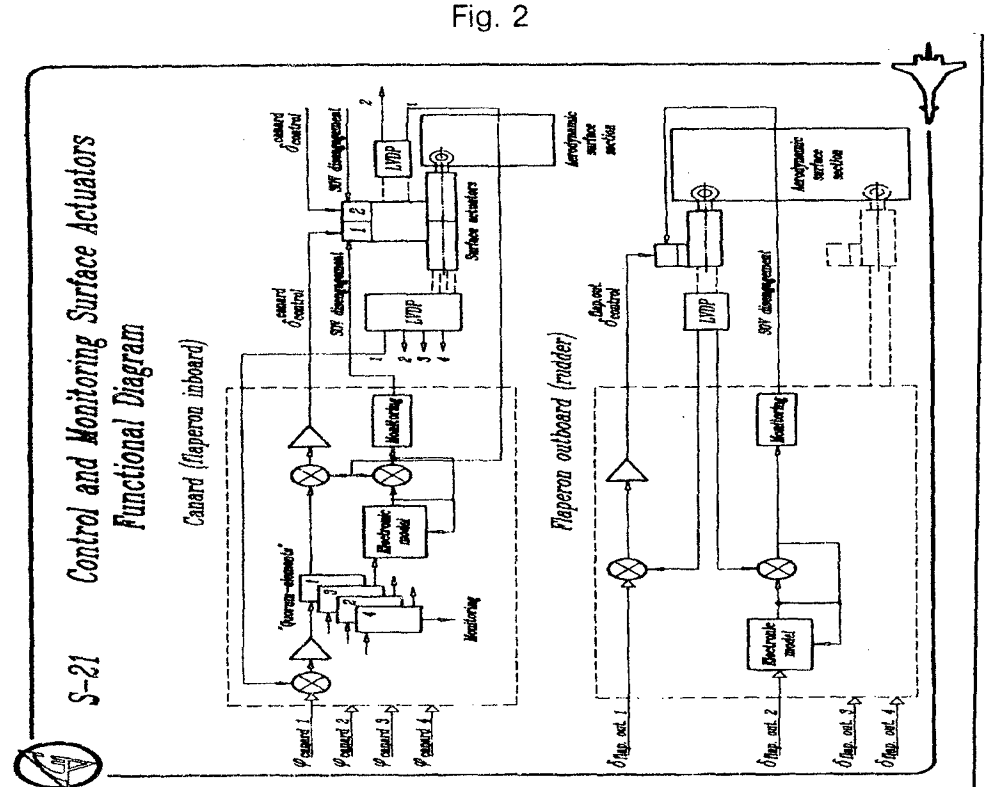

S-21
Control and Monitoring Surface Actuators
Functional Diagram
Canard (flaperon inboard)
Flaperon outboard (rudder)

Fig. 3

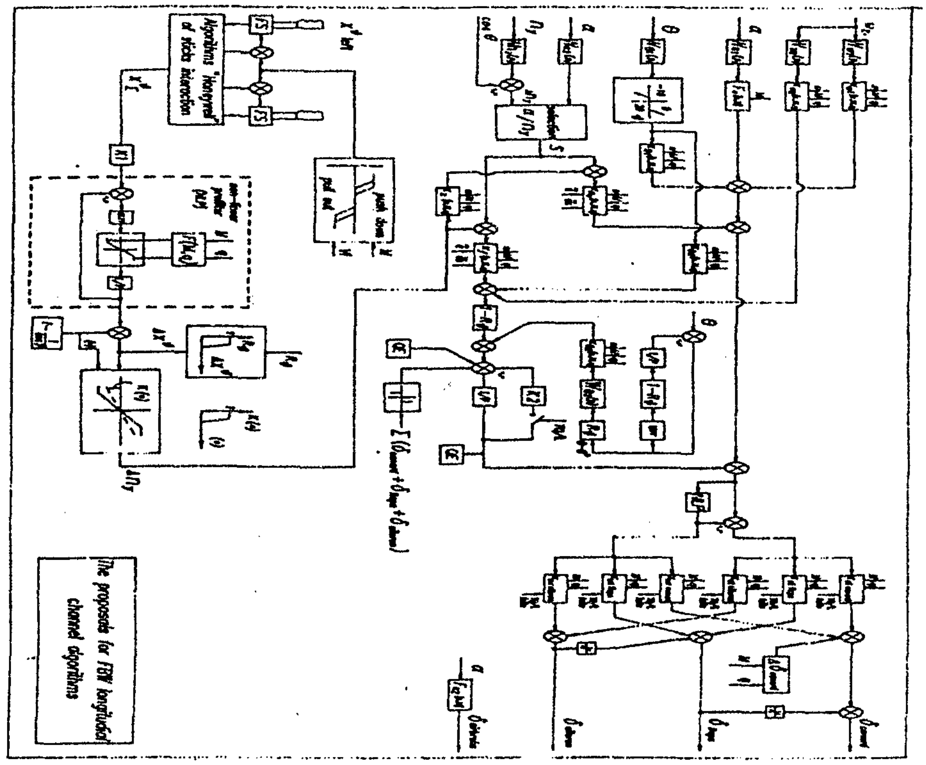


Fig. 4

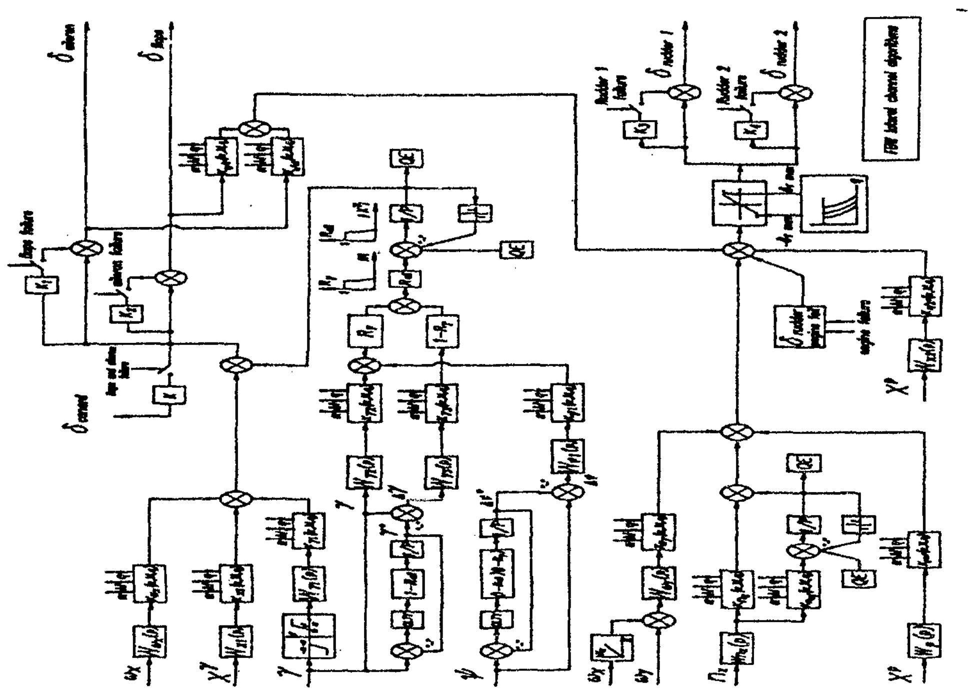


Fig. 5

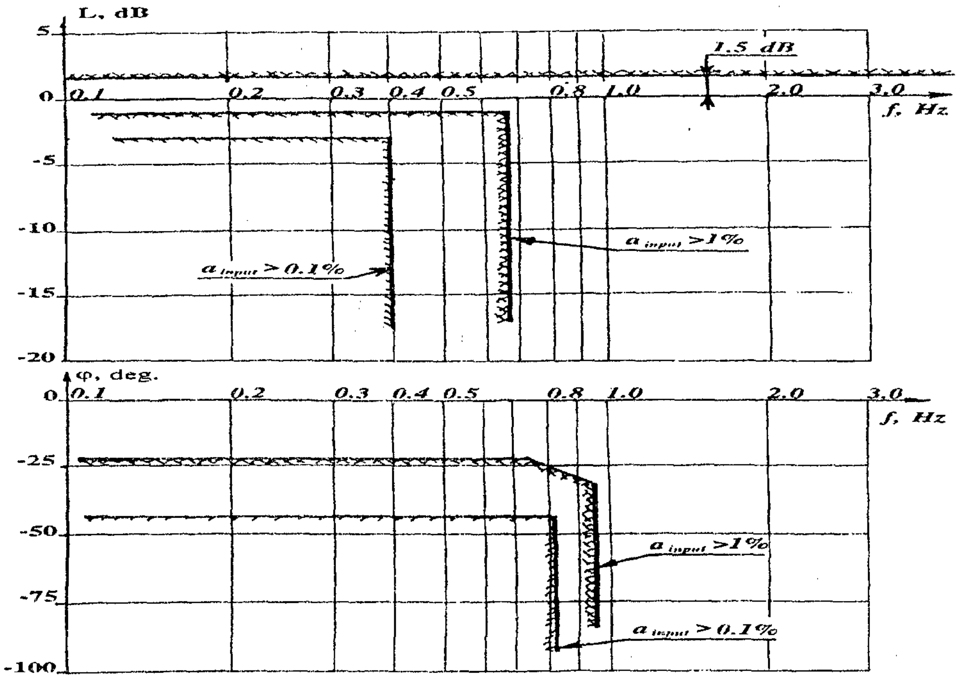

L, dB
1.5 dB
f, Hz
φ, deg.
$a_{input} > 1\%$
$a_{input} > 0.1\%$

Fig. 6,a

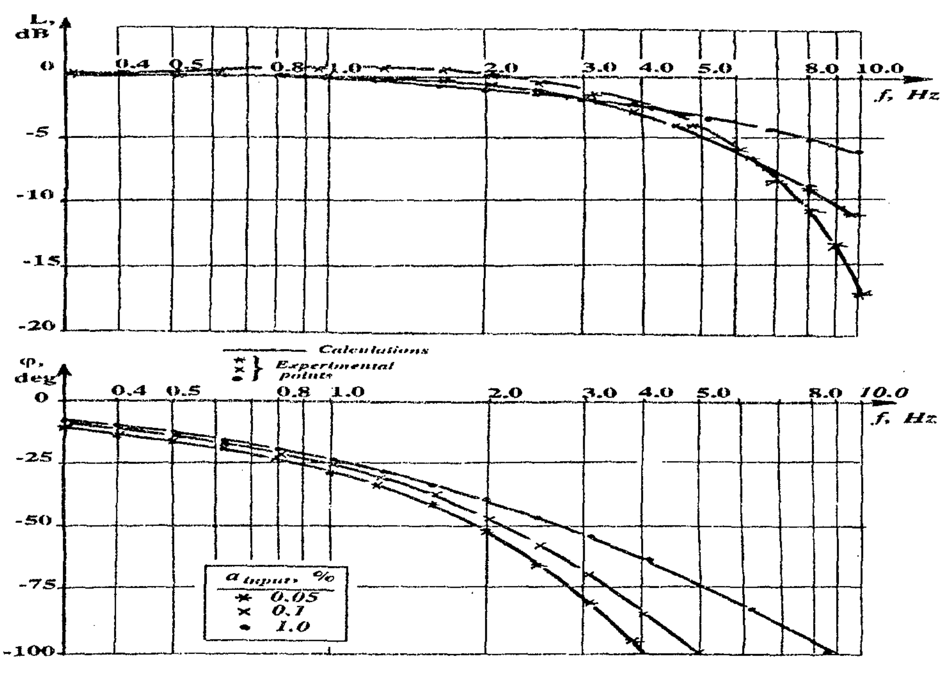
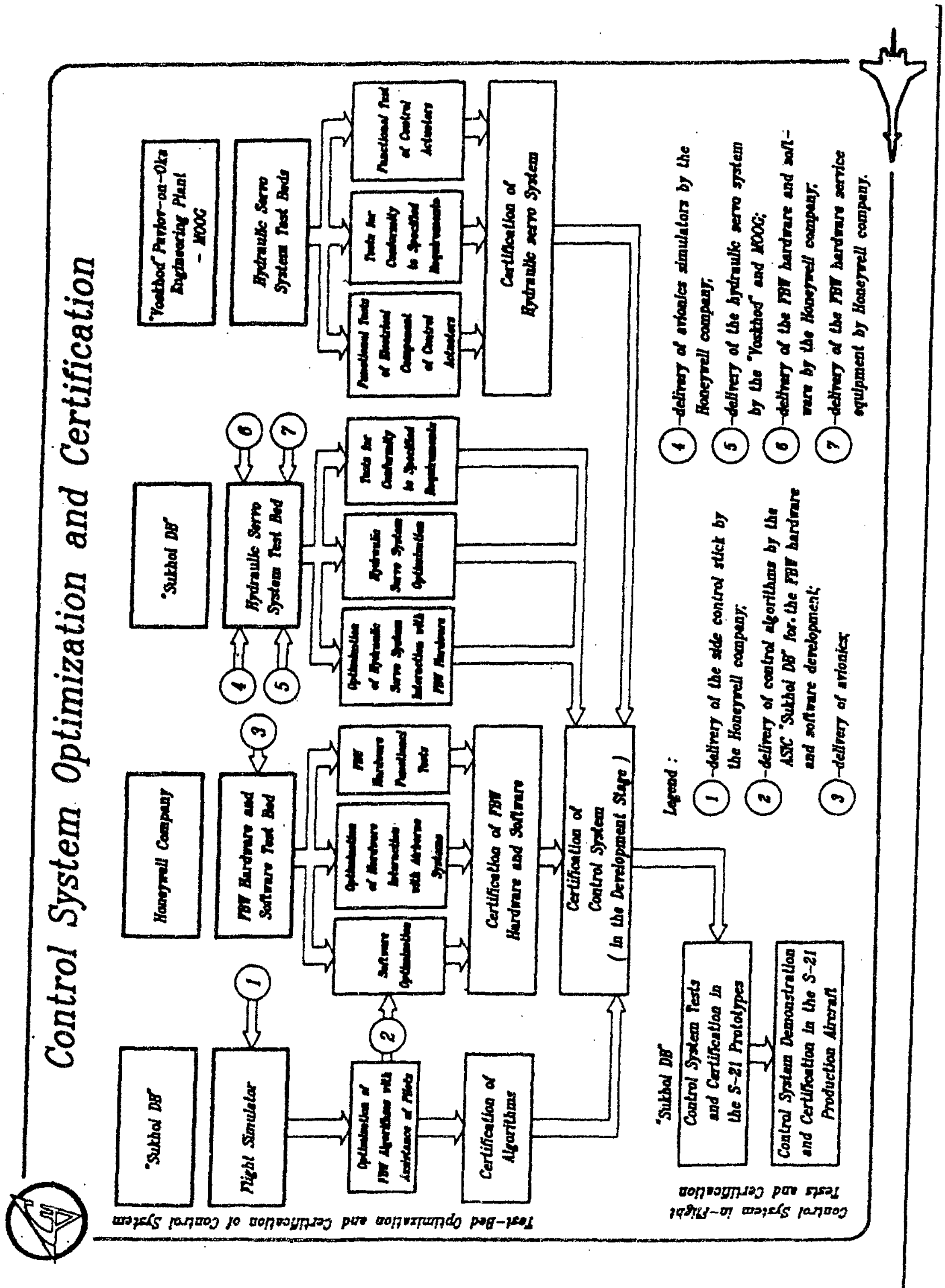

L, dB
f, Hz
φ, deg
Calculations
Experimental points
a_{input}, %
0.05
0.1
1.0

Fig. 6b



Control System Optimization and Certification
Test-Bed Optimization and Certification of Control System
Control System in-Flight Tests and Certification
"Sukhoi DB"
Flight Simulator
Optimization of FBW Algorithms with Assistance of Pilots
Certification of Algorithms
Honeywell Company
FBW Hardware and Software Test Bed
Software Optimization
Optimization of Hardware Interaction with Airborne Systems
FBW Hardware Functional Tests
Certification of FBW Hardware and Software
"Sukhoi DB"
Hydraulic Servo System Test Bed
Optimization of Hydraulic Servo System Interaction with FBW Hardware
Hydraulic Servo System Optimization
Tests for Conformity to Specified Requirements
"Voskhod" Pavlov-on-Oka Engineering Plant - MOOG
Hydraulic Servo System Test Beds
Functional Tests of Electrical Component of Control Actuators
Tests for Conformity to Specified Requirements
Functional Test of Control Actuators
Certification of Hydraulic servo System
Certification of Control System (in the Development Stage)
"Sukhoi DB"
Control System Tests and Certification in the S-21 Prototypes
Control System Demonstration and Certification in the S-21 Production Aircraft
Legend :
1 -delivery of the side control stick by the Honeywell company;
2 -delivery of control algorithms by the ASIC "Sukhoi DB" for the FBW hardware and software development;
3 -delivery of avionics;
4 -delivery of avionics simulators by the Honeywell company;
5 -delivery of the hydraulic servo system by the "Voskhod" and MOOG;
6 -delivery of the FBW hardware and software by the Honeywell company;
7 -delivery of the FBW hardware service equipment by Honeywell company.

Fig. 7

RECOMMENDED FLIGHT PATH DISPLAY AS THE MEANS OF PILOT'S MOTOR ACTIONS DURING MANEUVERING INTELLIGENT SUPPORT

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It is known that as one of the important ways of modern and perspective aircraft efficiency increase (in first queue high maneuverable planes) aviation specialists of leading countries consider the development of biocybernetic cabin. The hardware and software of the last one must provide timely information and intelligent support to the pilot during different tasks solution (from flight planning and its preparing up to the landing and after-flight analysis).

It's caused by the fact that the requirements produced to the pilot of fourth generation aircraft T_{λ} surpass his potential as man-operator possibilities B_{λ}. This leads to aviation technique utilisation respective efficiency $\overline{Э}$ increase and to significant increase of accidents number (fig. 1). Utilisation of intelligent pilot's support system (IPSS or СИПЛ in Russian transcription) increase the summary pilot's possibilities — $B_{\lambda+\text{СИПЛ}}$ — so that they even excel the required ones T_{λ} and give to the pilot the opportunity of creative solutions of the flight tasks.

However the most known designs in this field are limited by operative-tactical tasks solution support and not drop to the level of realisations and motor actions [1], in spite of the fact that it has been just motor errors which has led to flight accidents (on pilots opinions) (fig. 2) [2,3].

From our point of view IPSS structure must cover all levels of decisions making and include itself subsystem of pilot's motor actions support (conventionally named "Maneuver" on fig. 3). This subsystem must render unobtrusive [2,3] assistance to the pilot during maneuvering at all flight stages (fig. 4).

From one hand the problem is in operative on-board definition of the most suitable and worthwhile in concrete tactical conditions basic trajectory or basic controls, and from another hand-in bringing of this trajectory or controls to the pilot, that is strictly speaking motor actions support.

As number of investigations [2—6] showed the most desirable by pilots majority interface during execution of such more or less long-term maneuvers as: take-off and climbing, route flight, ground based target attack, decrease and landing (the relative neediness for which on pilots opinions is illustrated by fig. 5), is head-up and multifunction display "road-in-the-sky" visualization in one of variants being shown on fig. 6 (concrete image depends on flight stage [2,3]). During short-term high-maneuverable air combat pilots would prefer support in the view of tactile signals (being correlated with basic controls) [2,3,7].

Assistance in the image of "road-in-the-sky" permits to realise new director control regime — "director control with eyes" (with sight or with prediction), when pilot see and try to eliminate not only current errors but also he forms beforehand control image for future. The history of this question development and its features are disclosed in papers [4—6], and fig. 7 on example of yaw angle and cross displacement errors demonstrates what kind of information pilot can obtain from nearest and distant road cross sections positions.

The methodological basis of mentioned above trajectories bank formation is the solving of optimisational boundary tasks of flight dynamics by means of direct method of variational calculus [8—10], in which optimal trajectory-coordinates $(x_i,\ i=\overline{1,3})$ and velocity $(x_4 \equiv V)$— is given by polynomials from the conditional arch length τ:

$$x_i(\tau)=\sum_{j=0}^{3} a_{ij}\tau^j+\sum_{j=4}^{5} a_{ij}\left(\tau-\tau_{i,j-3}\right)^3\left[\chi_{i,j-3}\right],\quad i=\overline{1,4}$$

so that

$$\dot{x}_i=\frac{dx_i}{d\tau}\frac{d\tau}{dt}=x'(\tau)\lambda(\tau),\quad i=\overline{1,3},$$

where $\lambda(\tau)$ — velocity of movement on a conditional arch calculated in any moment of time under the formula

$$\lambda(\tau)=x_4(\tau)\left[\sum_{i=1}^{3} x_i'^2(\tau)\right]^{-1/2},$$

a_{ij}, $i=\overline{1,4}$, $j=\overline{1,5}$ — polynomial coefficients, which is searched from the requirement of obligatory satisfaction to boundary values of phase coordinates;

$$\left[\chi_{ij}\right]=\begin{cases}0, & if\quad \tau\le\tau_{ij}\\ 1, & if\quad \tau\ge\tau_{ij}\end{cases},\quad i=\overline{1,4},\quad j=1,2,$$

and

τ_{ij}, $i=\overline{1,4}$, $j=1,2$, λ_0 and λ_k — variated parameters.

Then it is quested goal function minimum (in case the functional is time basic polynomial is not defined and instead of this it's added one more variated parameter t^* — relay shifting of engines moment of time).

If being solved for the preliminary defined grid nodes (calculation cases), which being obtained as a result of task formalisation (see example on fig. 8) [11,12], fields of solutions may be obtained:

$$\tau_{ij} = \tau_{ij}(\varphi_0, V_0, H_0, D_0, m_0, \varphi_f, H_f, P_{max}, n_{y\max}),\ i=1,2,\ j=\overline{1,3},$$

$$\lambda_0 = \lambda_0(\varphi_0, V_0, H_0, D_0, m_0, \varphi_f, H_f, P_{max}, n_{y\max}),$$

$$\lambda_f = \lambda_f(\varphi_0, V_0, H_0, D_0, m_0, \varphi_f, H_f, P_{max}, n_{y\max}),$$

$$t^* = t^*(\varphi_0, V_0, H_0, D_0, m_0, \varphi_f, H_f, P_{max}, n_{y\max}).$$

They are laid in on-board computer in the view of "basic trajectories zero approximations bank". (In case of route reconfiguration support their is used some another approach [13], but IPSS-pilot interface remains constant.)

In real flight in real tactic conditions multiparametrical interpolation is performed (see graphic interpretation on fig. 9):

$$f(\bar{x}_*) = \sum_{k=1}^{2^n}\left[\prod_{j=1}^{n}\left(\tilde{x}_j(2i_j - 1) + 1 - i_j\right)\right] f_{(i_1, i_2 \dots i_n)_k}$$

where $\bar{x}_* = \{x_{1^*}, x_{2^*}, \dots x_{n^*}\}$ — n-size "trajectories bank" input vector,

$\tilde{x}_j = \dfrac{x_j - x_{j0}}{h_j} \in [0;1]$ — relative position of "j" input on segment $[x_{j0}\,;x_{j1}]$ $(h_j = x_{j1} - x_{j0})$, $f_{(i_1 i_2 \dots i_n)_k}$ — parameter value in "k" node of multisize parallelepiped

$$\left(i_j = \begin{cases} 0, & for\ x_{j0} \\ 1, & for\ x_{j1} \end{cases} \right)$$

After this by help of defined from "trajectories bank" zero approximation in real time scale concrete optimization task is solved [8].

Reconstructed by this way trajectory is visualised to the pilot in view of "road-in-the-sky"for its further tracking in director with prediction or automatic control regime. (In spite of the fact that the majority of pilots do not want the full exchange of themselves by the automatics [2,3], for special cases [8] the regime of automatic trajectory tracking is nevertheless foreseen [14].)

In conclusion the data of fig. 10 show some results of semi-nature director tracking of recommended basic trajectory of landing under the action of turbulent atmosphere modelling [4—6,15], being held with the goal of optimal width of the "road" determination.

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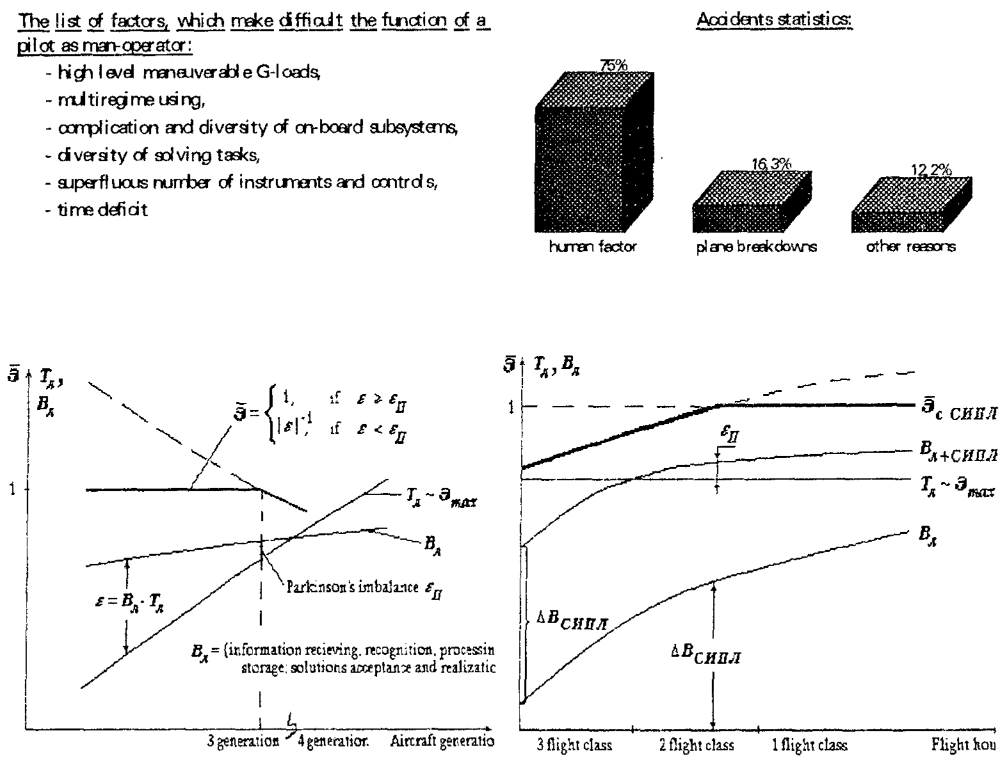


Figure. 1. Objective premises for intelligent pilot's support system development.

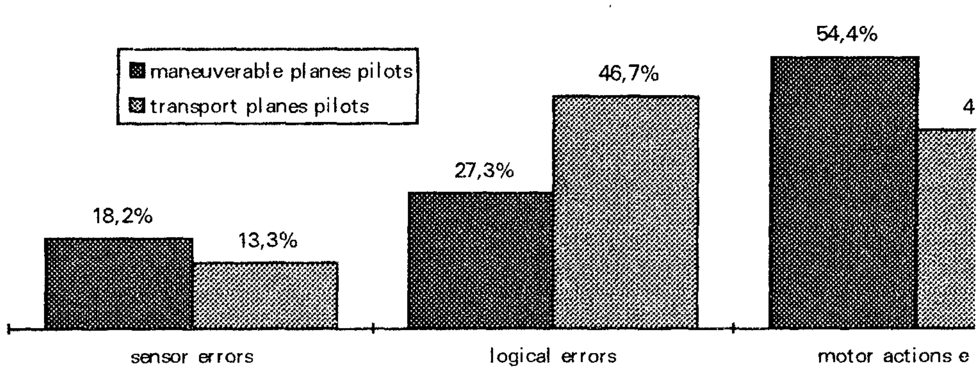
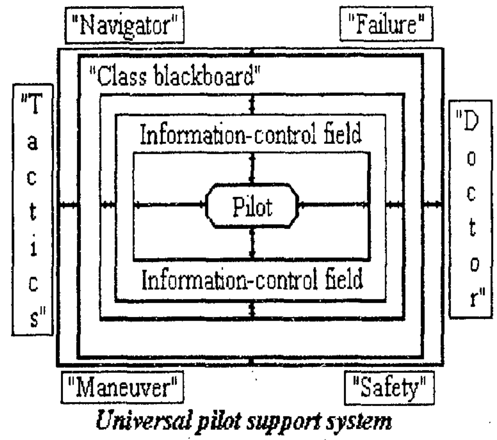


Figure. 2. Pilot's opinions about errors which evoked flights accidents.



Universal pilot support system

Decisions acceptance levels

1	global		task choice
2	strategic		nearest target choice
3	operative-tactical		task execution method choice
4	tactical (realizations)		task execution method realization way choice
5	actions		plane control pilot's motor actions

Subsystem	Solving tasks	Decisions acceptanc 1	2	3	4
"Tracking"	*subsystems*				
"Navigator"	flight task preparation, ingineering-navigation calculus making	+	+	+	
"Tactics"	targets recognition and ranging, current operation plan formation		+	+	+
"Maneuver"	take-off, route flight, ground based and air targets attack, returning to base, landing				+
"Monitoring"	*subsystems*				
"Failure"	actions when their are any failures of engines, automatics, weapons, avionics, radioelectronics, etc.				
"Doctor"	decrease of the G-load, altitude, temperature, etc. factors on a pilot; actions during pilot's safety equipment failure				
"Safety"	no exit on a critical regimes support, safety during group maneuvering, etc.				

Figure. 3. Universal IPSS conception.

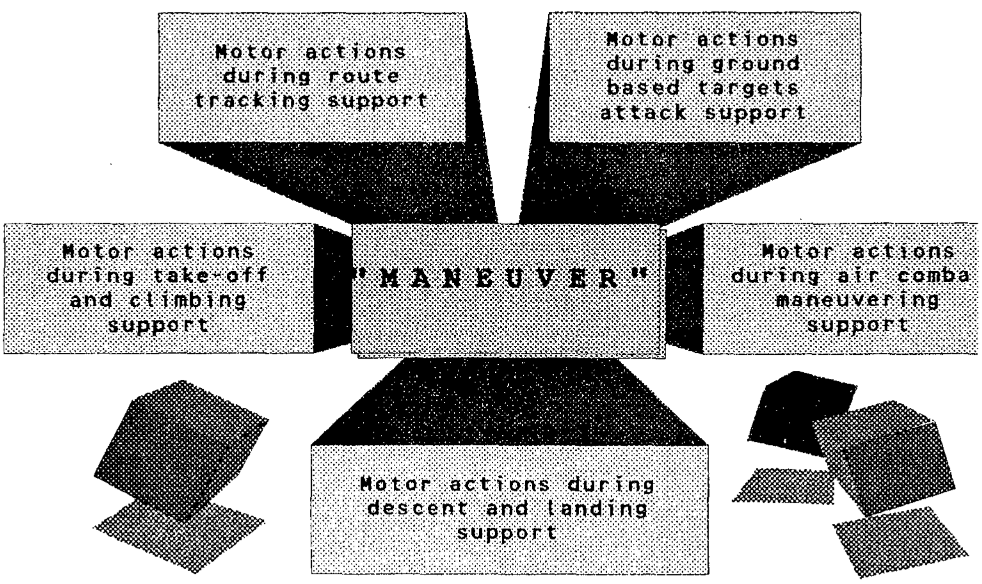


Figure. 4. Subsystem "Maneuver" components.

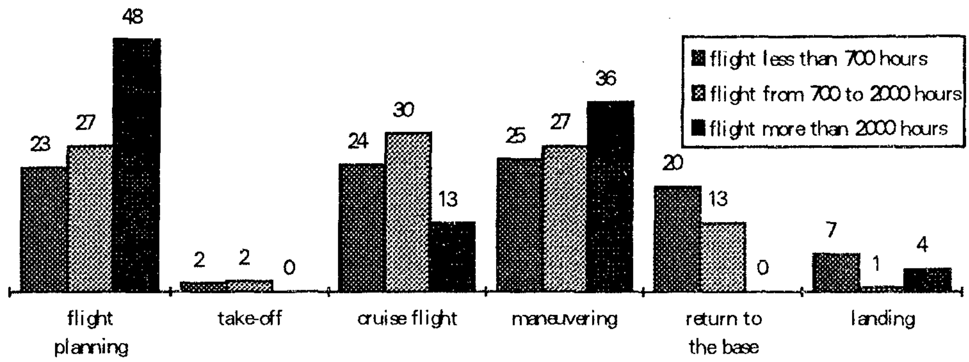


Figure. 5. Pilots opinions concerning prime importance of different stages tasks support.

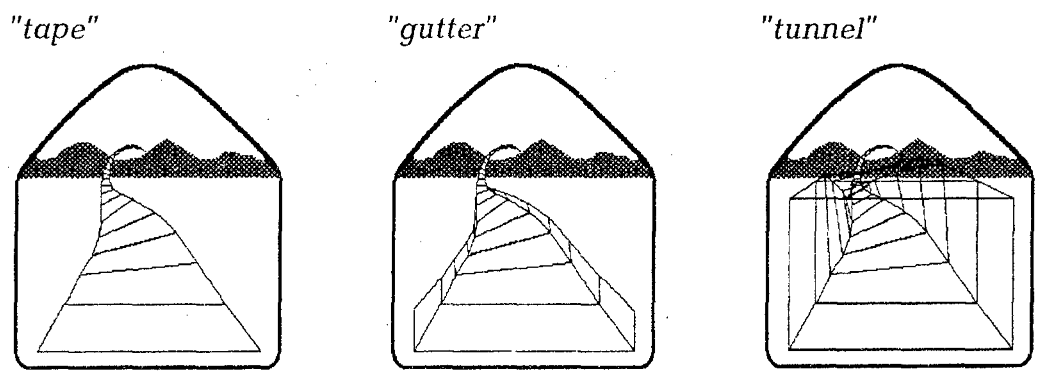


Figure. 6. "Road-in the sky" image variants.

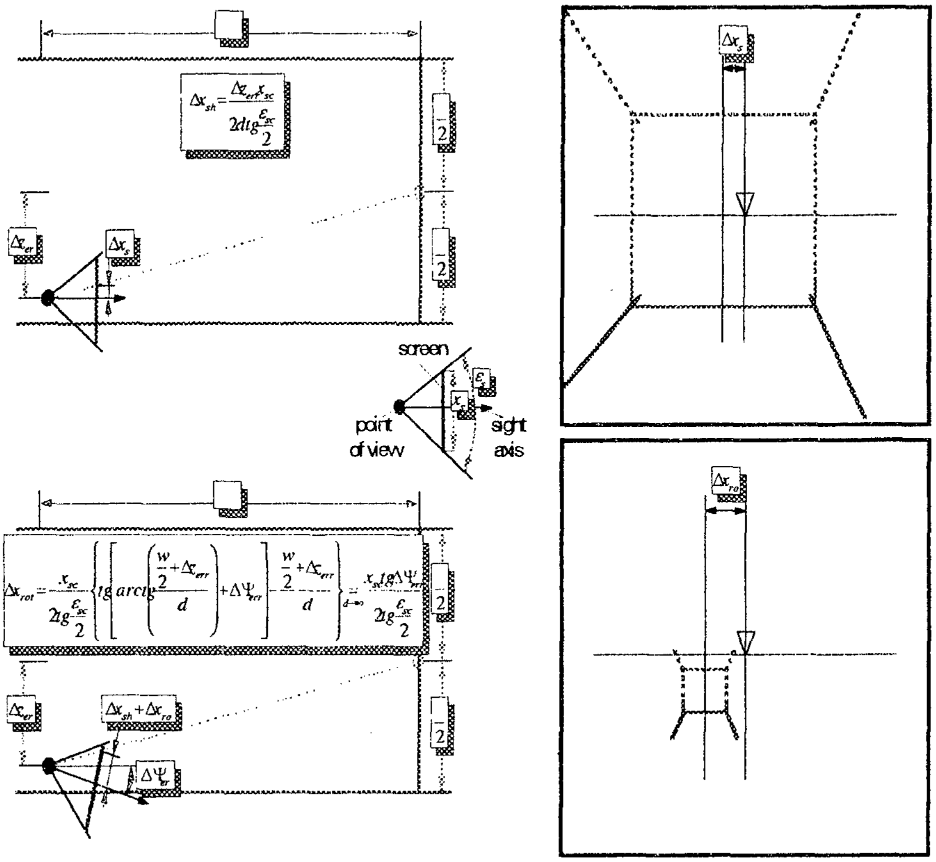


Figure. 7. Director with prediction control regime elucidation.

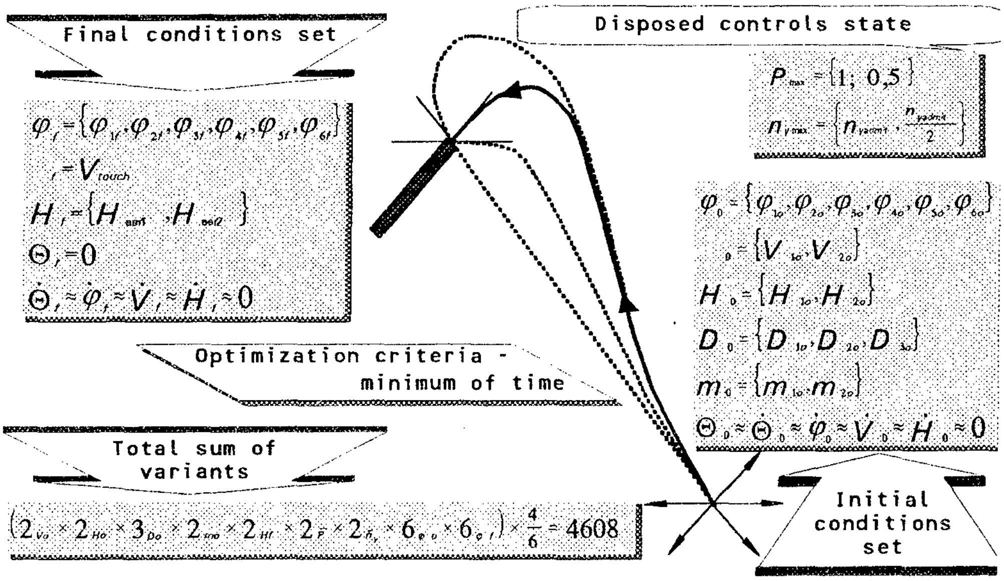


Figure. 8. An example of descent and landing task preliminary formalisation.

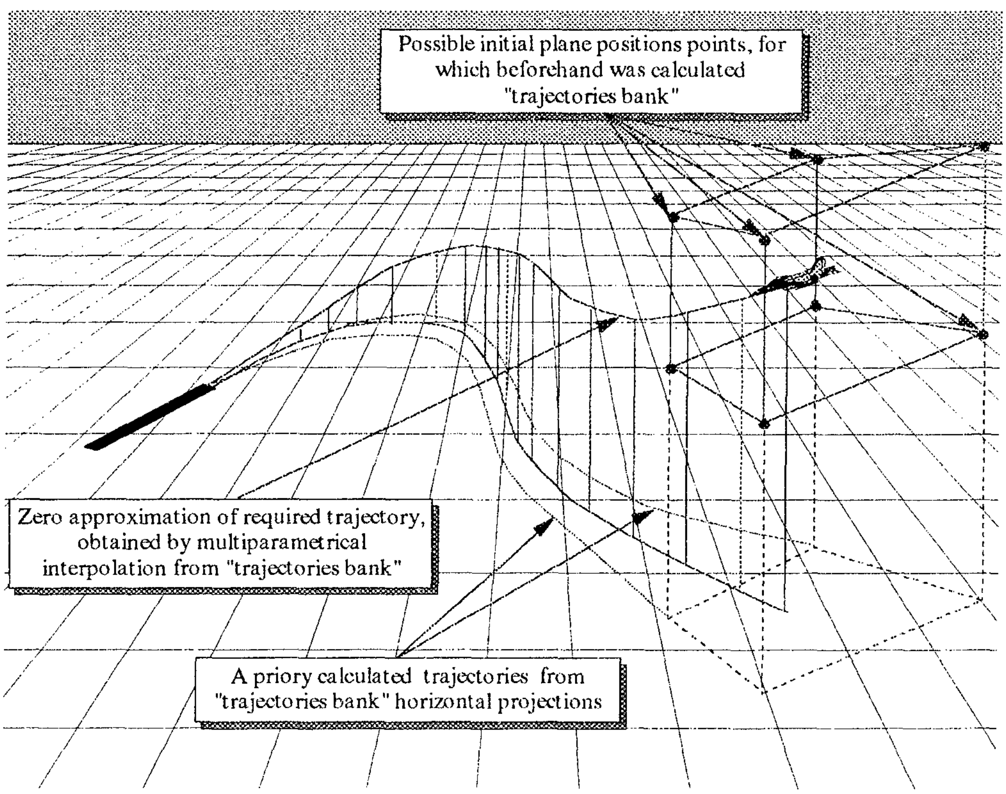


Figure. 9. Zero approximation of searching trajectory from "trajectories bank" illustration.

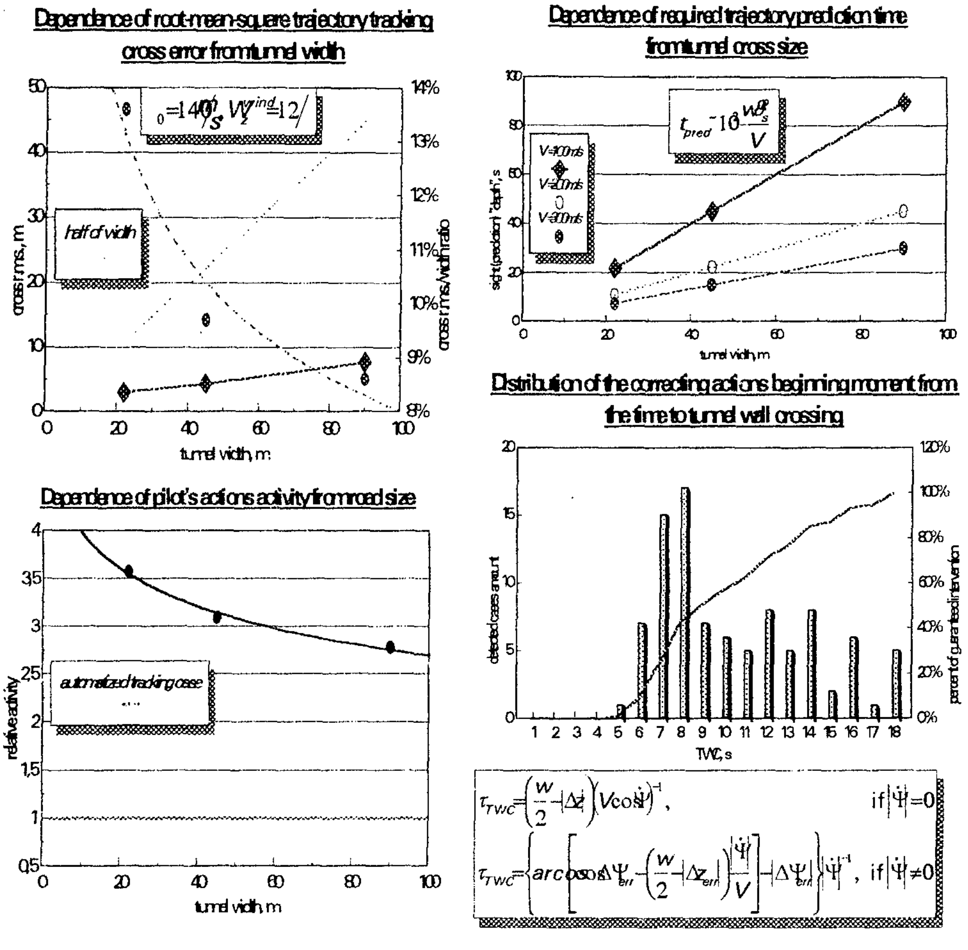


Figure. 10. Optimal road width definition.

COCKPIT INTERFACE ISSUES FOR ASTOVL AIRCRAFT – A PILOT PERSPECTIVE

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Abstract

Recent work on ASTOVL integrated flight and propulsion controls in the low speed region of the flight envelope is summarized. To date, evaluations of the combination of cockpit inceptors and augmentation have been narrowly focused, examining a limited set of tasks with a specific control scheme and limited transitioning between modes; consideration of failure modes has been deferred. With the prospect of an experimental or prototype ASTOVL aircraft on the horizon, it is now appropriate to consider a complete control system that is task oriented, accommodates mode transitions without complex procedures or adverse handling qualities and accommodates failure modes with graceful degradation to at least a safely flyable configuration. This paper presents a control design approach from the pilot's point of view, discusses the relevence of pertinent task requirements, examines several response types and proposes a combination of cockpit inceptors and response types that satisfies all mission requirements.

Nomenclature

ACAH	attitude command attitude hold
CL	conventional landing
CTO	conventional takeoff
RC	rate command
RCAH	rate commmand attitude hold
RVTO	rolling vertical takeoff
RVL	rolling vertical landing
STO	short takeoff
SL	slow landing
TRC	translational rate command
TV	thrust vector
VL	vertical landing
VTO	vertical takeoff
α	angle of attack
γ	flightpath angle
$\dot{h}$	height rate
θ_j	thrust vector angle

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Introduction

The next generation tactical jet aircraft will almost certainly be a multi-role strike fighter. Shrinking defense budgets and the emergence of a variety of potential third world adversaries dictate that future U.S. fighters be capable of performing a variety of missions, operating from ships at sea, friendly airfields bordering the theatre of operations, or in the absense of such airfields, austere expeditionary airfields comprised of short runways or sections of existing roadways. U.S. Marine Corps experience with the AV-8 Harrier V/STOL aircraft has demonstrated the considerable advantages to be gained by employing V/STOL tactical jet aircraft and has proven the concept of operating them from remote sites close to the battlefield in support of ground troops and from a variety of land-based and shipboard environments.

In recent years NASA, in collaboration with other government agencies and industry in the United States, the United Kingdom and Canada, has been engaged in the development of technology for ASTOVL fighter/attack aircraft. Several propulsion concepts have been explored including mixed-flow, vectored-thrust (MFVT), ejector augmentation, remote augmented lift (RALS) and tandem fans. In 1989 the range of configurations under consideration was reduced to the MFVT and ejector augmentor.

The integration of flight and propulsion control is essential to achieve the flexibility offered by

ASTOVL propulsion configurations. Additionally, the level of augmentation required to accommodate the low speed region of the flight envelope offers the dual benefits of reduced workload and increased precision for terminal area operations. Several integrated flight and propulsion control (IFPC) design methodologies have been investigated and subsequently evaluated in simulation. For example, a nonlinear inverse system was applied to the MFVT configuration to produce command modes of attitude, flightpath and flightpath acceleration for transition and a translational velocity command mode for hover and vertical landing. The results of the piloted simulation on the Ames Vertical Motion Simulator (VMS) showed that Level 1 handling qualities could be achieved for approaches, transitions and landings in a wide variety of conditions.[1]

The same nonlinear inverse methodology was applied to the General Dynamics E-7A configuration and evaluated in a piloted simulation on the Ames VMS for similar conditions. Those results also showed that Level 1 handling qualities could be achieved when attitude and velocity stabilization and command modes were provided. Interestingly, borderline Level 1/Level 2 handling qualities were achieved when only attitude augmentation was provided and the pilot was required to manually regulate thrust magnitude and thrust deflection as on Harrier.[2]

Modern control theory using a linear quadratic regulator design has been applied to the E-7D configuration and recently evaluated in another piloted simulation on the Ames VMS. In that simulation a combination of cockpit <u>force</u> inceptors and control augmentation was used to permit a 'frontside' technique all the way to vertical landing. This was compared to other augmentation which permitted a backside technique as in Harrier. Only hover and vertical landing tasks in a limited range of atmospheric disturbances were completed successfully. Pilots generally preferred the backside arrangement but a major objection to the force controllers overshadowed the results.[3]

Based on the results of several Ames Research Center simulation investigations of V/STOL aircraft performing decelerating approaches in instrument conditions, the V/STOL Research Aircraft (VSRA) was developed. The VSRA is the YAV-8B prototype, highly modified to use a limited authority control system to conduct in-flight evaluations of advanced V/STOL control and display systems with the intent of validating the results of the previous simulations. This aircraft is nearing completion of the checkout phase and will soon be ready to begin the important evaluation phase.

Concurrently, the Defense Research Agency (DRA) Bedford, United Kingdom, have been developing the VAAC (acronym for Vectored thrust Aircraft Advanced flight Control) with a 'two-inceptor' control law.[4] The strategy is based on transitioning from flightpath control at cruise to height control at the hover (at constant attitude) using the stick and transitioning from thrust control at cruise to longitudinal velocity (to control position at the hover) with the throttle. This leads to the somewhat unnatural input at the hover of forward stick to land vertically but it does illustrate what can be accomplished with control augmentation.

A recent Russian V/STOL Technology Assessment project, sponsored by the Naval Air Systems Commmand, has provided some flight experience in the Yak-38 Forger aircraft and insight into the Russian design philosophy for the supersonic Yak-141 Freestyle. The Yak-38 is a lift plus lift/cruise configuration that incorporates a fly-by-wire (FBW), fully integrated flight and propulsion control system in the longitudinal axis. The system was developed in the early 1970s and became operational in 1975. The Yak-141 represents an evolutionary refinement of that design. Although the system is complex, integrating the control of three engines into a single throttle for the semi-jetborne regime, the handling qualities of the Yak-38 in transition and hover are excellent.[5]

The Advanced Research Projects Agency (ARPA) has just begun a 3 year program to investigate two conceptual lift fan configurations, a gas-driven and a shaft-driven concept. These configurations are similar in many respects to the Russian Yak-141 Freestyle or previously considered tandem fan configurations with the fan rotated 90 degrees. Both of these configuration studies will lead to the design and development of an integrated flight and propulsion control system that will be evaluated in a piloted simulation. A prototype or experimental ASTOVL aircraft is considered feasible following completion of the two lift fan study programs.

To date, the combination of cockpit inceptors and augmentation has been narrowly focused, examining a limited set of tasks with a specific control scheme and limited transitioning between modes; consideration of failure modes has been deferred. With the prospect of an experimental or prototype aircraft on the horizon, it is now appropriate to consider a complete control system that is task oriented, accommodates mode transition without complex procedures or adverse handling qualities and that accommodates failure modes with graceful degradation to at least a safely flyable configuration. The components have been identified and now is the time to 'assemble' the complete flight control system and evaluate it as thoroughly as possible in simulation without the pressure of a first flight milestone.

This paper presents a control design approach from the pilot's point of view, discusses the relevence of pertinent task requirements, examines several response types and proposes a combination of cockpit inceptors and response types that satisfies all mission requirements.

Design Approach

It is reasonable to expect that a controls engineer would naturally start with an augmentation scheme based on perfect sensors for a task that is well understood such as the instrument approach. Development expands to include other modes and tasks but failure modes are generally not considered unless implementation in an actual aircraft is contemplated. On the other hand, a pilot, skeptical by nature, is always wondering how it will fly when it fails. For the pilot, the natural starting point in a control design is the failure mode that will most likely be presented after the worst combination of failures. Then it is a matter of adding modes with higher augmentation to reduce workload and increase precision. The finished product performs the same in either case but the pilot wants to understand the failure progression that will have to be dealt with.

Consideration of failure modes influences the choice of cockpit inceptors and control response types. In semi-jetborne flight, a presumed failure mode is one that leaves the pilot with direct control of the aerodynamic surfaces and manual (or direct) control of thrust and thrust deflection. Fortunately, this mode is well understood through Harrier experience and is safely flyable to landing for a cockpit layout that includes a throttle and TV lever. A 'two-inceptor' cockpit begs the question of which inceptor controls thrust and how (or if) the thrust deflection is to be adjusted.

The suggestion, then, is to begin the design process with direct control of thrust and thrust deflection and add increasingly sophisticated layers of augmentation, only as tasks may require, thus establishing a path for graceful degradation.

Task Description

In the discussion of task requirements frequent mention is made of either 'frontside' or 'backside'. There is a distinction to be made between 'backside' characteristic and 'backside' technique. The characteristic refers to the operation of the aircraft on the backside of the power required curve where the aircraft's flightpath response to speed perturbation is unstable. The technique refers to the pilot's leading input to affect a vertical change in flightpath. With the frontside technique, the leading input is with the stick and the follow up input is with throttle to maintain a constant speed. With the backside technique, the leading input to change flightpath is with throttle and the follow up is with the stick to maintain constant speed (or α). A typical carrier approach, where touchdown precision is a paramount concern, illustrates this technique.

The 'frontside' technique is clearly suitable and warranted for all flying on the frontside of the power required curve. In-flight refuelling is a task that illustrates the required control in the x-, y- and z-axes. In that task the pilot is controlling the flightpath in the y-z plane with the stick (right hand) and regulating the relative position of that y-z plane with the throttle (left hand) as depicted in Figure 1. It is a requirement of the task that he stabilize the relative position in the y-z plane before adjusting the throttle to generate closure in the x-direction.

When flying an approach on the backside of the power required curve the 'backside' technique works reasonably well where the 'frontside' technique will not produce adequate flightpath response. The vertical landing is the extreme backside characteristic task. Position control in the x–y plane is achieved through attitude response. When the position is stabilized in the x–y plane through the stick (right hand), closure in the z–direction for landing is regulated with the throttle (left hand) (see Figure 1). As with in-flight refuelling, these are requirements of the task. In the operational shipboard environment the task requirement for position stabilization prior to descent is enforced by the Landing Signal Officer (LSO).

With the above distinctions regarding frontside and backside characteristics and techniques it is now appropriate to examine required tasks in the flight envelope. The task requirements dictate the control parameters; the cockpit inceptors and piloting technique should be compatible with the task.

Takeoff

The conventional takeoff is rather straightforward in that full power is set and the remainder of the task is flightpath control and configuration management. The pilot desires flexible flightpath control that ranges from level flight with maximum acceleration to the maximum climb angle at the available thrust. There is no question that a 'frontside' technique is most appropriate as it provides flightpath control in the y–z plane with a single-handed input through the stick.

The difficulty comes with powered lift and deciding how to deal with control of the thrust vector angle for the additional takeoff modes of VTO, RVTO and STO. Obviously, thrust is vertical for the VTO but then obstacle clearance may influence the pilot's decision regarding height to begin the accelerating transition and how quickly to change the flightpath from vertical to horizontal. The RVTO begins with some horizontal component of thrust and, after a short run, thrust is deflected to near vertical and the aircraft is airborne in the semi-jetborne regime. The remainder of the accelerating transition is as described above. A STO requires horizontal thrust for the initial ground run and then thrust vector deflection at a computed speed based on the gross weight and ambient conditions for the shorebased STO while a shipboard STO requires thrust vector deflection at the end of the deck run regardless of the airspeed.

Control of the thrust vector for takeoff must accommodate a wide variety of conditions and situations such that full automation may not be viable. The last two generations of Harrier have dealt with the issue with a manual thrust vector control which provides exceptional flexibility; the next generation of vectored thrust aircraft should retain at least that level of flexibility.

Cruise and Maneuvering

The task requirements for Cruise and Maneuvering flight are mentioned only briefly and for the sake of completeness. Flightpath control is primary and 'frontside' is the only technique that makes sense for these maneuvers as illustrated by formation and in-flight refuelling. The issue of thrust vector control in maneuvering flight must be considered if the propulsion system permits that capability. The use for pitch pointing or longitudinal deceleration and acceleration or post-stall attitude control is a tactical decision best left to the pilot and implemented with a manual TV control.

Approach

The task requirement for approach for any aircraft is precise flightpath and speed control. In the initial stages of the approach (when airspeed is relatively high) there is often the need to fly in formation and that would most likely occur at constant speed. A frontside technique would be most appropriate to accommodate that requirement.

For an ASTOVL aircraft, the additional parameter of longitudinal acceleration must be controlled to arrive at a hover position. At some point in the approach deceleration must begin and the aircraft will ultimately transition to the backside of the power required curve. The backside technique is most appropriate at the hover as previously discussed and the point of transition of aircraft characteristic is the most logical for transition of the piloting technique. The control law structure and degree of augmentation should facilitate that transition smoothly. It should be noted that VSTOL pilots have been making that transition successfully with only thrust control for decades in Harrier, Forger, Freestyle, helicopter and numerous other V/STOL aircraft such as tiltwing, tiltfan and tiltrotor aircraft. Direct control of longitudinal acceleration offers a significant reduction in workload.

Landing

For landing, vectored thrust aircraft have many options: CL, SL, RVL or VL. Task requirements for the specific mode differ and are illustrated by a comparison between CL and VL. The preceeding discussion regarding approach at relatively high speed and the potential requirement for formation flying apply directly to the CL and the frontside technique is appropriate for good single-handed flightpath control while thrust (or acceleration) is controlled with the throttle. The VL requires precise horizontal position acquisition and control at constant altitude, especially shipboard. The backside technique is most appropriate as it provides single-handed horizontal position control through the stick and separate height control through the throttle. Task requirements for the SL are aligned with the CL while those for the RVL are closely related to the VL.

Control Response Types

With a design approach and task requirements established, control response types can now be examined with the dual goals of reducing workload and increasing precision. The extensive research by NASA and industry, described earlier, explored and developed various control systems with specific response types. Figure 2 summarizes and compares the handling qualities of the basic response types for approach and landing tasks.

Handling qualities for rate command systems are generally Level 2 and deteriorate rapidly with turbulence or deck motion. This is consistent with fleet Harrier experience especially as wind, turbulence and rough sea states limit the range of operations due to the dramatic increase in pilot workload required to precisely control aircraft attitude and translational rates during approach and vertical landing. A significant improvement in handling qualities can be achieved by adding an attitude hold loop. This is evident in the data of Figure 2 for the VSRA which is a RCAH system although it also includes flightpath command. Attitude hold is most appreciated when workload approaches the saturation point and the added stabilization increases the pilot's capacity for divided attention tasks.

Attitude command systems reduce workload and increase accuracy by allowing the pilot to close the loop on attitude directly. Figure 2 shows that Level 1/Level 2 handling qualities can generally be achieved for ACAH systems and the deterioration with turbulence and deck motion is moderate. It is important to note that for these data the thrust and thrust deflection were controlled manually.

There is a synergistic reduction in workload when the attitude command mode is combined with a decoupled flightpath command mode. In the flightpath command mode the pilot is able to close separate loops on flightpath and longitudinal acceleration along the flightpath. Because the responses are decoupled, the workload is low and an increase in precision naturally follows the reduction in workload. This is reflected in the data of Figure 2 for the Attitude plus Flightpath Command configuration where Level 1 handling qualities are maintained except for the extremes of turbulence and deck motion.

For hover and vertical landing tasks Translational Rate Command has demonstrated yet another incremental improvement in handling qualities. This mode turns over attitude stabilization completely to the flight control computer but enables the pilot to command longitudinal and lateral translational velocities directly. These data show that Level 1 handling qualities can be maintained in spite of rather extreme turbulence and deck motion. The synergism described above for attitude and flightpath command modes is extended to the hover and vertical landing by incorporating a height rate command mode in the vertical axis.

A response type not yet investigated but worthy of consideration is a Translational Acceleration Command/Velocity Hold. This mode would eliminate the need for separate attitude and velocity trim functions.

It is important to point out that no one control mode can be employed to control an ASTOVL type vehicle throughout its entire flight envelope. Whereas a normal acceleration command mode would work quite well for high speed maneuvering, it would be ineffective during the low speed approach and landing phase. Conversely, the TRC mode which has proven effective for hover and vertical landing would be inappropriate for high speed maneuvering flight. It is necessary to match the merits of a control scheme to the task at hand.

Cockpit Implementation

The number, location and function of cockpit inceptors is a controversial subject. The essence of the problem is whether or not to transition to a backside control technique with appropriate augmentation.

The Two-Inceptor Cockpit

As noted earlier, it is feasible (and desirable) to decouple the flightpath and speed responses so that the pilot can concentrate primarily on flightpath control and workload is thereby reduced. It is possible to fly a backside aircraft (characteristic) with a 'frontside' technique provided the flight control system is suitably augmented as with the Quiet Short-Haul Research Aircraft (QSRA). In that design the system gains were selected to provide a one-to-one ratio of flightpath to pitch attitude response in the steady state so that the flightpath command is generated with a frontside control technique where commanded flightpath rate and commanded pitch rate are equivalent.[6]

Control augmentation can even extend the frontside technique to the hover and vertical landing. In fact, this is considered by some to be a candidate control scheme for the ASTOVL aircraft. As previously noted, the DRA at Bedford have begun to examine this concept in the VAAC with a 'two-inceptor' control law. The main argument in favor of this approach is that consistency of response is maintained throughout the flight envelope.

It is significant to note, however, that the 'frontside' technique afforded by the two-inceptor law is incompatible with the requirements of the task in that position stabilization in the x–y plane requires a left hand input for x-position control and a right hand input for y-position control while avoiding contamination in the z-direction. This would be analogous to cursor control on the computer screen in the up-down direction with a left hand mouse and left-right direction with a right hand mouse. Up-down motion with the right hand mouse would be zoom control of the screen. This scheme is possible and feasible but not really compatible with the task requirement for good cursor control. Diagonal and curved paths are difficult to follow with two handed inceptors for either cursor control or aircraft position control.

Consideration of failure modes presents a dilemma. If reversion to manual control of thrust and thrust deflection were to occur in the semi-jetborne or jetborne regime, which cockpit inceptor (the throttle or stick) should control thrust? And how is thrust deflection to be managed? Thrust control through the stick would maintain a consistent feel but that would place pitch dynamics on the throttle which would be odd. Reversion to a backside technique with thrust on the throttle obviates the consistency argument. Obviously, there are an insufficient number of inceptors to deal with direct control of aerodynamic surfaces, thrust and thrust deflection simultaneously.

The Three-Inceptor Cockpit

The vertical landing is inherently a backside characteristic task. The three- (or more) inceptor cockpit is an acknowledgement that the control technique will also be backside. Proponents of the two-inceptor design approach have pointed to the inconsistency in control technique as a problem because the pilot is required to transition from the frontside to backside technique as the vehicle transitions from cruise to hover. This should not be considered a problem. For over 75 years U.S. Naval Aviators have routinely and effortlessly transitioned from frontside to backside control technique during approach and landing aboard ship. Experience gained from over 20 years of Harrier flight operations further demonstrates the ease of transitioning from one control technique to another during approach to landing.

The term 'two-inceptor' is misleading by implying that a cockpit with three (or more) inceptors is a high workload environment. Regardless of the number of inceptors in the cockpit, the pilot will only be working with two at a time and then it is generally in a sequential fashion. The typical three-inceptor control system evaluated in past simulations has featured a conventional throttle inceptor, a control stick inceptor (either side- or center-mounted) and a thumbwheel mounted on either the throttle or stick to provide longitudinal acceleration along the flightpath. As previously noted, the results show significant decrease in workload and improvement in handling qualities with an increase in the level of augmentation.

The three-inceptor cockpit accommodates both the frontside and backside control technique. High speed maneuvering frontside is quite natural while the backside technique for hover and vertical landing enables single-handed maneuvering control in the horizontal plane.

Another important advantage offered by the three-inceptor control approach is the graceful degradation during failure as the control system reverts to manual control. Failure modes are reasonably handled as there are sufficient inceptors to deal with the three parameters of pitch, thrust and thrust vector. Graceful degradation to manual control is possible because the throttle is always associated with thrust and the stick is always associated with pitch dynamics.

Proposed Scheme

Having discussed a design approach, task requirements and the merits of various response types, a candidate cockpit for the next generation ASTOVL aircraft can be postulated (see Table 1). The starting point is the simplest configuration. This is a manual mode that is suitable for reversion in response to system failures and it is also appropriate for all takeoff modes and for cruise and for maneuvering flight. The conventional stick produces rate responses in pitch and roll; the throttle controls thrust; a TV lever controls θ_j; and, conventional pedals produce sideslip. This is essentially the proven concept of the Harrier cockpit except that the propulsion system will be more complex and the 'manual' TV lever is analogous to the DEL (direct electrical link) reversionary mode in the longitudinal control system of the F/A-18. It is recognized that manual control of the many actuators and nozzles of an ASTOVL propulsion system is impractical. This arrangement provides the flexibility for all takeoff modes including shipboard where TV deflection at deck exit would be difficult to automate.

Some propulsion concepts, such as the Forger,

Table 1: Proposed Cockpit Controls and Responses

	TAKEOFF/CRUISE	APPROACH/SL$^{(2)}$	HOVER/VL/RVL$^{(6)}$
Stick	RC	RCAH	ACAH or TRC
Throttle	Thrust	Thrust $\rightarrow \gamma^{(3)} \rightarrow \dot{h}^{(4)}$	$\dot{h}$
Thumbwheel	—	Accel cmd/speed hold$^{(5)}$	Accel cmd$^{(7)}$
TV lever	$\theta_j{}^{(1)}$	(backdriven)	(backdriven)

Notes: (1) Thrust vectoring may not be possible in CRUISE
(2) Requires discrete switch; thumbwheel must be in detent
(3) When pitch dynamics lose effectiveness for γ control (approximately $50°\theta_j$)
(4) Becomes $\dot{h}$ below 60 kts
(5) Speed hold not active below 60 kts
(6) ACAH if thumbwheel in detent and speed below 60 kts; TRC if selected
(7) TRC automatically disengaged if thumbwheel moved out of detent

Freestyle or Lift-Fan configurations, require an engine start or configuration management to prepare for a semi-jetborne takeoff or approach. Once the propulsion system is configured, the approach, like the takeoff, could be made with manual TV control and direct thrust response of the throttle. This is also a manageable reversionary mode to deal with sensor failures.

For additional augmentation on the approach, attitude hold loops in pitch and roll are desired for the reduction in workload that they provide. Longitudinal acceleration command with speed hold should be provided through a thumbwheel on the throttle, but below 60 kts, speed hold should be replaced with θ_j–hold to permit a smooth transition to translational response through attitude control in the hover. The augmented approach is envisioned as a selectable mode with the pre-requisite that the acceleration command thumbwheel be initially in its detent. The TV lever would be backdriven but should the pilot move the TV lever the system would revert to the basic system described for takeoff. This is essentially a waveoff consideration.

Velocity command rather than acceleration command was considered but rejected based on experience with the VAAC. In that system velocity is commanded and the acceleration is proportional to the velocity error. All of the pilots fly it like an acceleration command system by commanding a large enough speed error to generate a desired deceleration rate and then zeroing the speed error to capture a given speed.

Throttle lever response is somewhat complicated as it must transition from thrust command initially on the approach to height command at the hover. Thrust response is satisfactory until the γ response through pitch dynamics becomes sluggish. Then a switch to γ–command is desired for a reduction in workload, especially if pitch is decoupled from γ after the switch. Below 60 kts the desired response is $\dot{h}$–command. Although the switch points are discrete, they cause no perturbation if the reference position for γ–command is chosen as the γ established prior to the switch. This requires some tuning so as to ensure that maximum thrust is demanded before the command is limited by maximum control throw. This arrangement was flown in simulation with satisfactory results for a mixed flow, vectored-thrust configuration.[1]

At the hover with the thrust vector fixed, the aircraft responses are ACAH through the stick and $\dot{h}$–command through the throttle. A TRC mode is desired to extend the range of conditions in which this aircraft might operate. This should be a discrete mode engaged by switch selection only when the acceleration thumbwheel is in detent. Movement of the thumbwheel out of detent will put the aircraft back in the approach mode.

The proposed scheme identifies four inceptors. Three would be manipulated with the left hand but only two at a time would be used. This combination of inceptors and response types provides flexibility for mission accomplishment and accommodates failures by allowing graceful degradation to manual control of thrust and thrust vector.

Concluding Remarks

The vertical landing is inherently a backside task. The frontside control technique is not compatible with the task requirements for vertical landing, especially in the demanding shipboard environment. The two-inceptor cockpit actually increases workload for this task by separating the inceptors for maneuvering in the horizontal plane. Experience has demonstrated that multiple inceptors does not necessarily increase pilot workload because the pilot generally employes only two-inceptors at any one time.

Failure modes and reversionary control schemes need to be considered early in the design. Confusion arises with the two-inceptor cockpit when failure modes are considered in the semi-jetborne regime. At least three inceptors are required to deal with a reversionary mode that includes manual or direct control of thrust and thrust deflection. The three-inceptor cockpit accommodates graceful degradation where the throttle is always associated with thrust and the stick is always associated with pitch dynamics.

A control scheme based on task requirements has been proposed. Four cockpit inceptors combined with response types appropriate to the task have been identified. The manual thrust vector lever is necessary to accommodate the wide variety of takeoff modes and to allow for direct control of thrust deflection during failure modes. This design provides all the flexibility required of the ASTOVL aircraft to accomplish the strike fighter mission.

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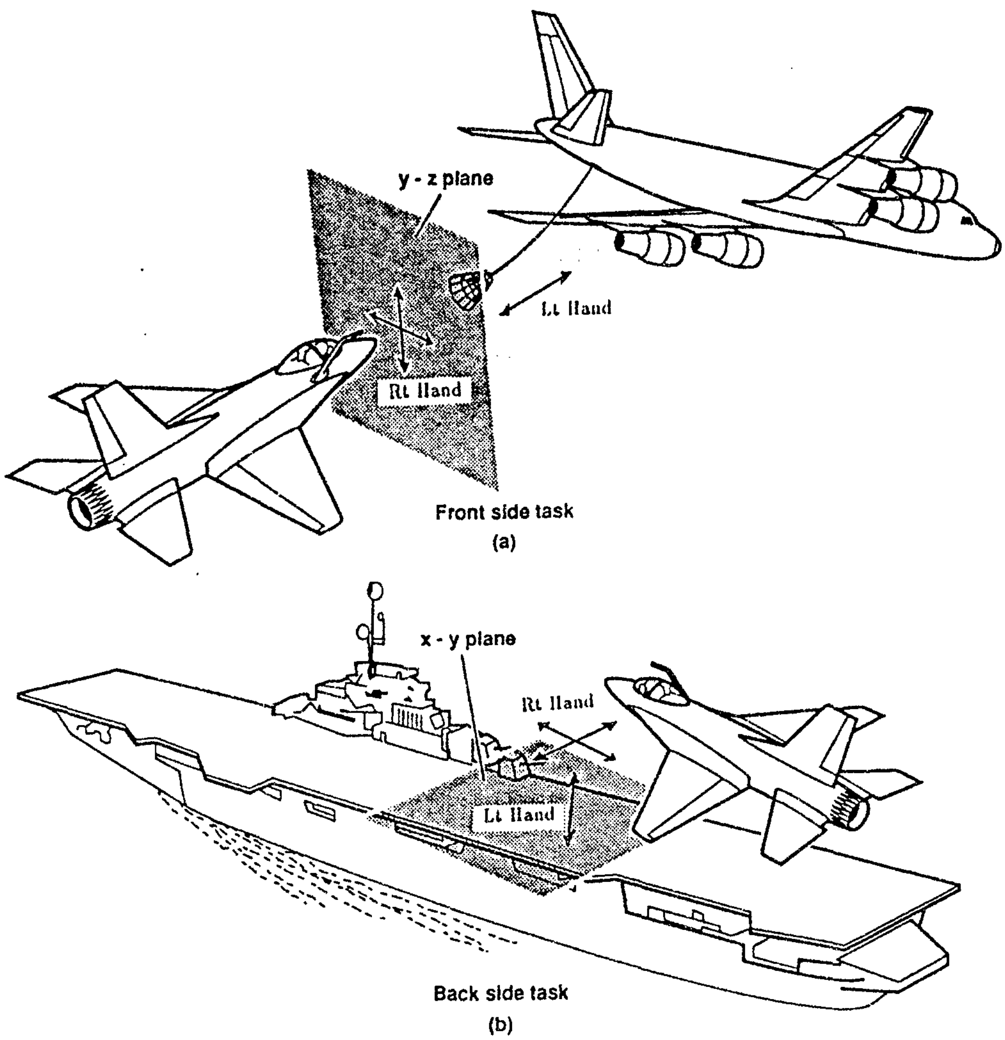

Figure 1: Frontside vs. Backside Task Requirements

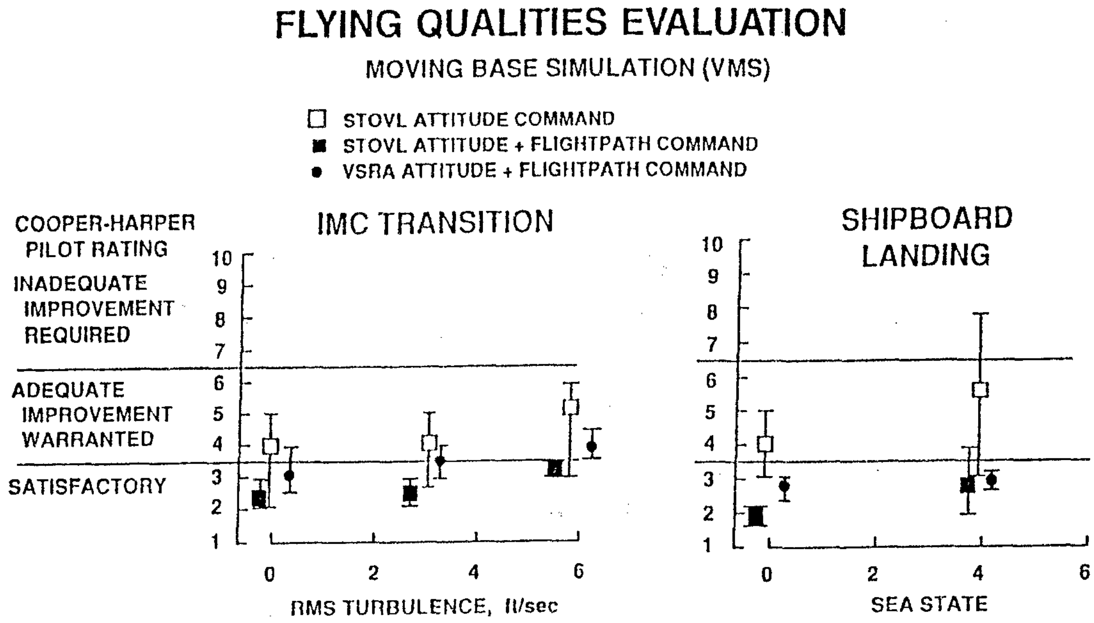

Figure 2: Handling Qualities Experience with Various Response Types

Electromagnetic Aircraft Launch System - EMALS

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Abstract—**With the proliferation of electromagnetic launch systems presently being designed, built, or studied, there appears to be no limit to their application. One of the intriguing applications is electromagnetically catapulting aircraft from the deck of an aircraft carrier. The U.S. Navy had foreseen the substantial capabilities of an electromagnetic catapult in the 1940's and built a prototype. However, it was not until the recent technical advances in the areas of pulsed power, power conditioning, energy storage devices, and controls gave credence to a fieldable electromagnetic aircraft launch system. This paper presents the U.S. Navy's Electromagnetic Aircraft Launch System (EMALS) being developed in partnership with Kaman Electromagnetics (Hudson, MA). It addresses the EMALS's present design and the technologies involved, as well as the ship and operational impacts, advantages, disadvantages, and compatibility issues for today's and tomorrow's carriers.**

I. Introduction

The U.S. Navy is presently pursuing electromagnetic launch technology to replace the existing steam catapults on current and future aircraft carriers. The steam catapults are large, heavy, and operate without feedback control. They impart large transient loads to the airframe and are difficult and time consuming to maintain. The steam catapult is also approaching its operational limit with the present complement of naval aircraft. The inexorable trend towards heavier, faster aircraft will soon result in launch energy requirements that exceed the capability of the steam catapult. An electromagnetic launch system offers higher launch energy capability, as well as substantial improvements in areas other than performance. These include reduced weight, volume, and maintenance; and increased controllability, availability, reliability, and efficiency.

II. Present Steam Catapults

The existing steam catapults currently installed on U.S. carriers consist of two parallel rows of slotted cylinders in a trough 1.07 m deep, 1.42 m wide, and 101.68 m long, located directly below the flight deck. Pistons within these cylinders connect to the shuttle which tows the aircraft. The steam pressure forces the pistons forward, towing the shuttle and aircraft at ever increasing speed until takeoff is achieved.

While the catapult has many years of operation in the fleet, there are many drawbacks inherent in the steam system. The foremost deficiency is that the catapult operates without feedback control. With no feedback, there often occurs large transients in tow force that can damage or reduce the life of the airframe. Also, extra force is always added due to the unpredictability of the steam system. This tends to unnecessarily overstress the airframe. Even if a closed loop control system was added to the steam catapult, it would have to be highly complex to significantly reduce the thrust transients to a reasonable level.

Other drawbacks to the steam catapult include a high volume of 1133 m^3, and a weight of 486 metric tons. Most of this is top-side weight that adversely impacts the ship's stability and righting moment. The large volume allocated to the steam catapult occupies "prime" real estate on the carrier. The steam catapults are also highly maintenance intensive, inefficient (4-6%), and their availability is low. Another major disadvantage is the present operational energy limit of the steam catapult, approximately 95 MJ. The need for higher payload energies will push the steam catapult to be a bigger, bulkier, and more complex system.

III. EM Aircraft Launch System - EMALS

The requirements of the EMALS are driven by the aircraft, the carrier, and the operational requirements of the carrier's airwing. These requirements are:

TABLE I
EMALS REQUIREMENTS

Endspeed	28-103 m/s
Max Peak-to-Mean Tow Force Ratio	1.05
Launch Energy	122 MJ
Cycle Time	45 seconds
Weight	< 225,000 kg
Volume	< 425 m^3
Endspeed Variation	-0 to +1.5 m/s

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The present EMALS design centers around a linear synchronous motor, supplied power from pulsed disk alternators through a cycloconverter. Average power, obtained from an independent source on the host platform, is stored kinetically in the rotors of the disk alternators. It is then released in a 2-3 second pulse during a launch. This high frequency power is fed to the cycloconverter which acts as a rising voltage, rising frequency source to the launch motor. The linear synchronous motor takes the power from the cycloconverter and accelerates the aircraft down the launch stroke, all the while providing "real time" closed loop control. The details of each component's design are presented in the following paragraphs.

A. Disk Alternator

The average power from the prime power is rectified and then fed to inverters. With power from the inverters, the four disk alternators operate as motors and spin up the rotors in the 45 seconds between launches. The disk alternator is a dual stator, axial field, permanent magnet machine (see Fig. 1). The rotor serves both as the kinetic energy storage component and the field source during power generation and is sandwiched between the two stators. There are two separate windings in the stators, one for motoring and the other for power generation. The motor windings are placed deeper in the slots for better thermal conduction to the outside casing. The generator windings are closer to the air gap to reduce the reactance during the pulse generation. The use of high strength permanent magnets allows for a high pole pair number, 20, which gives a better utilization of the overall active area. The rotor is an inconel forging with an inconel hoop for prestress. The four disk alternators are mounted in a torque frame and are paired in counter-rotating pairs to reduce the torque and gyroscopic effects. The rotors operate at a maximum of 6400 rpm and store a total of 121 MJ each. This gives an energy density of 18.1 KJ/KG, excluding the torque frame.

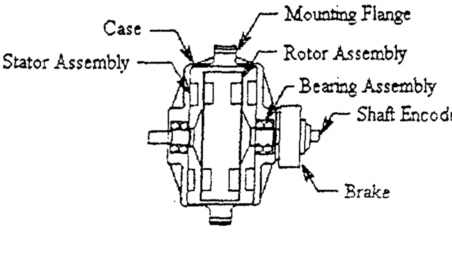

Fig. 1. Disk Alternator - Cross Section.

Each disk alternator is a six phase machine with phase resistance and reactance of 8.6 mΩ and 10.4 μH, respectively. At max speed, the output of one of the disk alternators would be 81.6 MW into a matched load. The frequency of this output is 2133 Hz and drops to 1735 Hz at the end of the pulse, for a max launch. Machine excitation is provided by the NdBFe 35 MGOe permanent magnets, which are housed in the rotor. These magnets have a residual induction of 1.05 T at 40 $^{\circ}$C and create an average working air gap flux density of 0.976 T, with tooth flux densities approaching 1.7 T. The stator consists of a radially slotted laminated core with 240 active slots and liquid cold plate. The maximum back EMF developed is 1122 V. Maximum output voltage is 1700 V (L-L) peak and current is 6400 A peak per phase.

The disk alternator's overall efficiency is 89.3%, with total losses of 127 KW per alternator. This heat transfers out of the disk alternator through a cold plate on the outside of each stator. The coolant is a WEG mixture with a flow rate of 151 liters/minute. The average temperature of the copper is 84°C, while the back iron temperature is 61°C.

B. Cycloconverter

The cycloconverter, or power electronics in general, is the pivotal technology allowing EMALS to become a reality aboard ship. With a 103 m long motor, power electronics permit efficient operation by turning on only the coils that can affect the launch at a particular time rather than the entire motor at once. It also permits EMALS to operate at its most efficient point at all speeds by allowing for a variable voltage, variable frequency supply.

The cycloconverter is a naturally commutated 3ϕ-1ϕ bridge circuit. The output of one bridge is paralleled/seriesed with outputs of other bridges to attain the power levels required. By paralleling/seriesing the bridge outputs and not the switches themselves, the design eliminates the current sharing reactors and the series capacitors. The output of a cyclo is 0-644 Hz and 0-1520 V(L-L). Simulations of the operation of the cycloconverter have been completed. Fig. 2 shows the results of a typical output waveform of the cycloconverter. As can be seen in the figure, the peak current output is 6400 A for a max launch.

The cooling for the switching assembles takes place through liquid cold plates to which the components are mounted. The medium is de-ionized water at 35°C input, 100 psig max, 1363 liters/minute. This is required to dissipate 528 KW lost in the cycloconverters.

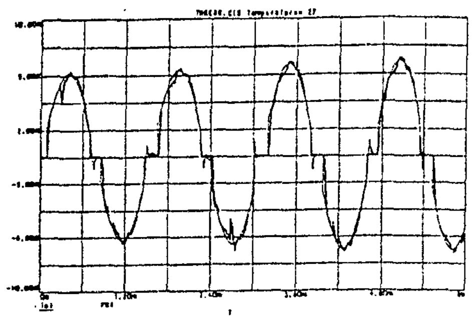

Fig. 2. Cycloconverter Simulation Results.

C. Linear Synchronous Motor

The launch motor is a linear synchronous "coilgun", as shown in Fig. 3. The trough is the same as the steam catapult trough to allow for backfit capability. The motor itself is a dual, vertical stator configuration with the active area facing outwards. The rotor, or carriage, sits over the stators much like a saddle and protrudes through the flight deck to be attached to the aircraft. The carriage contains 160 full permanent magnets, the same type used in the disk alternator, NdBFe. The carriage is restrained in two axes by rollers. The rollers run in channels welded to the stator frame. This allows both the stator and trough to flex with the ship and the carriage to follow this flexure while maintaining a consistent air gap of 6.35 mm. The stator consists of 0.640 m long segments, which are 0.686 m high and almost 0.076 m thick. These segments turn on and off as the carriage passes. The position sense system is based on Hall Effect sensors, much as in today's rotary brushless commutated motors. As can be seen in the figure, the stators are protected by offsetting them from the slot in the flight deck. This is due to the contaminants, typically jet fuel, nuts, bolts, wrenches, hydraulic oil, etc., that constantly invade the trough through the slot and could, over time, affect the stators. Between the stators, in an environmentally sealed housing, are the busbars and the static switches,

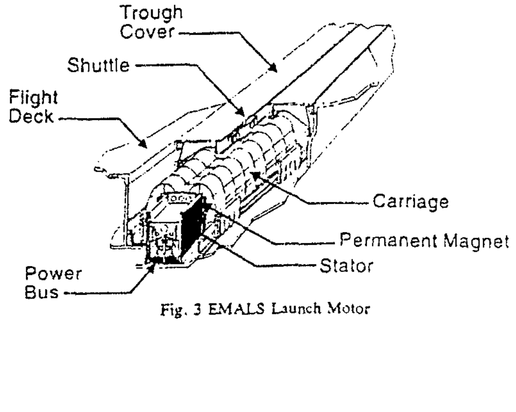

Fig. 3 EMALS Launch Motor

which are SCRs used to control the power to the stator segments.

The launcher stator is based on the modular unit called a segment. There are a total of 298 segments, 149 per side, for the entire launch motor, each 0.640 m long. The segment is wound as a three phase lap winding with 6 turns per slot and a total of 24 slots. This translates to 8 poles per segment and a pole pitch of 8 cm. These coils are epoxied into a slotless stator structure with G10 separating the coil legs. The slotless stator design keeps the phase inductance low at 18 μH. The phase resistance is 41 mΩ while the bus resistance is 0.67 mΩ. The air gap working flux is 0.896 T with the armature reaction of approximately 0.24 T. At full thrust, the permanent magnets experience a shear stress of 38 psi. At the end of the 103 m power stroke, the front of the carriage enters the brake. This brake consists of shorted stator segments, which act as eddy current brakes. At the same point in time, the carriage is still covering a number of active stator segments. Two phases are switched in these segments so that reverse thrust is initiated to help with the braking force.

With a projected efficiency of 70% and peak losses of 13.3 MW in the stator, active cooling will be necessary. Maximum coil action is 4.36e6 A^2s, resulting in a maximum copper temperature delta of 118.2°C. The launch motor has an aluminum cold plate to remove this heat from the attached stator windings and back iron. The cold plates consist of stainless steel tubes in an aluminum casting. The peak temperature reaches approximately 155°C and, after cooling for the 45 second cycle time, cools to 75°C. The carriage that houses the permanent magnets will be cooled by convection, since there will be only slight heating from eddy currents in the carriage structure and magnets.

IV. Ship Impact

The introduction of EMALS would have an overall positive impact on the ship. The launch engine is capable of a high thrust density, as shown by the half scale model that demonstrated 1322 psi over its cross section. This is compared to the relatively low 450 psi of the steam catapult. The same is true with energy storage devices, which would be analogous to the steam catapult's steam accumulator. The low energy density of the steam accumulator would be replaced by high energy density flywheels. These flywheels provide energy densities of 28 KJ/KG. The increased densities would reduce the system's volume and would allow for more room for vital support equipment on the host platform.

Another advantage of EMALS is that it would reduce manning requirements by inspecting and troubleshooting itself. This would be a significant improvement over the present system, which requires substantial manual inspection and maintenance. The EMALS, however, will require a transition of expertise from mechanical to electrical/electronic.

EMALS eliminates the complexity of the present system's conglomeration of different subsystems. The steam catapult uses about 614 kg of steam for a launch, it uses hydraulics extensively, water for braking, and electromechanics. These subsystems, along with their associated pumps, motors, and control systems tend to complicate the launch system as a whole. With EMALS, launching, braking, and retraction would be achieved by the launch motor, thereby reducing all the auxiliary components and simplifying the overall system. The hydraulic oils, compressed air, etc. would be eliminated as well as the cylinder lubricating oil that is expelled into the environment with each shot. The EMALS would be a stand alone system, completely independent of the ship's main plant. This will allow greater flexibility in the design of the ship and more efficient ship propulsion schemes.

One of the major advantages of electromagnetic launch is the ability to integrate into the all electric ship. The Navy has directed substantial research into its Advanced Surface Machinery program that is developing electric derived propulsion schemes for the next generation of surface combatants. There has also been a good deal of work in high power electric weapon systems [1]-[3]. As such, more and more of a ship's systems will evolve into the electrical counterparts of old mechanical systems. This is true of the launch, and eventually, the arresting gear. The average power required by EMALS is only 6.35 MVA. Taking these power levels off the grid should not be a problem in an all electric ship, considering multi-megawatt pumps already exist on carriers for various applications.

Perhaps the most interesting aspect of electromagnetic launch is the flexibility it offers in the way of future aircraft and ship designs. An electromagnetic launcher could easily be sized down to perform as a launch-assist system, augmenting the short takeoff of a STOVL aircraft. It can also be easily incorporated into the contour of a ramp, which provides a more efficient fly-away angle for the aircraft being launched. This reduces the required endspeed, the commensurate energy supplied, as well as the stresses on the airframe. Overall, an EM launcher offers a great deal of flexibility to future naval requirements and ship designs.

On the other hand, there are drawbacks to the EMALS. One of these is that high power electromagnetic motors create electromagnetic interference (EMI) with electronic equipment. As in the case of an electromagnetic launcher, there would be sensitive aircraft equipment sitting directly above the launch motor. Along with the aircraft equipment is the ship's own equipment, which may be affected by the electromagnetic emissions. Through proper EMC design and a "magnetically closed" motor design, EMI will be minimized.

Another drawback of an electromagnetic launcher is the high speed rotating machinery associated with pulsed power applications. The disk alternator rotors are spinning at 6400 rpm, each storing 121 MJ, for a total of 484 MJ. In a laboratory, this is not a problem, but put these rotors on a heaving, jarring platform and it becomes more complicated. In order to ensure safe operation, the flywheel and bearings are to be a stiffer design than conventional.

V. Operational Impact

Due to the inherent high level of elegant control of electronic equipment, it is possible to reduce the stresses imparted to the aircraft. The present steam catapult has relatively high peak-to-mean acceleration profiles (nominally 1.25, with excursions up to 2.0). This results in high stresses in the airframe and generally poor performance. With an electromagnetic system it would be possible to correct for deviations in the acceleration profile in typically hundreds of milliseconds, which would result in low peak-to-means. A simulation was conducted that analyzed the level of controllability of the proposed design. As shown in the simulation results in Fig. 4, the acceleration profile is smooth and flat, compared with a typical steam catapult profile shown in Fig. 5. The simulation shows that for various load conditions, the EMALS is capable of operating within the 1.05 max peak-to-mean acceleration requirement. The result of this reduced peak-to-mean is reduced stress on the airframe. To quantify the effects of a reduced peak-to-mean, a Fracture Mechanics analysis was conducted on the airframe [4] with both the steam catapult and EMALS peak-to-means. The results from this analysis show a peak airframe life extension of 31% due to the reduced stresses on the airframe. This is becoming more important as tight budgets are forcing the Navy to procure fewer aircraft. This also has the benefit of a safer operational environment, since when the EMALS experiences any unforeseen problems during a launch, it has the capability to quickly adjust and correct for them, even if a component fails during the launch.

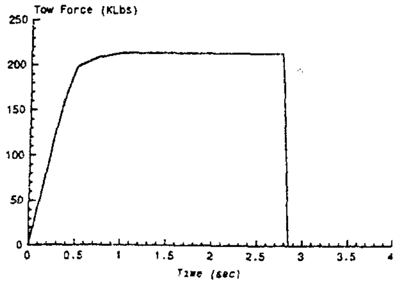

Fig. 4. EMALS Force Profile

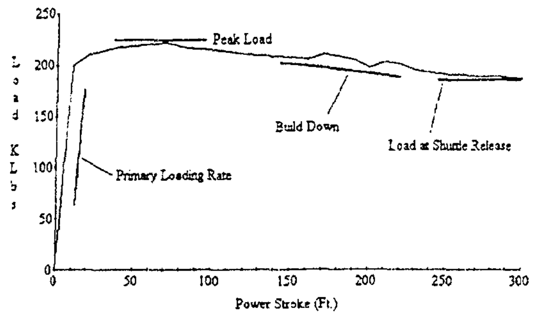

Fig. 5. Steam Catapult Force Profile

The EMALS offers the increased energy capability necessary to launch the next generation of carrier based aircraft. The steam catapult is presently operating near its design limit of approximately 95 MJ. The EMALS has a delivered energy capability of 122 MJ, a 29% increase (see Fig. 6). This will provide a means of launching all present naval carrier based aircraft and those in the foreseeable future.

VI. Present Work

The program is now in DEMVAL in a Critical Component Demonstration (CCD) phase. It is fundamentally a risk reduction phase, in which components, subsystems, and systems which pose the greatest amount of technical risk will be researched and developed to ensure that the technical issues are manageable before proceeding to full scale design. The components being developed are the cycloconverter, the stator, permanent magnets, and control system. These components are required to individually demonstrate their full design capability. For the cycloconverter this means power density, waveform generation, thermal management, and all the per unit electrical parameters the design requires. For the stator section, thrust density, thermal management, and all the per unit design parameters must be demonstrated. The permanent magnets must be able to withstand the harsh environment of the present catapult trough. This includes heat, cold, corrosive agents, shock, etc. Once these components demonstrate their design requirements, they will be integrated with each other in a test fixture. This complete launch test fixture will enable the components to mimic a launch system. It will verify the operation of the EMALS, at all speeds and thrust levels, to the required specifications of the overall launch system.

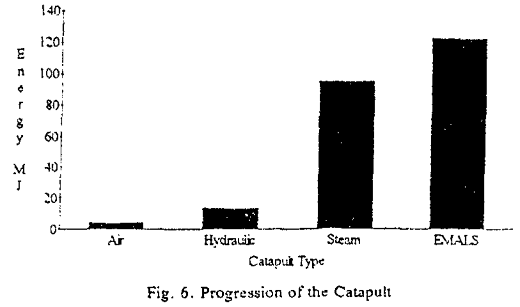

Fig. 6. Progression of the Catapult

Also, electromagnetic interference (EMI) is an issue that must be addressed early on in the design process. It must be fully understood and manageable before proceeding to the next phases of development. The high fields occur in relative close proximity to the aircraft, which houses sensitive avionics, weapons, and magnetic anomaly detection gear. It is, therefore, of prime importance to ascertain the probability of EMI between the EMALS and its neighboring systems. CCD offers a chance to address the issue of EMC. Using an electromagnetic FEA code, the shielding effectiveness of the catapult trough will be determined, as will the effects of various trough geometries. This model will be verified with hardware at low power levels. Once there is good agreement between the simulation and the empirical data, the simulation will be scaled up to the levels of EMALS.

The simulation model takes advantage of the symmetry of the trough and launch motor in the Y-axis. Since the major area of concern of the EMI issue is the fields on the flight deck, only the end turns are modeled. The coil legs running in the vertical direction will contribute little to the fields on the deck. This simulation is run at the complete spectrum of frequencies that the launch motor will produce, and the fields above the deck will be compared to the sensitivities of the various aircraft equipments. Fig. 7 shows the magnetic vector potential A for a 100 Hz, 10,000 A source. This is just a representative source to show the shielding effectiveness of the trough. As can be seen from the figure, little energy is escaping the trough structure. The magnetic fields are 0.07 mTat 10 cm above the deck at the center of the slot. Along the flight deck, the fields reach a maximum of 0.3 mT within 2.5 cm above the deck right over the coil. They fall to the Earth's ambient level at 5 cm above the deck.

VII. Conclusion

Electromagnetic motors for both launching and recovery of aircraft aboard a carrier are now possible due to a myriad of technical advancements. The advantages of electromagnetic motors are their improved performance capability over present systems and the resultant reduced weight and volume because of the high power, force, and energy densities possible. These savings are especially important on a carrier where they are precious commodities. In the future Navy, weight and volume may be of even

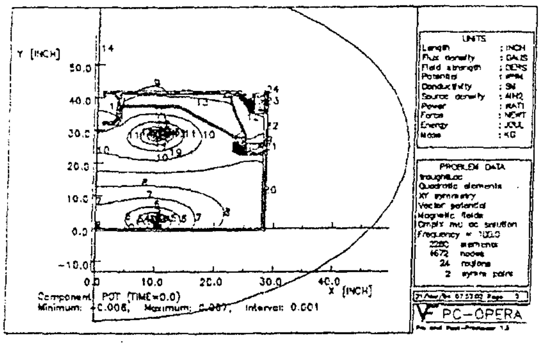

Fig. 7. Magnetic Vector Potential A, 10,000 A, 100 Hz source

higher importance as smaller budgets may demand smaller ships, and future design will require, just as in automobiles and space vehicles, etc., more performance out of smaller boxes. Electromagnetics offers this advantage. These systems would also provide the inherent controllability that comes with electrical machinery allowing for safer, less mechanically stressing operations. This will lead to extended life of airframes, nose-gear, and tail-hooks. Most importantly, electromagnetic motors will provide high level forces and greater efficiencies, which will permit the future generations of heavier, faster aircraft to operate off a carrier. Systems need to be developed that can produce the necessary performance. Electromagnetics offers a viable option.

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FLIGHT CONTROL OPTIMIZATION BASED ON THE PREDICTION OF THE FUNCTIONAL STATE OF THE CREW

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1. INTRODUCTION

The trend towards the integration of technical systems, which has begun to show itself in the last decade, will inevitably end in the integration of the man-operator and the technical system (machine) as the components of a united man-machine system.

The present lecture reflects an insight into the problem of control in the MMS from the standpoint of an expert in automation and even in optimal control. Some of the statements presented here may meet with objections among psychologists and physiologists as well as among engineers; however, I will try to expound these statements.

2. CONTROL BASED ON PREDICTION

The idea of using the prediction of the state of a system for the purpose of forming control was advanced independently by a number of scientists as long ago as the pre-war years. At a later date the idea of prediction in the interests of control was extended in different aspects and it was embodied in a number of published works.

We call your attention to the fact that the prediction of the motion of a system, as a principle, can and must play a considerably more important part than it plays now. In addition, many problems still remain to be resolved.

3. MINIMIZATION OF THE GENERALIZED-WORK FUNCTIONAL

As the central element of this method, the so-called generalized-work functional (GWF) is used, which takes, in a sufficiently general case, the form of the scalar functional

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$$I_u = V\left(x(t_\kappa)\right) + \int_{t_0}^{t_\kappa}\left[Q(x,t) + U\left(u,u_{opt},t\right)\right]dt \tag{1}$$

where $[t_0,t_k]$ is the time interval of the control procedure; $u(t)$ is the required m-dimensional control vector; $u_{\text{opt}}(t)$ is an as yet unknown optimal m-dimensional control vector; $x(t)$ is a n-dimensional vector of the state of the system; V and Q are positive-definite scalar functions.

The controlled object admits of a highly generalized formula

$$\dot{x} = f(x,t) + \varphi(x,t)u, \quad x(t_0) = x_0 \tag{2}$$

where f, φ are differentiable functions of the arguments mentioned above.

Without presenting here the statements of theorems concerning the solution of the optimization problem we formulate a generalized result permitting algorithms with prediction to be designed. This result redused, apparently, to the fact that the location of extremum of the principal part of functional (1), which has the form

$$I_* = V\left(x(t_\kappa)\right) + \int_{t_0}^{t_\kappa} Q(x,t)\,dt \tag{3}$$

agrees with the location of extremum of functional (1), which has been calculated on the same time interval.

This provides a possibility of forming an optimal (in the sense of (1)) control for (2) on the basis of the analysis of uncontrolled motion of the objects, i.e. on the basis of solving the equations

$$\dot{x} = f_M(x,t), \quad x_M(t_u) = x(t), \quad t > t_u\,. \tag{4}$$

Here f_M approximates, to an arbitrary precision, the corresponding function in (2).

4. VERSIONS OF ALGORITHMS AND COMPUTATIONAL ASPECT

Six versions of algorithms with predictive models are presently known; they have been studied to a variable extent. These versions differ in the way the gradient of the function surface over the state space is formed; as a result they lead to different (in terms of computational procedures) algorithms.

5. TECHNOLOGY OF CONTROL WITH PREDICTION

A control module, implementing the algorithms described previously, must be used in order to control different processes. As the central element (but not the sole one) of this module we must have a special-purpose processor whose function is to predict the behavior of an object and to process the results of prediction.

The processor must have three programmed inputs, namely:

— an input for current data on the state of an object;

— an input for data on the characteristics of an object;

— an input for the requirements imposed on the process.

The output of the module is supplied to actuators, display systems of a crew or to the knowledge base of an artificial-intelligence system (AI system).

One of the fields of possible efficient application of control systems with prediction, which is of the exceptional interest, represents man-machine systems.

Thus, from the point of view of the formation of an optimal control for a united man-machine system we need a formalized representation of the following: first, the models of processes running their course in these systems; second, the goals of the system.

6. CONCEPTION OF A BIOCYBERNETICAL CREW CABIN

The first published works relating to the scientific and technological trends, which can be united under the name "The problem of creating a biocybernetical crew cabin", go back to the late sixties. This line was approved at the Chicago Conference which had been organized in 1978.

A closer examination of present-day studies on this problem suggests that if we disregard various peculiarities the investigations can be classified in three lines of inquiry:

1. optimization of the distribution of functions among flight-crew members and technical systems;
2. intellectual support of decisions made by the crew;
3. biocontrol, i.e., the control of hardware by means of thought.

In our opinion, processes and algorithms for the formation of control actions in such a biocybernetical crew cabin have not been adequately investigated. At the same time, the problem of control of a complicated man-machine system is, in all aspects, among the most complex ones.

A new line for attacking the problem of the formation of control actions in man-machine systems is proposed here.The case in point is comprehensive optimization of control procedures in ergatic systems with the priority of all levels of the professional work of an operator as well as a highly perfect adaptation of the system to his personal factors.

7. KEY POINTS OF MAN-MACHINE SYSTEMS OPTIMIZATION

In the engineering psychology, two approaches to disigning man-machine systems are distinguished: machine-oriented approach and antropocentric one. The distinction lies in the dominant position either of the technical part of the system or of the man-operator and his professional work.

Fig. 1 presents two diagrams: single-loop diagram (a) for the machine-oriented approach, and double-loop diagram (b) for the antropocentric approach.

In the first case, the man is a link of the system, which has even the unusual functions. Any assertions concerning the priority of the man or machine do not change the situation from the point of view of an expert in control theory.

A different situation arises with diagram (b). Here an additional loop of the function state of the man has been introduced, which is a synonym or generalization of the psychophysiological state. Although the function state is: intimately connected with the man, this method enables us to look at the system from two sides at ones. Firstly, the man is a link of a closed system (as in diagram (a)), and, secondly, his psychophysiological essence in inexorably associated with this system. The space of activity of an expert in automatic conrtol is now a union of the physical and psychophysiological (spiritual?) spaces.

8. FORMALIZMS OF FUNCTIONAL STATE OF THE MAN-OPERATOR

The attempt of investigators to reveal, more concretily and informatively, the substance of the functional state gave rise to two approaches of attacking this problem: physiological approach and psychological one. In a simplified and, at the same time, descriptive form, distinctions between these approaches can be illustrated by the following structural formulas:

for the physiological approach

$$En \to Or \to FS \to PhP \tag{5}$$

and for the psychological appoach

$$\begin{array}{c} Mo \\ \downarrow \\ En \rightarrow Or \rightarrow FS \rightarrow EW \\ \downarrow \\ PhP \end{array} \qquad (6)$$

Notation: En = environment, Or = organizm, FS = functional state, PhP = physiological parameters, Mo = motivation, EW = effectiveness of work.

For the adherents of the physiological approach, all mappings from the environment, which determines the activity conditions of the man-operator, towards physiological parameters available for measurements make up one chain. This means that the reversibility of the mapping $FS - PhP$ ensures the unambiguous determination of the FS on the basis of the set of the PhP for any stage of the processes which run their course in the system.

The formula of the psychological approach deffers fundamentally. First, motivation is present in an explicit form, which unites, in this case, memory, experience, practical knowledge, and realized goals of the operator. Second, an integral index characteristic of the FS is the effectiveness of work (productivity, tempo, rate, number of errors, body of manufactured products, quality of manufactured products, etc.). Third (and the most important for us) the FS and PhP stand in different parallel branches. As a result, these entities are no longer an explicitly connected pair. Even with the reversibility of the mapping $Or - PhP$, there is no way or at least it is difficult to determine the FS because it is not possible to realize exhaustive checks of two inputs, namely Mo and En.

9. EXAMPLE. g-LIMITATION SYSTEM

As an illustration of the implementation of the proposed approach to the optimization of control in a MMS let us consider a system of limiting a maximum permissible (according to the state of the pilot) g-load for a highly manoeuvrable aeroplane. For methodics purposes we give here a simplified statement and solution of the problem.

An applied problem consists in the fact that the realization of manoeuvrable capabilities of an aeroplane requires a maximum complete use of the range of permissible normal g-loads and that for reasons of flight safety the boundaries of this range must be kept with a maximum accuracy.

Speaking about endurance we mean not only health and consciousness conservation but supporting of a pilot efficiency at the satisfactory level.

Solving this problem we have chosen the level of a pilot's eyesight keeness loss as an index of the pilot's functional state. The eyesight disorder may be simply connected with some aggregat of physiological parameters which is including, for example, a pulse rate, an effort of muscle groups, etc. In the g-load term the result of investigation looks like

$$\tilde{n}_y(t) = n_y(t) + \sum_{i=1}^{m} A_i \,, \tag{7}$$

$n_y(t), \tilde{n}_y(t)$ are current and permissible g-loads.

An enlarged diagram of full-scale unit for corresponding experiments is shown in the fig. 2. The standart model reproduces in speeded up time the g-load processes which are characterized by pilots as the best ones. The functional state block predicts in the same time scale the limiting g-load using the formula (7). Optimal controller realizes one law described above with functional (1) where functions V and Q have forms:

$$V = 0, \quad Q = \begin{cases} 0 & \text{if} \quad n_y(t) \le \tilde{n}_y(t) \\ \omega\left(n_y(t) - \tilde{n}_y(t)\right) & \text{if} \quad n_y(t) > \tilde{n}_y(t) \;. \end{cases} \tag{8}$$

The flight dynamics block simulates an aircraft flight in the real time. The result of simulating is made use of for control of the cenrtifuge and in the predictive model.

The results of the experiment have shown the substantial increase of the aircraft piloting exactness.

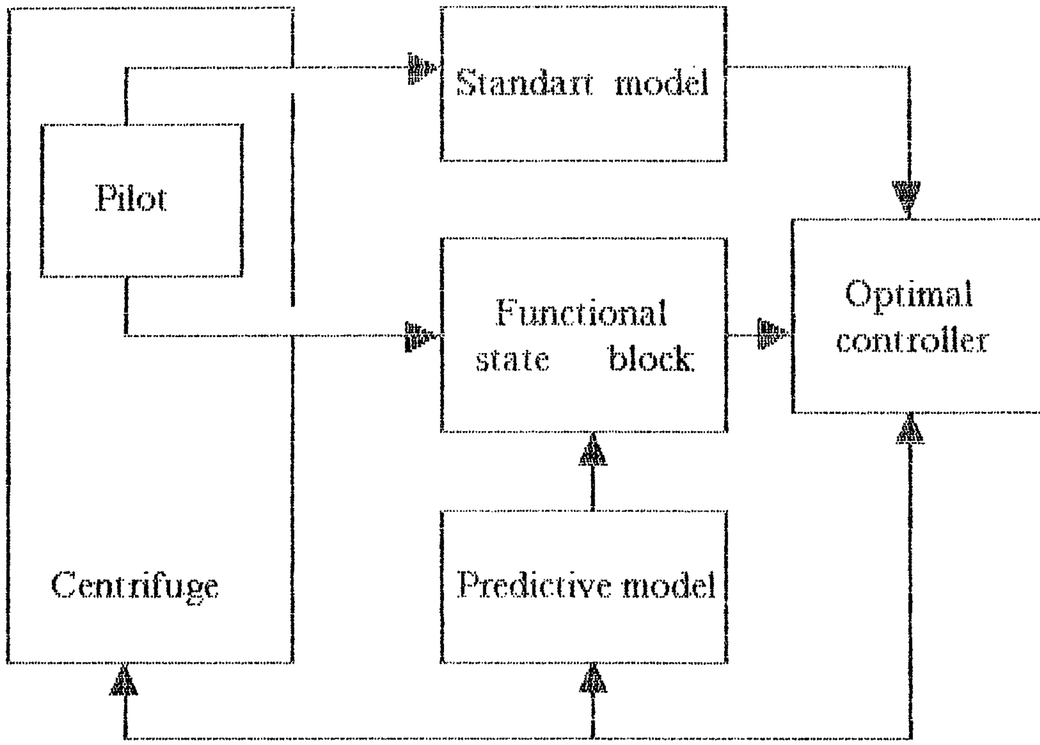

Figure 1

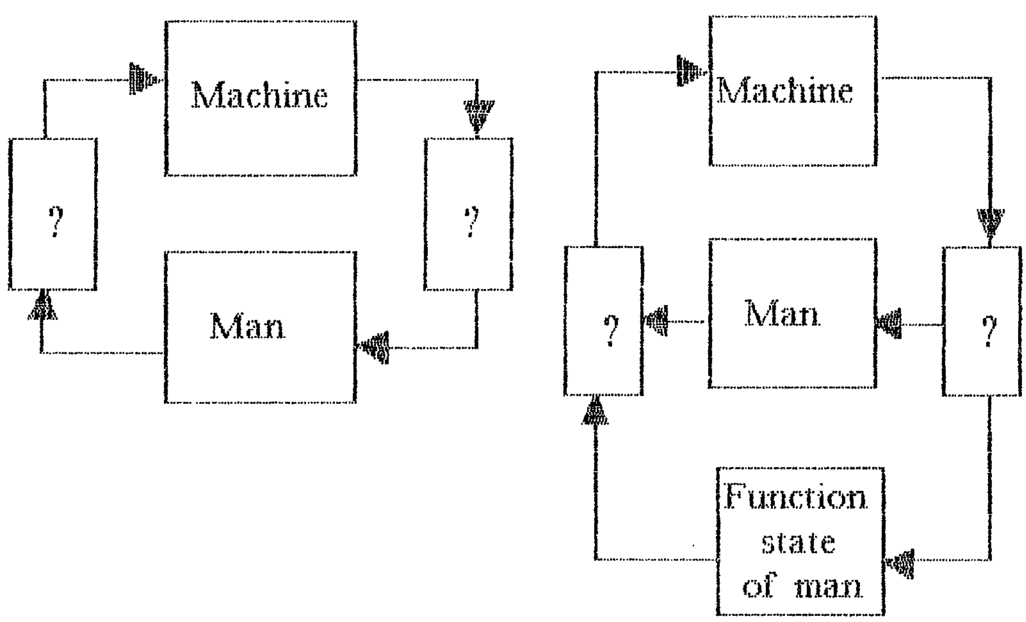

Figure 2

Robust Control Law Design for an Aircraft Longitudinal Motion.

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Abstracts.

The formalized method of the robust control law design taking into account the uncertainties of the aerodynamic characteristics and wind disturbances is proposed. The term "robustness" means, that closed-loop system will follow by the prescribed trajectory with specific accuracy under considered uncertainties. The proposed method permits to decompose an aircraft spatial motion on some uncoupled modes as in Nonlinear Inverse Dynamics Method. The method also can be used for sertification purposes, when the extreme aircraft capabilities are investigated by means of mathematical modeling.

Introduction. The main features of the used approach.

Modern aircraft is highly nonlinear and multi-dimensional control plant, especially at critical high angle of attack regimes. For such regimes aircraft mathematical model possesses some structural and parametric uncertainties due to separated flow aerodynamics at high alpha range. Flight in strongly disturbed atmosphere (windshear, "microburst"-type flows) is also very complicated flight dynamics problem. The required dynamic characteristics and flight safety in such cases can be provided by automatic control system. But traditional control law design methods using PID approach are frequently unable to provide the needed dynamic characteristics and even stability of a closed-loop system with mathematical model uncertainty and large wind disturbances. So there is a need in automatic control law design methods, which would be able to operate with aircraft characteristics uncertainties and large external disturbances.

Present report contains such approach using the so-called binary algorithm synthesis method [1]. The main idea of this method is to generate such feedback, which provides convergence of the closed-loop system to the predescribed trajectory with some accuracy σ. The value of σ is specified by the desired control accuracy and disturbances level.

Synthesis procedure involves the step-wise process of so-called inner feedback generating. These inner feedbacks are designed for a number of system state variables, which are considered as the formal control functions.

The examples of the proposed method implementation both for angular and trajectory aircraft motion are presented. The robust control laws are derived both for the case of aerodynamic coefficients uncertainty and the case of wind disturbances. The examples of closed-loop system dynamics simulation are presented. The last one demonstrates the trajectory control during landing through the region with windshear, generated by "microburst".

To solve an aircraft trajectory control problem the following chain of sub-problems: (L,H) $\rightarrow$ (V,θ) $\rightarrow$ (α, ω_z) have to be solved. The each "lower" control level must provide desired variation of "higher" level of variables. Thus, thrust-to-weight ratio and stabilizer deflection must be derived from the closed-loop asymptotic stability condition providing the L,H program following. After that the next level, i.e. the following in the (α,ω_z) subspace has to be solved, etc. The sequential connection of all these levels of control synthesis leads to "multi-stage" system structure. The successful operation of each control level depends on the similar successful operation on the lower control level. For example, the accurate tracking of the program trajectory for V^*, θ^* is impossible without fulfillment the needed accuracy of tracking on the lower control levels α, $\overline{p}$, ω_z, φ, etc. So, the fulfillment of the "global" problem is reached through the fulfillment of a number of "partial" subproblems.

<u>**Mathematical task formulation.**</u> The next system of equations is considered through process of control law design:

$$\begin{aligned} \dot{V} &= g(-\sin\theta - C_{x_a}\frac{qS}{G} + \overline{p}\cos\alpha), \\ V\dot{\theta} &= g(-\cos\theta + C_{y_a}\frac{qS}{G} + \overline{p}\sin\alpha), \end{aligned} \tag{1}$$

$$\begin{aligned} \dot{\alpha} &= w_z - \dot{\theta}, \\ \dot{\omega}_z &= m_z\frac{qSb_a}{I_z}, \end{aligned} \tag{2}$$

$$\begin{aligned} \dot{L} &= V\cos\theta, \\ \dot{H} &= V\sin\theta. \end{aligned} \tag{3}$$

We assume possibility of hierarchical division of the full system of equations onto set of subsystems: so-called phugoid (describes the center of gravity motion) - group (1), short-period (angular motion around center of gravity) - group of equations (2), implicitly supposing the characteristic time scales differences for these two subsystems. Finally, the group (3) describes center of gravity trajectory.

1 Control law design for rotational motion.

For example, 'short-period' equations for α and ω_z will be considered:

$$\begin{aligned} \dot{\alpha} &= \omega_z - \frac{\rho VS}{2m}C_{y_a}(\alpha, \omega_z, \varphi, ...), \\ \dot{\omega}_z &= \frac{qSb_a}{I_z}M_z(\alpha, \varphi, \overline{\omega}_z, \overline{\dot{\alpha}}, ...), \end{aligned} \tag{4}$$

and the satisfaction of conditions of stable tracking of the σ_α – proximity of the program α^* signal will be required.

$$\frac{d}{dt}(|\alpha - \alpha^*| - \sigma_\alpha) \leq -k_\alpha|\alpha - \alpha^*|. \tag{5}$$

Taking into account the $\dot{\alpha}$ - equation the following inequality can be obtained:

$$sgn(\alpha - \alpha^*)(\omega_z - \frac{\rho VS}{2m}C_{y_a} - \dot{\alpha^*} + k_\alpha(\alpha - \alpha^*)) - \dot{\sigma_\alpha} \leq 0. \tag{6}$$

From this inequality by means of extreme estimation (majorization) of the "central" values of the lift force $< C_{y_a} >$ coefficient by the value of the possible variation $[C_{y_a}]$ of this coefficient value (assuming the only α - dependency of the $< C_{y_a} >$ and $[C_{y_a}]$ functions), one can to transfer (6) to the next equality:

$$sgn(\alpha - \alpha^*)(\omega_z - \frac{\rho V S}{2m} < C_{y_a} > -\dot{\alpha}^* + -k_\alpha(\alpha - \alpha^*)) + \frac{\rho V S}{2m}[C_{y_a}] + \sigma^f_\alpha - \dot{\sigma}_\alpha = 0. \tag{7}$$

Inside the σ_α - proximity we decrease the requirements on inducing of the proper (5) - α - feedback by means of introducing of the smoothing monotonic $Err(\alpha - \alpha^*, \sigma_\alpha)$ multiplier ($Err(x, \sigma_x)$ is odd on first argument, $Err(0, \sigma_x) = 0, Err(x, \sigma_x) \simeq 1$ при $x >> \sigma_x$)

Further one can obtain the expression for the ω_z^* control signal, which provides inequality (5) satisfaction:

$$\omega_z^* = \dot{\alpha}^* - k_\alpha(\alpha - \alpha^*) + \frac{\rho V S}{2m} < C_{y_a} > -(\sigma^f_\alpha - \dot{\sigma}_\alpha + \frac{\rho V S}{2m}[C_{y_a}])Err(\alpha - \alpha^*, \sigma_\alpha). \tag{8}$$

Stabilizer φ control signal for the stable tracking of the ω_z^* – program σ_{ω_z} - proximity is generating in similar way:

$$\frac{d}{dt}(|\omega_z - \omega_z^*| - \sigma_{\omega_z}) \leq -k_{\omega_z}|\omega_z - \omega_z^*|. \tag{9}$$

Taking into account the equation for $\dot{\omega}_z$. one can obtain inequality similar to (6):

$$sgn(\omega_z - \omega_z{}^*)((M_{z_1} + M_z^\varphi \varphi)\frac{qSb_a}{I_z} - \dot{\omega}_z^*) + k_{\omega_z}(\omega_z - \omega_z^*)) - \dot{\sigma}_{\omega_z} \quad < \quad 0. \tag{10}$$

We are taking into account the next structure of the aerodynamical moment coefficient mathematical model: $< M_z > = < M_{z_1} > (\alpha, \omega_z) + < M_z^\varphi > (\alpha)\varphi$.

Further, by extreme estimation (majorization) of aerodynamic characterisics uncertainty, we obtain the following expression:

$$\begin{gathered} sgn(\omega_z - \omega_z{}^*)((< M_{z_1} > + < M_z^\varphi > \varphi)\frac{qSb_a}{I_z} - < \dot{\omega}_z^* > + k_{\omega_z}(\omega_z - \omega_z^*)) - \dot{\sigma}_{\omega_z} + \\ ([M_{z_1}] + [M_z^\varphi]\varphi_{max})\frac{qSb_a}{I_z} + [\dot{\omega}_z^*] + \sigma_{\omega_z^f} = 0. \end{gathered} \tag{11}$$

After smoothing of the control law in σ_{ω_z} - vicinity of the program ω_z^* trajectory one can obtain the following expression for φ^* control signal:

$$\begin{aligned} \varphi^* \;=\; & ((< \dot{\omega}_z^* > - k_{\omega_z}(\omega_z - \omega_z^*))\frac{1}{\frac{qSb_a}{I_z}} - < M_{z_1} > - \\ & ((\sigma^f_{\omega_z} - \dot{\sigma}_{\omega_z} + [\dot{\omega}_z^*])\frac{1}{\frac{qSb_a}{I_z}} + ([M_{z_1} + [M_z^\varphi]\varphi_{max}))Err(\omega_z - \omega_z^*, \sigma_{\omega_z}))\frac{1}{< M_z^\varphi >}. \end{aligned} \tag{12}$$

where $< \dot{\omega}_z^* >$ и $[\dot{\omega}_z^*]$ define correspondingly 'central' and uncertain parts of the program angular acceleration ω_z^* value.

Angular program acceleration defines by means of differentiating of the angular velocity ω_z^* program assuming $\dot{\alpha}$ - equation:

$$\dot{\omega_z^*} = (\omega_z - \frac{\rho VS}{2m} C_{y_a}) R_1 + R_2, \tag{13}$$

where

$$\begin{aligned} R_1 &= \frac{\rho VS}{2m}(< C_{y_a} >^\alpha - [C_{y_a}]^\alpha Err(\alpha - \alpha^*, \sigma_\alpha) - k_\alpha - \\ &\quad (\frac{\rho VS}{2m}([C_{y_a}] + \sigma_\alpha^f - \dot{\sigma_\alpha}) Derr(\alpha - \alpha^*, \sigma_\alpha), \\ R_2 &= \ddot{\alpha^*} + (k_\alpha + (\frac{\rho VS}{2m}[C_{y_a}] + \sigma_\alpha^f - \dot{\sigma_\alpha}) Derr(\alpha - \alpha^*, \sigma_\alpha))\dot{\alpha^*} + \\ &\quad \ddot{\sigma_\alpha} Err(\alpha - \alpha^*, \sigma_\alpha), \end{aligned} \tag{14}$$

$(Derr(x, \sigma_x) = \frac{d}{dx} Err(x, \sigma_x))$. As the result the following expressions can be written:

$$< \dot{\omega}_z^* > = (\omega_z - \frac{\rho VS}{2m} C_{y_a}) R_1 + R_2,$$

$$[\dot{\omega}_z^*] = \frac{\rho VS}{2m} [C_{y_a}] |R_1|,$$

The control algorithm presented is designed in the assumption of the accessibility of exact values of the α и ω_z phase coordinates and specified above representation of the aerodynamic characteristics. All other factors is reasonable to be regarded as disturbances.

Fig.1 presents the example of the closed-loop simulation of the $\alpha^*(t)$ - program robust tracking with the prescribed σ_α accuracy in the conditions of a priory uncertainty of the aerodynamic characteristics. The aerodynamic anti-damping effect was not taking into account in control system synthesis in the $\alpha \approx 30^0$ proximity so as the effects of pitch angular velocity and stabilizer deflections on lift force coefficient value. From the result presented one can see that the accuracy of program signal tracking is proper in conditions of high level of uncertainty of aerodynamic characteristics.

Simple approach can be used in control synthesis procedure for α and thrust-to-weight-ratio $\overline{p}$ with the aim of air velocity V and path angle θ programs tracking. Fig.2 presents the examples of the closed-loop system dynamics in conditions of specified trajectory motion tracking. Specified program requires the significant deceleration on velocity from 100 m/sec to 30 m/s in straight and level flight conditions. At the top of the picture the examples of solving of an ideal task without taking into account the disturbances and control constraints. At the bottom of the picture the example of the effect of thrust-to-weight-ratio constraint on accuracy of the program trajectory tracking. Because the impossibility of thrust reversing the specified program trajectory is physically unrealizable.

Trajectory control during landing in the region with large wind disturbances.

It is known that low-height shear atmosphere flows are very dangerous through taking-off and landing phases of flight for different types of aircraft. Special USA research program has been realized under head of FAA and NASA with the aim of investigation of this problem and increasing of flight safety by means of designing of ground-based and airborne systems of dangerous atmospheric phenomena identification.

Some aircraft dynamics distinctions in strongly disturbed air were revealed, e.g. as consequences of "microburst" atmospheric phenomenon and were formulated the recommendations for piloting technique. For example, when aircraft encounters a region of headwind, there is control, which results in pitch angle increasing, is needed. Further, when aircraft encounters region of tailwind, there is thrust increasing up to maximum value is necessary.

Similar to presented, the approach of synthesis of robust control system, is capable to solve the problem of effective compensation of wind disturbances through process of stabilization of trajectory motion through critical phases of flight in close proximity to ground.

Fig.3 shows an example of simulation the landing process when aircraft flies through region with significant wind disturbances. Wind disturbances in this case where modeled by the use of combination of two ground symmetrical circular vortices with the logarithmic profile of opposing wind. Intensity, dimensions and orientation of vortex were obtained from the condition of near-real values of strong wind disturbances.

The designed control laws provides near-to-exact tracking of the required landing trajectory, in spite of high level of disturbances. For compensation of the effect of strong downdraft during flight in the disturbed air region the control system actively changes the pitch angle and the value of thrust.

Conclusions

The robustness of the closed-loop system to the model uncertainties and wind disturbances is demonstrated by means of mathematical modeling. These results show that the proposed control law design method can be used for the development of advanced flight control systems.

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Список литературы

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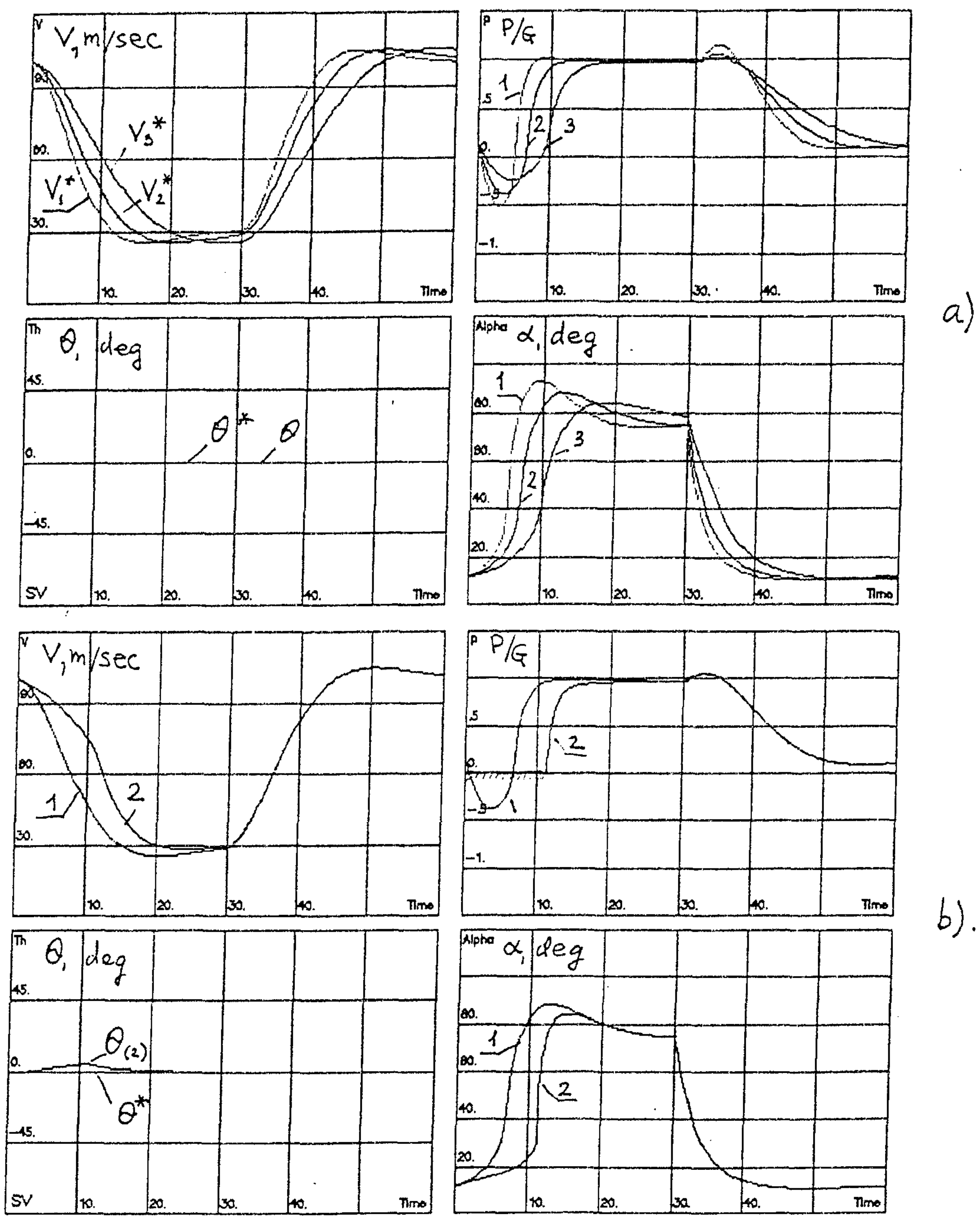

Fig. 1: Demanded $V^*(t), \theta^*(t)$ tracking; a) an ideal inverse dynamics ($\theta^* = 0$); b) thrust constraint influence.

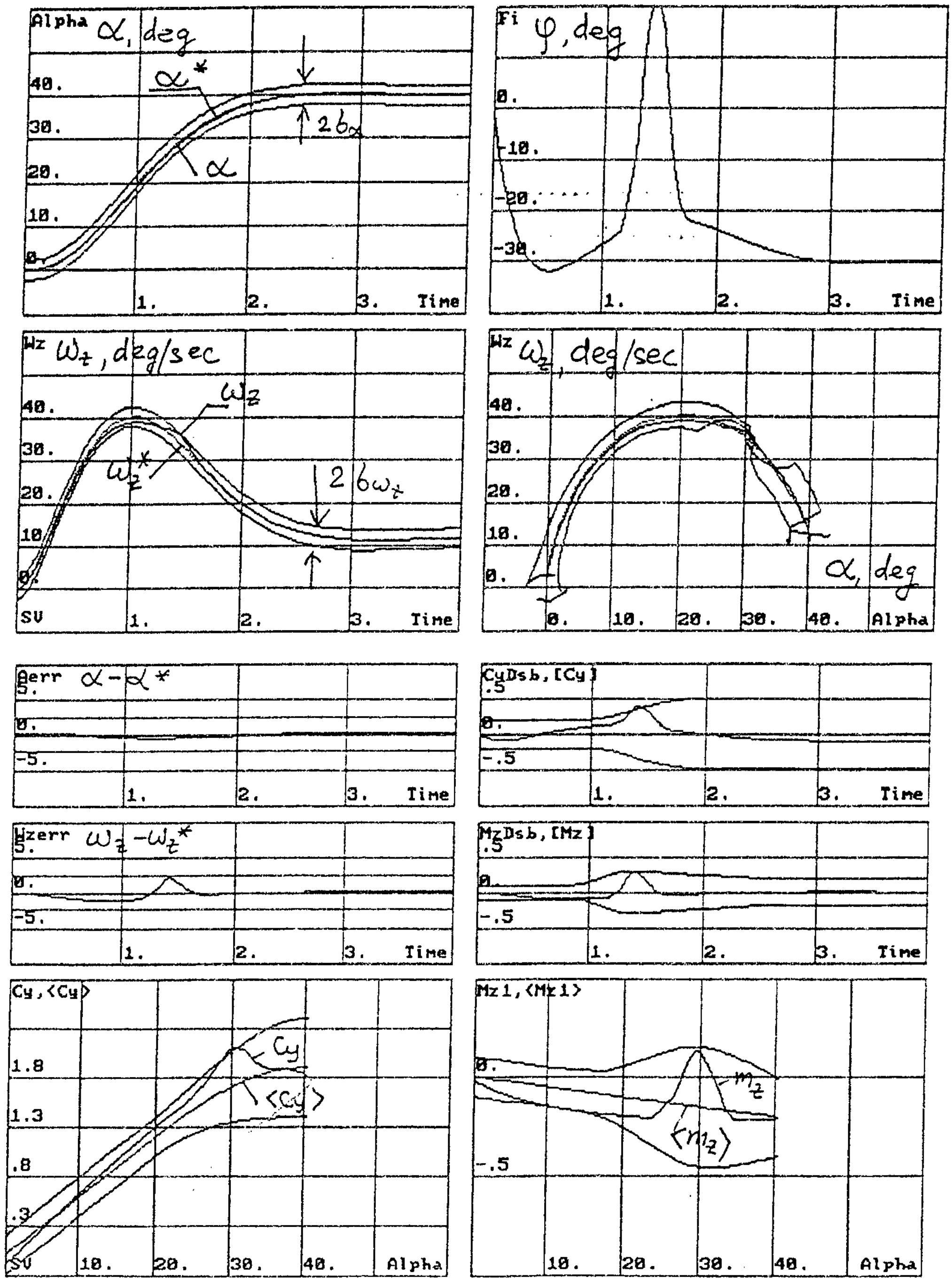

Fig. 2: Demanded $\alpha^*(t)$ tracking under aerodynamic characteristics uncertainties.

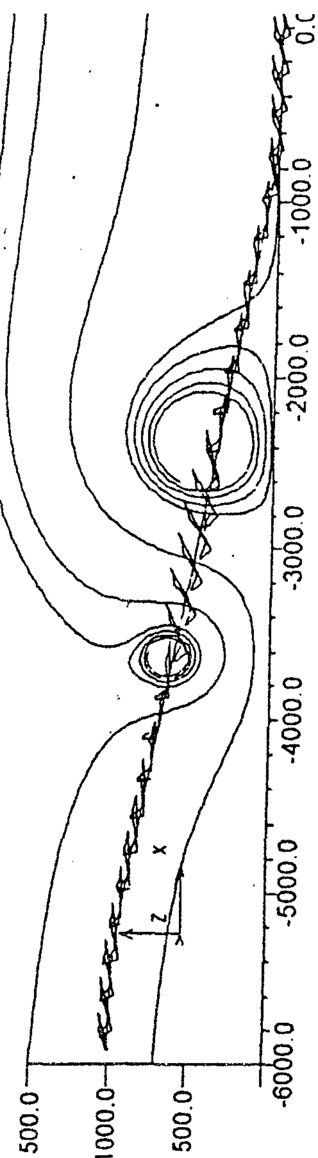

Fig. 3: Aircraft landing with robust control. The atmosphere disturbed flow.

SECTION 5

Flight Laboratories and Experimental Base for Full Aircraft Testing

LEAD FLIGHT RESEARCH OF FLIGHT SAFETY PROBLEMS USING REMOTELY COMPUTED FLIGHT SIMULATION COMPLEX

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ABSTRACT

The paper considers lead flight research of the urgent problems of design and efficiency improvement in the automated means for maneuverable aircraft flight safety.

An approach to construct the system of improving the spatial maneuvering safety and flight experiment complex for full-scale development of the advanced flight safety system main components is described. The complex is based on the Su-27 flying testbed and the ground flight experiment control system. Flight test results are given.

The report presents the concept developed by LII and TsAGI for improving the safety of maneuverable aircraft flying at high angles of attack. The concept principles are illustrated by ground and flight research data for developing airborne flight safety provision means including those for supercritical angles of attack.

INTRODUCTION

The analysis of flight accident (FA) statistics shows that among FA causes pilot errors account for a high percent and primarily those at combat maneuvering modes. As a rule, FA results from increasing effects of numerous factors and errors and leads to stalling and spinning spatial disorientation and ground impact.

Efficiency improvement of airborne warning indication and active prevention devices for limiting flight modes of maneuverable aircraft is related to introduction of digital computers, fully-deflected and quick-response control systems, electronic indicators and information system integration, state-of-the-art and the nearest future of development in these areas create good scientific and technical pre-requisites for successful solution of the problem described.

To develop principles of constructing the active systems of flight safety, stating the requirements for their structure, algorithms and equipment, flight experimental investigations of warning, indication and active flight safety devices are conducted for maneuverable aircraft.

Primary lines of research and development are following:
1. Improvement in aircraft departure and spin resistance and automatization of spin recovery.
2. Prevention of ground impact and breaking of $V_{\lim}$.

IMPROVEMENT OF FLIGHT-PATH MANEUVERING SAFETY

The concept of flight-path maneuvering safety provision, developed in LII, is based on successive solutions of the following tasks:

- revealing the abnormal situations (optimal integration of information systems, maneuver diagnostics, identifications of limiting mode, loss of consciousness);
- crew informing (boundary conditions indications [displaying] on CRT, command warning);
- active avoidance [drift from] of flight limiting modes (fully-deflected and quick-response remote control system (RCS) and auto throttle control, optimal avoiding of the hazardous mode).

To study problems arising in these situations and to develop the solutions obtained a technology was devised that includes:

- methods of analyzing and synthesizing algorithms for the systems under consideration and the appropriate complex of software for PC;
- flight experiment base for full-scale research and testing.

An important stage of research directed to devising means of warning, indication and active flight safety provision is flight experiment development of principles for constructing the system of flight safety, the requirements for its structure, algorithms and equipment, as well as direct flight test development of mockup specimens for advanced flight safety systems.

The flight experiment base of research must meet a number of specific requirements and primarily reproduce the capabilities of advanced airborne information and computer systems, RCS and information display systems (IDS). The requirements for the experimental base could be met through the use of the developed (state-of-the-art) technology of the remotely-computed flight simulation complex, including:

- a flying testbed aircraft (FTB) equipped with reliable telemetry and radio command equipment, fully-deflected quick-response remote controls, auto throttle control, advanced IDS;
- flight experiment control system (FECS) having sufficiently powerful means of processing of the telemetry and trajectory measurement information to generate real-time command and

video control and information exchange via down link and up link channels.

To support flight investigations there has been built a ground development-test facility including an aircraft cabin and identical-to-airborne kits of integrated data processing equipment, controls and a command video-control duplicate system.

Block-diagrams of flight simulation complex and its subsystems are presented in fig. 1.

Examples of real-life processing algorithms of specific situations prognosis and optimal reduction to the horizon are given in fig. 2—5.

FLIGHT SAFETY IMPROVEMENT AT HIGH ANGLES OF ATTACK

Creation of active high-angles-of-attack flight safety systems is a versatile problem.

First, algorithms for such systems can not be developed without valid data on aircraft aerodynamic characteristics at high angles of attack, information on control systems effectiveness included.

The latter is of great significance, as employment of controls at high angles of attack may considerably differ from that at low angles. E.g., a stabilizer differentially deflected at low angles of attack is used for bank control, but at high angles of attack it may serve to stop yawing.

To obtain data on aerodynamic characteristics for an adequate mathematical model to be built of the aircraft at high angles of attack there is a whole set of experiment means to be used, including wind-tunnels, free flying models and full-scale aircraft.

Secondly, automatic flight safety systems for high angles of attack flights have been developing as systems capable of warning the pilot, with help of tactile, sound and light signals of the permissible angle-of-attack exceedance and preventing an aircraft stall. Stall/Spin recovery of aircraft was placed entirely on the pilot.

As experience and air-accidents investigation results show aircraft stall/spin recovery is a difficult thing for an active pilot as it requires a proper evaluation of the critical mode and non-standard control stick actions on his part, and under the circumstances of time lack. And there can be no doubt about the urgency of aircraft control automation at critical modes recovery.

Development of aircraft having high-automation remote control systems permits aircraft control to be rendered automatic at critical modes recovery.

Thus, an auto (or director) stall/spin recovery system is an important part of the automatic high angle of attack flight safety system.

A director instrumentation is known to be utilized for F/A-18 spin recovery.

When being in the spin the pilot is given commands by means of command pointers of the digital display indicator (DDI) of the F/A-18 aircraft how to operate control levers.

Tests trials of the auto "deep" stall/spin recovery system were conducted on the Su-27 flying testbed in the LII. ADS/SRS was developed in the framework of the standard remote control system for the Su-27 aircraft.

"Deep" stall modes are the characteristic features for modern aircraft (F-16, Su-27) with small static stability reserve. The special type of control is required at small pitching moment reserves to recover an aircraft from "deep" stall. This is a so called pitch "waggle" of an aircraft. Fig. 6 presents the flight records of the pitch "waggle" system operation. The pilot put the aircraft from braking into "deep" stall mode at the indicator angle-of-attack 38—40°, then set the controller in the neutral position and activated the pitch "waggle" system. There immediately appeared the vibration on the angular pitch rate and angle-of-attack with an increasing amplitude. After two "waggle" cycles the angle of attack was reduced to α_{limit} and the system switched off, while the remote control system was transferred in the standard operation mode.

Fig. 7 presents the in-flight records of the spin recovery system operation. The pilot put the aircraft in the flat spin mode ($\alpha = 70^\circ$) with intensive spinning (angular yaw rate 1.4 s^{-1}), set the control levers in the neutral position and activated the spin recovery system. Immediately after the movement of the differential stabilizer to the recovery there appeared an acceleration for reduction of the angular pitch rate and after 7.5 s the aircraft reached the admissible angle of attack and the system deactivated. The remote control system entered the standard operation mode. Altitude loss at the spin recovery was 100 meters.

LIST OF SYMBOLS

V_i	→	indicated airspeed, [km. p. h]
H_i	→	indicated altitude, [m]
a_i	→	indicated angle of attack, [deg]
n_y, n_z	→	lateral, normal acceleration, [g]
p, q, r	→	roll, pitch, yaw rate, [rad/sec]
$q, \varnothing$,	→	pitch, roll angle, [deg]
d_h,	→	horizontal tail deflection, [deg]
X_h,	→	longitudinal stick deflection, [mm]
d_{dh},	→	differential tail deflection, [deg]
X_a,	→	lateral stick deflection, [mm]
d_r,	→	rudder deflection, [deg]
X_r,	→	rudder pedal deflection, [mm]
ADS/SRC	→	Auto(matic) Deep Stall/Spin Recovery System

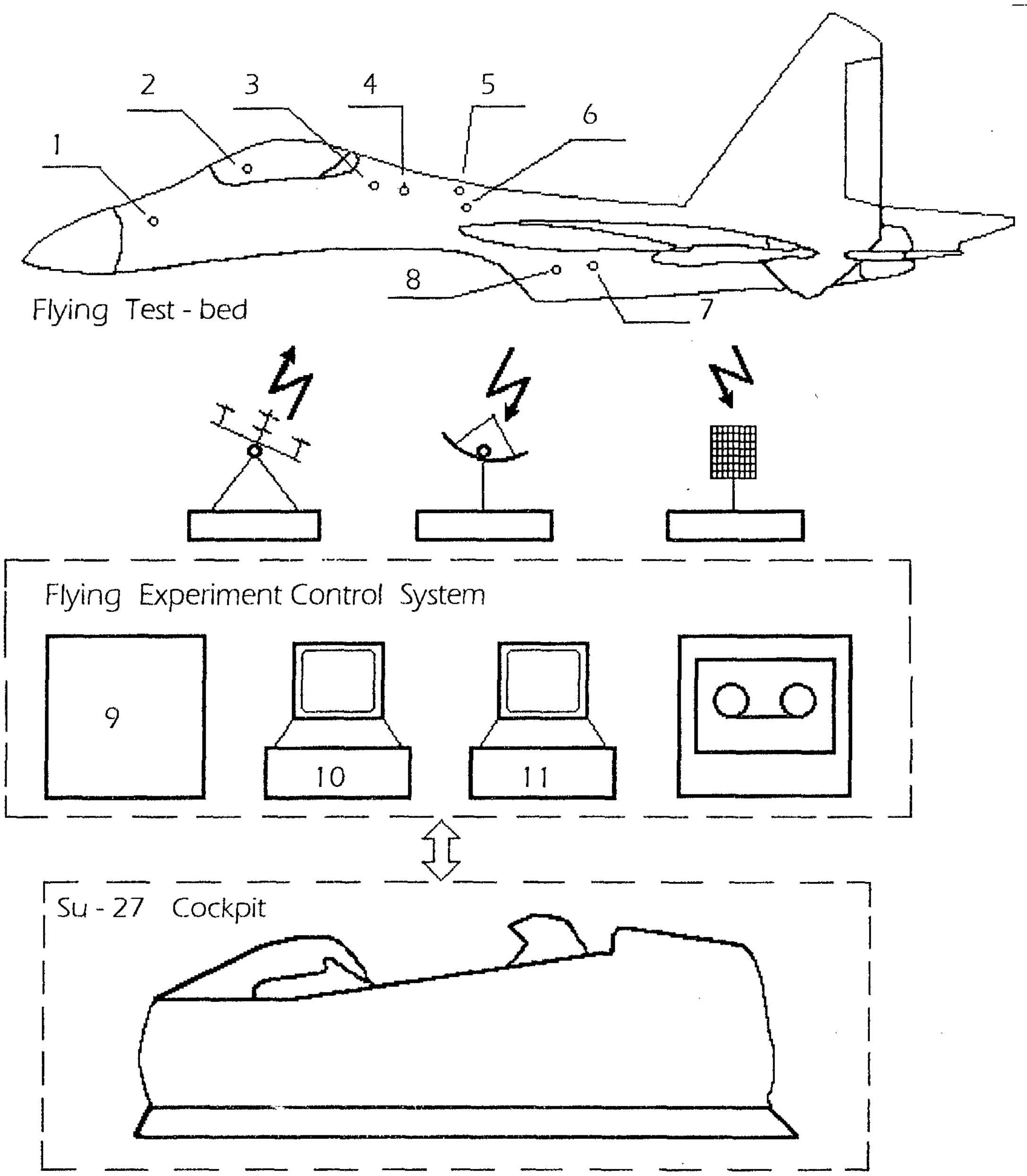
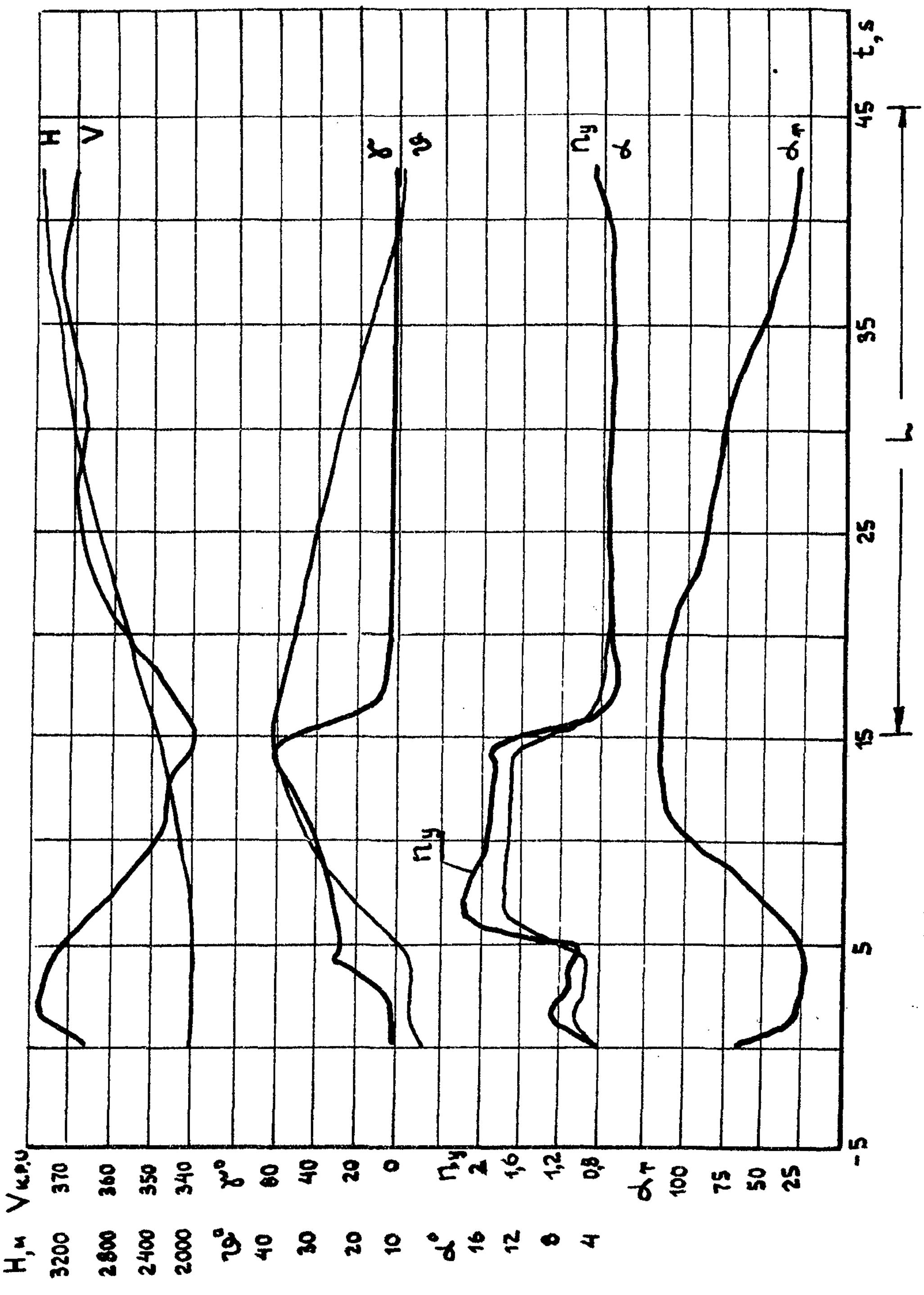
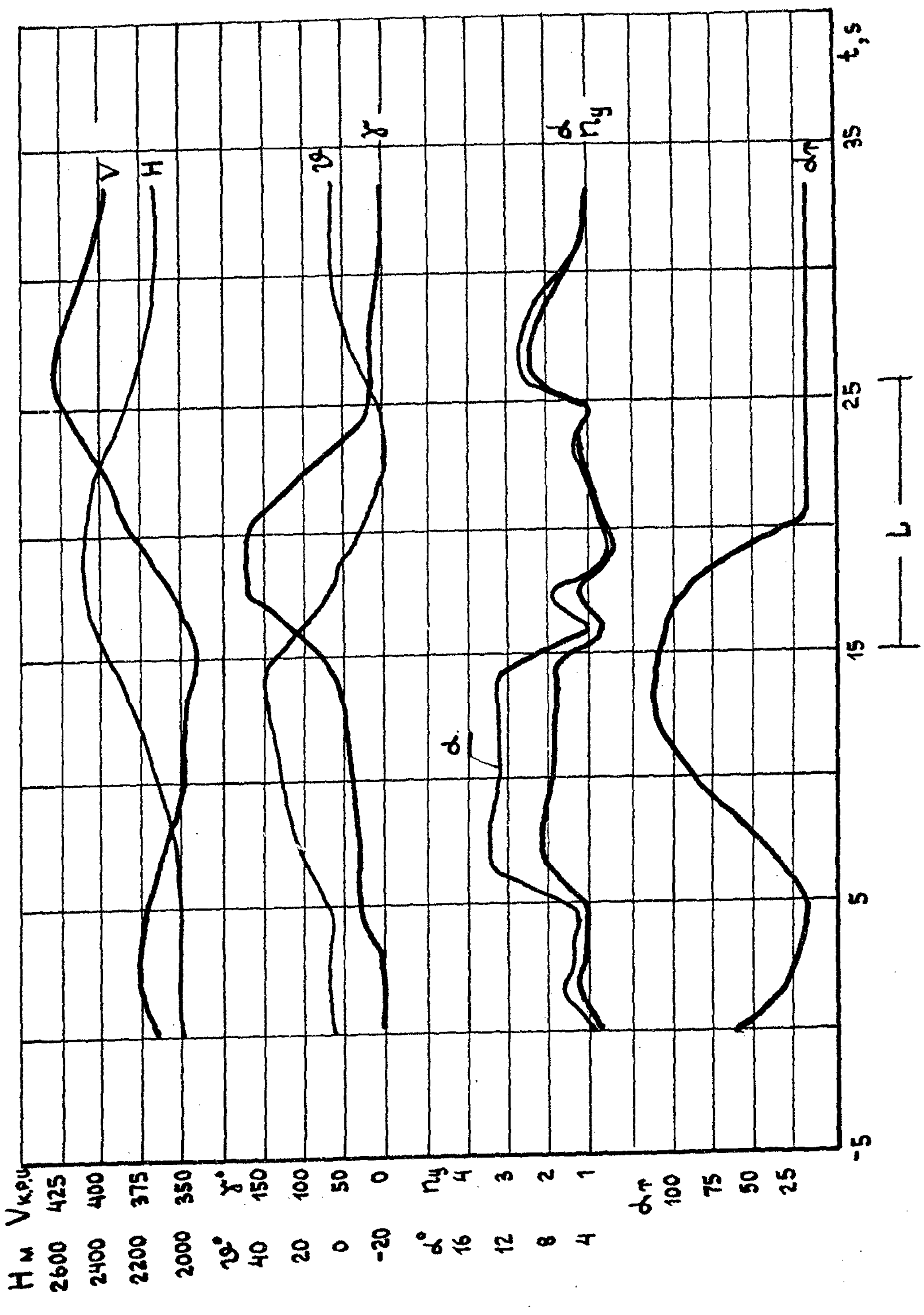


Figure 1. LMK - 2405 Complex Components

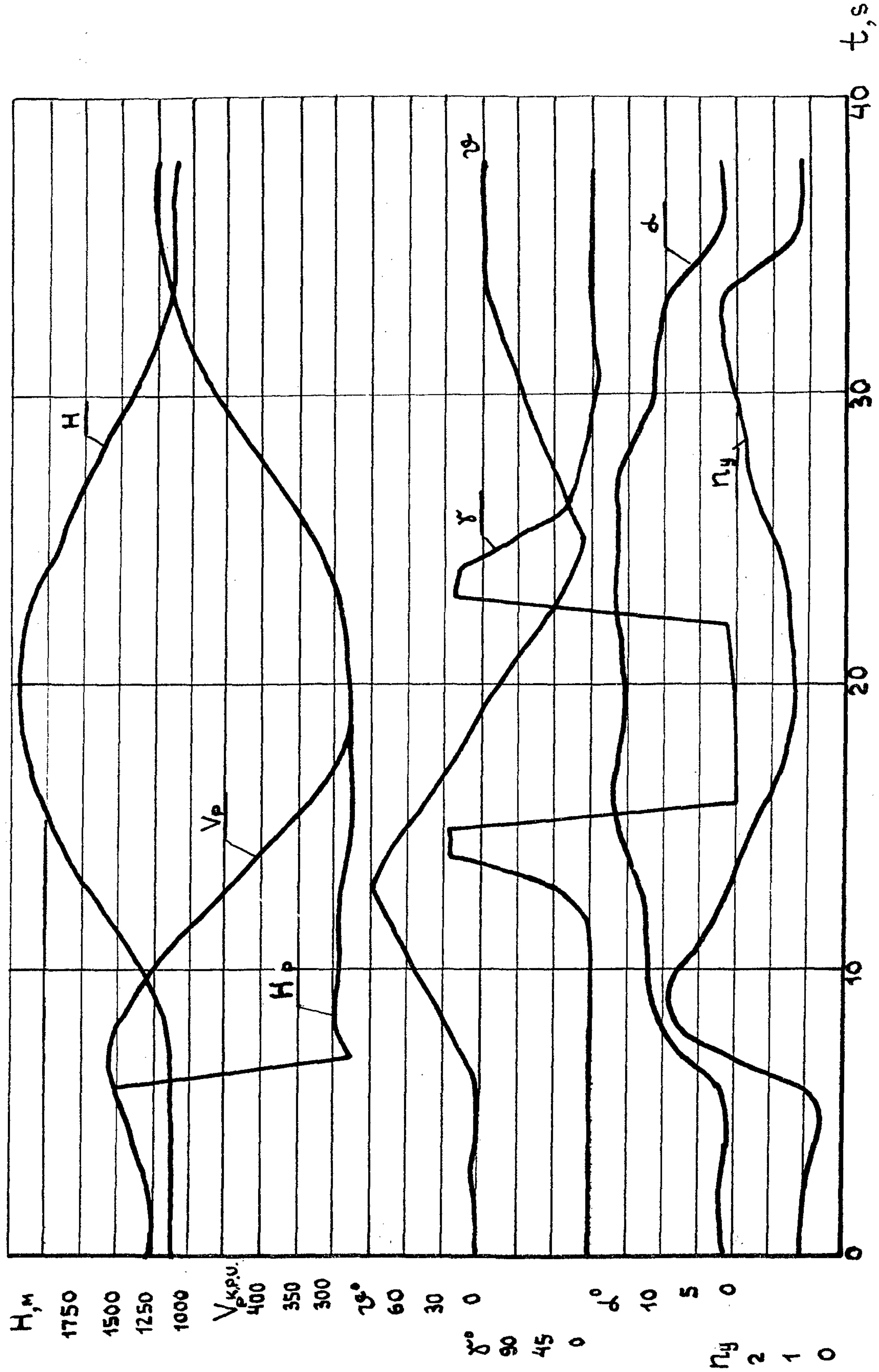
1. Television Command Link
2. Data Display System (experimental)
3. Aircraft Limitation System (experimental)
4. Automatic Control System (experimental)
5. On - board Test Instrumentation
6. Radiotelephone Station
7. Radiotelemetric Data
8. External Trajectory Measurements
9. Transmitter of Television Command Link
10. Command and Video Control Formation
11. Processing of Radiotelemetric Data and Trajectory Data, Simulation for the Ground - Test Development Stand



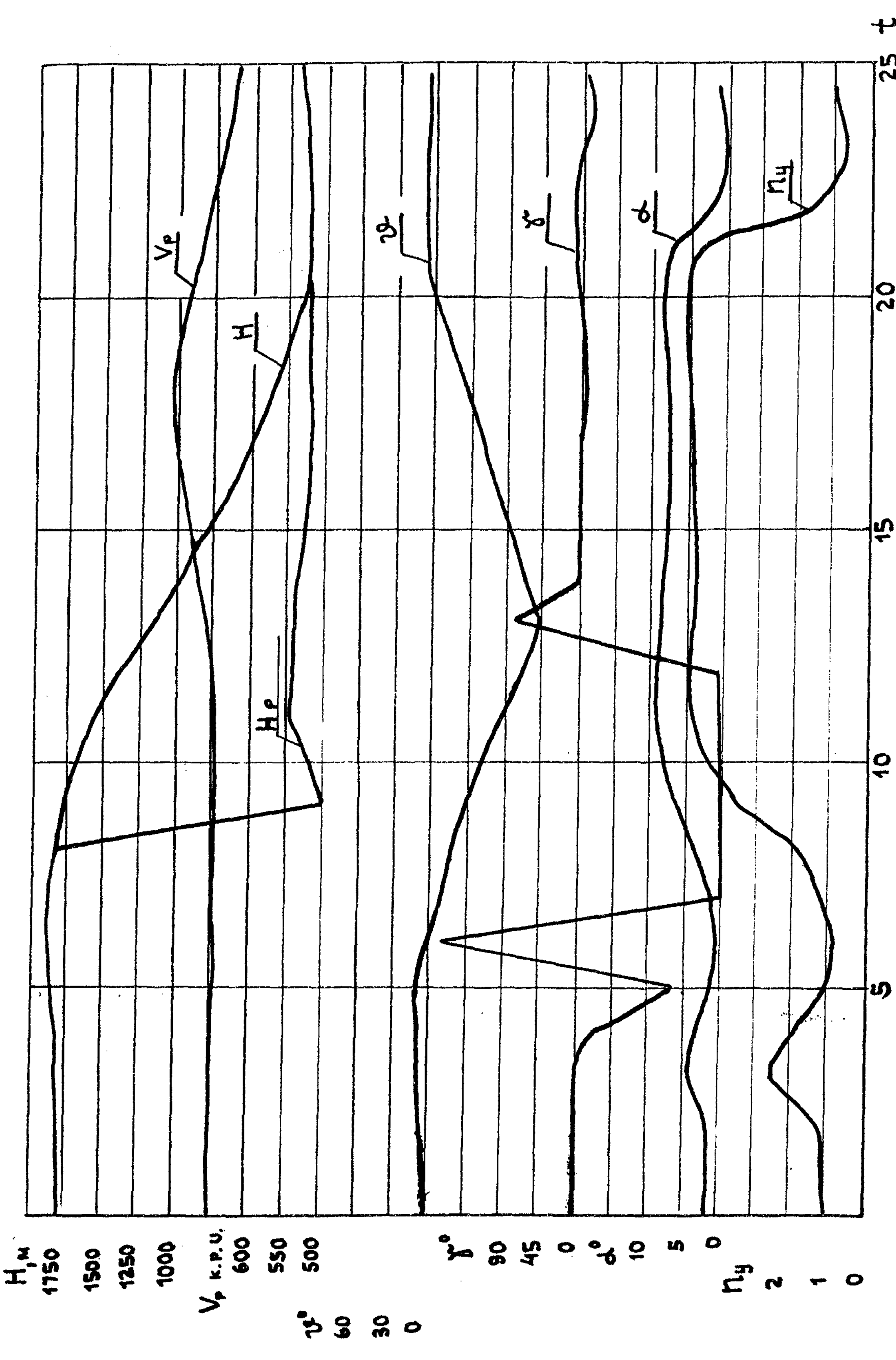
THE EXAMPLE OF USUAL LEVELING



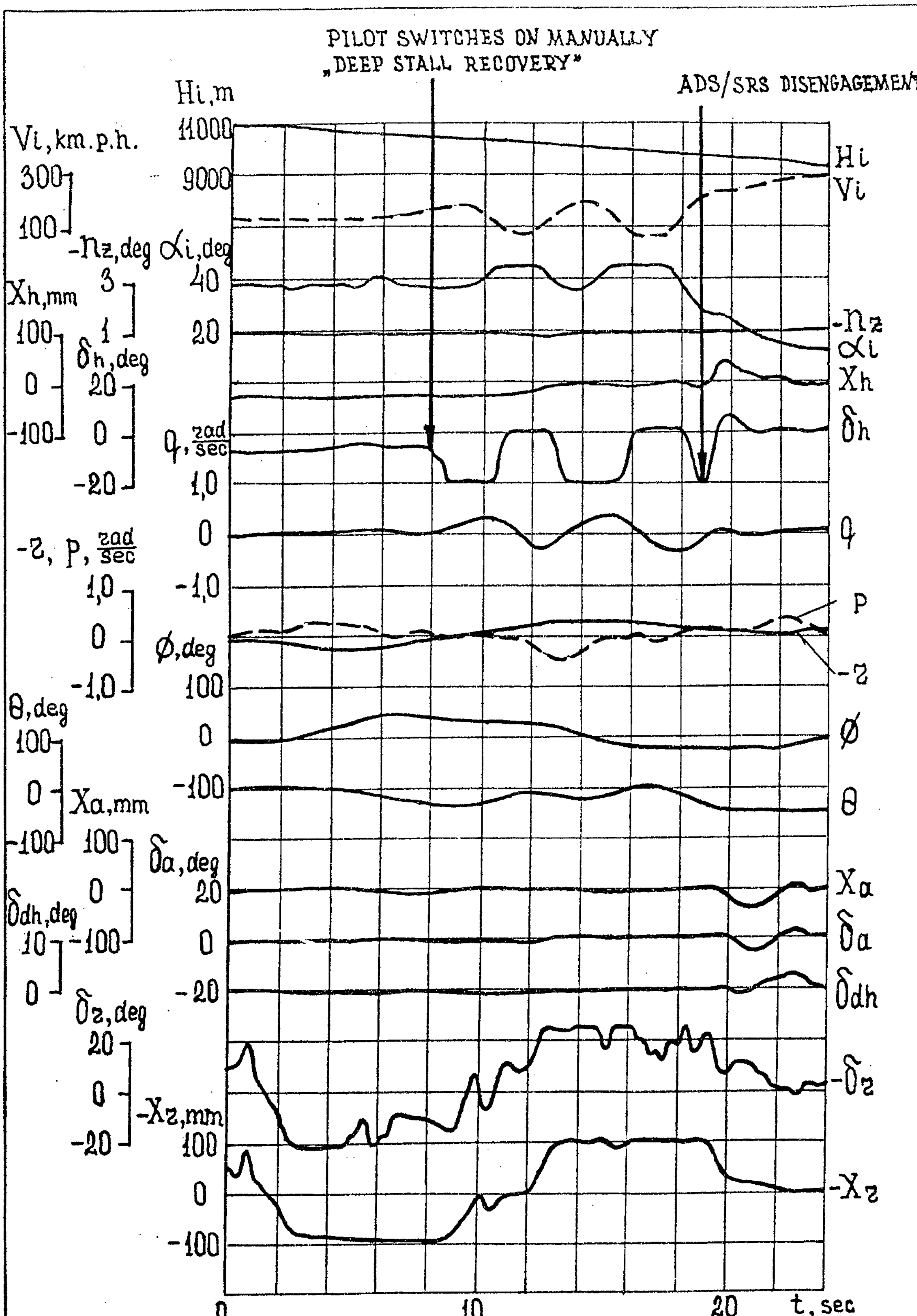
THE EXAMPLE OF MODERN LEVELING



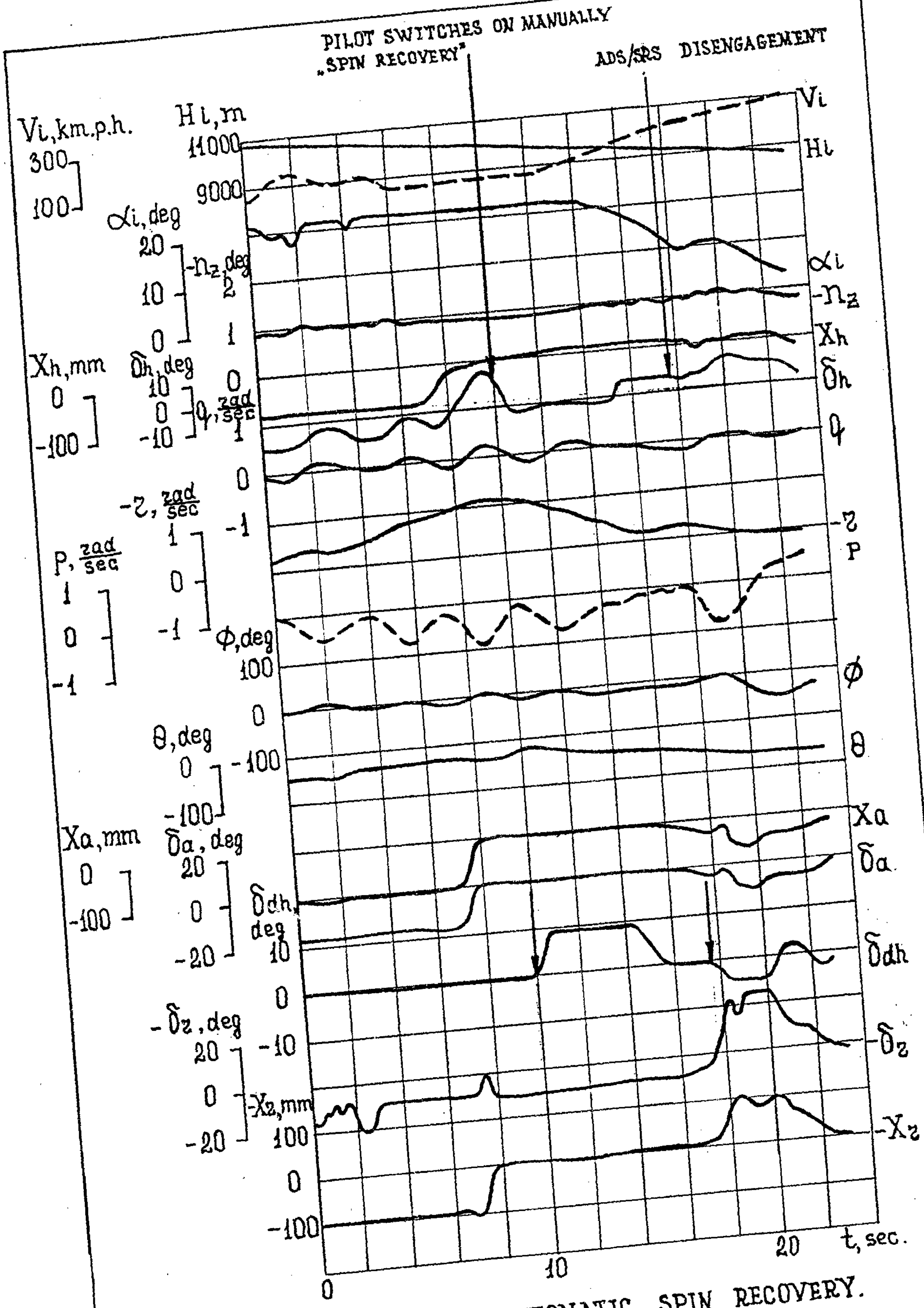
THE LOOPING MINIMUM AIRSPEED PREDICTION



THE REVERSE TURN MINIMUM ALTITUDE PREDICTION



TIME HISTORY OF AUTOMATIC DEEP STALL RECOVERY.



TIME HISTORY OF AUTOMATIC SPIN RECOVERY.

RECENT WRIGHT LABORATORY EXPERIENCE WITH FLIGHT DEMONSTRATION AIRCRAFT

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ABSTRACT

This paper will describe recent Wright Laboratory flight demonstration programs. Wright Laboratory has always had an ambitious agenda for flight demonstration of aero-technologies. To perform these demonstrations, Wright Laboratory uses both all new aircraft and modified existing aircraft. The paper will cover results and lessons on several of our recent demonstrator programs. Specifically covered will be the X-29, Short Take-off and Landing/Maneuver Technology Demonstrator, and Multi-Axis Thrust Vectoring Programs. The technologies demonstrated include vortex flow control, multi-axis thrust vectoring, flight in the post-stall region, in-flight thrust reversing, and the forward swept wing configuration. The challenges these programs presented, how problems were overcome, and the resulting successes are of interest to all aircraft developers. Finally, the author will discuss an approach to the future.

INTRODUCTION

Flight demonstrators have long been a part of aviation development. They serve at least six purposes; 1) Produce confidence in innovative concepts, 2) Accelerate the maturation of high risk technology, 3) Provide focus to research efforts, 4) Provide data based on actual flight to augment the overall data base, 5) Provide a platform for integration, and finally, 6) Bridge the vast gulf between an analytical paper design and a production airplane. Wright Laboratory has been involved in flight demonstrators since the 1930's and has gone through a long evolution of cost, risk, and technology as has most of the rest of the aviation world. The X-29, STOL/Maneuver Technology Demonstrator, and Multi-Axis Thrust Vectoring projects are three Wright Laboratory demonstrators from the "modern" era and hence of greatest application to current aviation problems.

These three projects covered a total of 11 individual technologies, but more importantly they had a large integration scope. Successful integration was, in all cases, essential to controlling cost, risk, schedule, and achieving a successful project. If there is a single conclusion from these projects, it is that if the integration fails then the project fails. For the most part, these projects were exercising the principals of Total Quality Management and Concurrent Engineering out of necessity long before these approaches received the attention that they now have.

Innovation and imagination also consistently play a role. Project support and funding is most likely if there is good balance between doing something that is achievable, yet will stretch the current paradigm or change it all together. For example, merely adapting digital flight control to a relaxed stability aircraft does not change the state of technology much and hence is less likely to become a funded project. By the same token, adapting digital flight control to an aircraft with 35% static instability is a significant, potentially valuable contribution, and at the limit of state of the art; this type of project is much more likely to be funded. Less obvious is that, generally, people in aviation want to see innovation and change and so will be more apt to go the extra mile to create and fund a project even though they have been told, "no," many times over.

X-29 PROJECT

This 18,500 lb aircraft is immediately recognizable by it's Forward Swept Wing and canard configuration as shown in Figure 1. Although the beginning of the X-29 program goes back to the late 1970's, it was still doing innovative research as late as 1992. The project was focused on the following six technologies:

1) Large (35%) Static Instability
2) Aeroelastic tailoring of the main wing
3) Three surface longitudinal control
4) Discrete variable camber main wing
5) Forward Swept Wing (30 degrees)
6) Vortex Flow Control

The objectives of this $215 Million (25% industry funded) program were to 1) Validate design options for future aircraft, 2) Examine issues related to flight at high angle-of-attack (AOA), and 3) Gather basic data on the viability of a gas nozzle to influence forebody vortices for yaw control.

To speed development, the X-29 utilized many subsystems such as the engine, landing gear, and fuselage sections from other aircraft. There were actually two X-29's built. One was used for envelope expansion at low and high speeds at "normal" AOA. This included flight up to Mach 1.48, and 50,700 feet. Low altitude, high speed flights were done at a dynamic pressure of 1500 lbs/ft*2 where the instability was such that the time for an uncorrected pitch divergence to double was 0.118 second. The other X-29 was used for high AOA and forebody vortex flow control experiments. Both aircraft flew a total of 282 flights.

The envelope expansion phase of the demonstration was focused on achieving Cooper-Harper Level I Handling Qualities with the X-29 at AOA's of under 20 degrees. The flight control system consisted of a digital triplex primary system with an analog triplex back-up. There were 254 flights (212 hours) flown in this phase of the X-29 program. Due to programmatic constraints, for the first 3 1/2 years of flight the handling qualities remained at level II. With control law and stick changes, however, level I was achieved. The fail-operate/fail-safe control system was used in a reversion mode only once in flight as a result of a hardware self-check. An uneventful return for landing was made.

The X-29 high AOA program was flown from May 1989 through Sep 1991. The objectives were to:

1) Define limits of X-29 controllability
2) Maneuver at 35 to 40 degrees AOA
3) Pitch up to 70 degrees AOA
4) Evaluate flying qualities
5) Understand flow effects

The X-29 was designed with high AOA flight in mind using aero-surfaces for control. The close coupled canard tends to keep the flow attached in the wing root area while the forward swept wings also keep the flow attached at the wing tips. In addition, the aft body strakes provide the high AOA center of pressure and control power to avoid the hung stall problem. In order to prepare for high AOA flight, the control laws were modified for AOA feedback, spin prevention, and aileron/rudder interconnect. The aircraft also had three AOA vanes and a spin chute added as well as an Inertial Navigation System for better AOA and side-slip data.

There were 120 flights flown (98 hours) in the X-29 high AOA program. Figure 2 shows the X-29 flying at high AOA with flow visualization smoke. Steady state AOA's of up to 50 degrees and dynamic angles of attack of up to 67 degrees were achieved. These flights were generally flown at under 200 knots while at high AOA. Flying qualities included good lateral control up to 40 degrees AOA, AOA dependent yaw bias between 40 and 50 degrees AOA, mild wing rock above 40 degrees AOA, and finally, no departure tendencies with pitch excursions to 67 degrees. Engine operability remained superb in all flight conditions.

The Vortex Flow Control phase of the X-29 program was flown with 60 flights from May through August 1992. The objective of this phase was to demonstrate a technique of restoring yaw control at high AOA's to address the loss of rudder effectiveness above approximately 25 degrees AOA. The specific technique was the use of gas jets to control the forebody vortex at high AOA. This demonstration included:

1) Production of yawing moments by the activation of a compressed gas fed nozzle.
2) Determination of response times.
3) Determination of modeling predictability.

There is a very large design space with regard to the type, number, and geometry of nozzles to control the forebody vortex. After some modeling and experimentation, we decided to go with two inwardly canted nozzles as shown in Figure 3. The system had to be very inexpensive so we used small ground serviced nitrogen bottles as shown in Figure 4. The system was not integrated into the flight control system, rather, the nozzles were turned on and off by a cockpit switch.

The effectiveness of the jet as AOA is increased is shown in Figure 5. These jets have the potential of providing the same yaw moment at high AOA that the rudder does at low angles of attack.

Finally, although we discussed continuing X-29 work with thrust vectoring and other projects, it was decided to end X-29 testing and move on to other demonstrator aircraft. The X-29's last flight was in August 1992.

STOL/ MANEUVER TECHNOLOGY DEMONSTRATOR (STOL/MTD)

The 40,000 lb STOL/MTD aircraft is recognizable by it's large canards added to the basic F-15. The configuration for this project also included two dimensional thrust vectoring and thrust reversing nozzles, rough field landing gear, and a quad redundant digital control system as shown in Figure 6. STOL/MTD emphasized four technologies:

1) Two dimensional rectangular thrust vectoring and reversing nozzles
2) Integrated Flight/Propulsion Control
3) Rough Field Landing Capability
4) Advanced Pilot Vehicle Interface

The objective of this $272 Million (50% industry funded) project was to 1) Demonstrate that the individual technologies were viable, and 2) Demonstrate the synergistic effect of the technologies working together. STOL/MTD flew 138 flights (174.6 hours) from Mach 1.6 at 40,000 feet to Mach 0.95 at 5,000 feet. No high AOA content was planned for this project.

This was our first experience with thrust vectoring as a flight control effector. This was achieved by integrating the pitch only vectoring capability of the nozzle (up to +/- 20 degrees at a rate of 40 deg/sec) with the flight control system's inner loop. The nozzle could provide up to 6000 lb of normal force on the rear of the aircraft regardless of airspeed or AOA. The first vectoring flight was in March of 1990. This nozzle also had thrust reversing as depicted in Figure 7.

This vectoring nozzle was designed to be a primary flight control element integrated with the aircraft's all new four channel digital "Integrated Flight/Propulsion Control". This system provides two-fail-operate with no dissimilar back-up. The were four control modes:

1) Conventional; thrust vectoring and reversing locked out.
2) Cruise; provide minimum 1g drag.
3) Combat; good attitude control and minimize drag at high g.
4) SLAND; fly at set airspeed with uncoupled attitude control.

With the vectoring nozzle augmenting pitch control and providing reverse, this system was able to achieve landing distances of approximately 1500 ft with touchdown boxes of 20 ft wide by 60 ft long. The system also held precise two point taxi attitudes at speeds as low as 40 knots.

We did experience one actual nozzle failure in transitioning between vectoring modes. This failure validated the failure accommodation of the IFPC that we had mechanized. In this case, the failed nozzle reconfigured to its fail-safe mode (full open), while the other nozzle reconfigured to conventional thrust mode allowing a routine landing.

Overall, we found thrust vectoring to be a particularly powerful control effector which can offer great design flexibility in a new aircraft, or increased capabilities in an existing aircraft. The STOL/MTD program ended in August 1991, but the aircraft itself had a good research potential and is still flying today doing research on axisymmetric pitch/yaw thrust vectoring nozzles.

MULTI-AXIS THRUST VECTORING (MATV)

The MATV program used the F-16 Variable-stability In-flight Simulator and Test Aircraft (VISTA) as a basis for demonstrating the world's first axi-symmetric multi-axis thrust vectoring nozzle. Although the program had it's roots in an aborted international effort, when reconstituted under Wright Laboratory management this program became one of the more successful aviation demonstrations in history.

This $32 Million program was 80% industry funded. Unlike the other two flight demonstrations discussed here, this project only had one technology: Axisymmetric Thrust Vectoring. Similarly, the single objective was to demonstrate axisymmetric thrust vectoring in an unlimited AOA envelope. This was done amply in 95 flights (135.7 hours) at steady state AOA's of up to 85 degrees and dynamic AOA's of +/- 180 degrees! Figure 8 shows the aircraft flying at nearly 80 degrees AOA. The airspeeds for these maneuvers ranged between 0 and 435 knots.

Because of time and funding constraints, aircraft modifications were kept to a minimum. The main modifications included the addition of the vectoring nozzle, nozzle electronic control, and spin chute as shown in Figure 9. This vectoring nozzle was not considered redundant in either control or actuation which led to a restriction that the altitude for testing be no lower that 20,000 feet for safety reasons. As shown in Figure 10 MATV utilized five control modes for the nozzle:

1) Kill; stop vector commands and center nozzle with secondary hydraulic actuation
2) Standby; system commands nozzle to center
3) Lim On; commands nozzle downward only to enforce standard

AOA limit
4) Lim Off; command nozzle for pitch rate and yaw rate with no AOA limits
5) Fail; system commands nozzle to center and requires reset

The first vectoring flight of MATV was in Sep 1993. This was our second experience with thrust vectoring, but our first experience with axisymmetric thrust vectoring (shown in Figure 11). The thrust vectoring system employed in MATV was capable of providing up to 17 degrees of vectoring at 45 degrees/second. With the engine in afterburner, this resulted in a normal force on the rear of the aircraft of 3,725 lbs regardless of airspeed or AOA. In a 28,000 lb aircraft this is a tremendous amount of control power as shown in Figure 12. Thus, given an operating engine and thrust vectoring system there is the potential of eliminating loss of control as we now know it.

Some of the maneuvers the MATV aircraft flew included the "MATV Hammerhead," a post stall loop with a rapid 200 degree pitch rotation at the top, the "helicopter," a high AOA flat descending spiral, and the "J-Turn," a rapid pitch-up to post stall followed by a yaw to nose down and then pitch pull to point 180 degrees from the original heading.

Once again we experienced virtually no engine operability problems flying at even extreme AOA and side-slip. One incident which did occur was a ruptured engine hydraulic cooler due to pressure spikes in combination with a weld defect. This resulted in a large loss of thrust due the nozzle going to nearly full open. An immediate landing was necessitated and safely accomplished.

At the end of the MATV project, in March of 1994, the thrust vectoring was removed from the aircraft so that VISTA could go on to serve in it's in-flight simulation role. This was the long-term mission for which VISTA was designed.

<u>OVERALL LESSONS LEARNED</u>

-Integration is an enabling technology. Mastery of integration is a necessity to compete in aviation right now. Of note is that Wright Laboratory has placed all flight demonstrator projects in a newly formed Integration Division.

-Use flight demonstrators as laboratories, that is, allow enough flexibility in the design to modify things easily and quickly when the unexpected problem or opportunity happens.

-Heavy use of simulation is required to reduce cost, schedule, and risk. Simulation is invaluable as part of the design, validation/ verification, and flight test process. Given the excellent cost effectiveness and fidelity, perhaps there is a larger future role for simulation in aircraft certification.

-Use small, agile teams of people. Not only does this reduce cost, but the project will achieve more in less time.

TECHNICAL LESSONS LEARNED

-Controllable flight at very high angles-of-attack is now feasible.

-Flight at high angles-of-attack did not cause significant engine operability problems in any of the aircraft.

-Thrust vectoring implemented with nozzles near the rear of an aircraft can eliminate out-of-control flight modes and dramatically influence vertical tail sizing.

-In retrofit to aircraft under approximately 50,000 lbs gross weight, thrust vectoring will add only a few hundred pounds to the aircraft and require minimal structural and control modification.

-Rudder/Vertical Fin buffet at very high AOA must be addressed in some manner as this phenomena tends to fatigue stress current tail designs.

-All three programs used various versions of in-flight selectable control law gains. The unanimous results were that the benefits are large and the risks low, provided all gain extremes are checked out on the simulator prior to flight.

THE FUTURE

There is always some new frontier in aviation, it just takes a while to realize what that frontier is. In my opinion the next great aviation challenge is absolutely affordability. Performance is at the point of diminishing returns, while optimization for efficiency has pushed the cost of aircraft into a bracket that promises stagnation if the trend continues. Stagnation meaning loss of creative critical mass because of declining acquisitions. In the climate of diminishing industry and government acquisition, a modern affordable aircraft is the customers' number one need.

CONCLUSION

The X-29, STOL/MTD, and MATV flight demonstration programs have been very successful in maturing various technologies. In particular, thrust vectoring, high AOA flight, and integrated flight control technologies have been advanced. As suggested by Figures 13 and 14, we may be headed for an era where each individual program is designed to demonstrate only one technology. In some cases this will mean shorter and less costly demonstration programs as in the case of MATV. Intense and focused programs may be perceived to be a more productive technology investment. Another potential trend is shown by Figure 15. As the total size of the programs shrink, the amount of industry participation appears to be increasing greatly.

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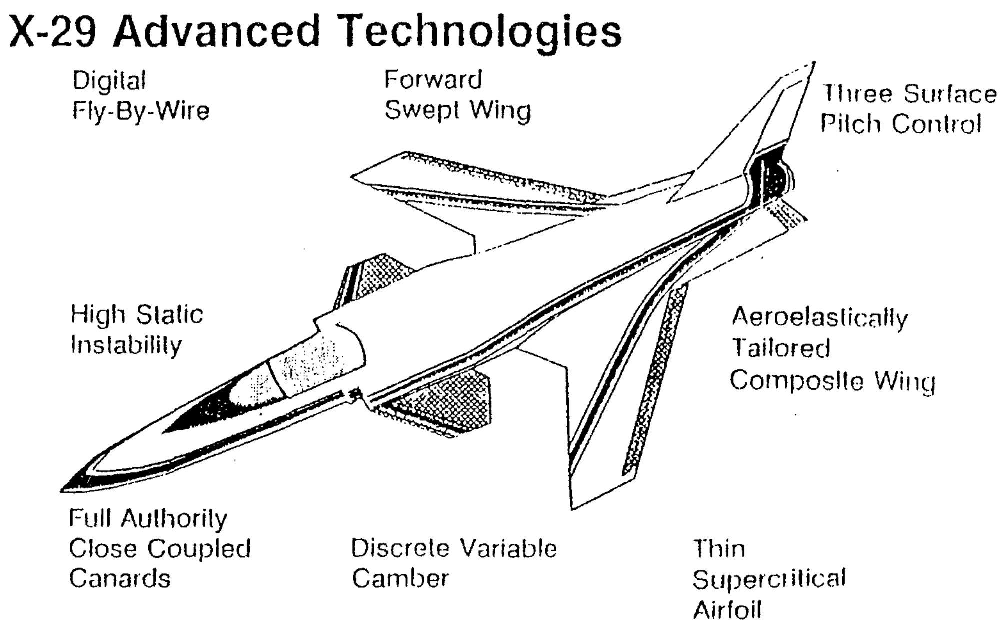

Figure 1, X-29 Technologies

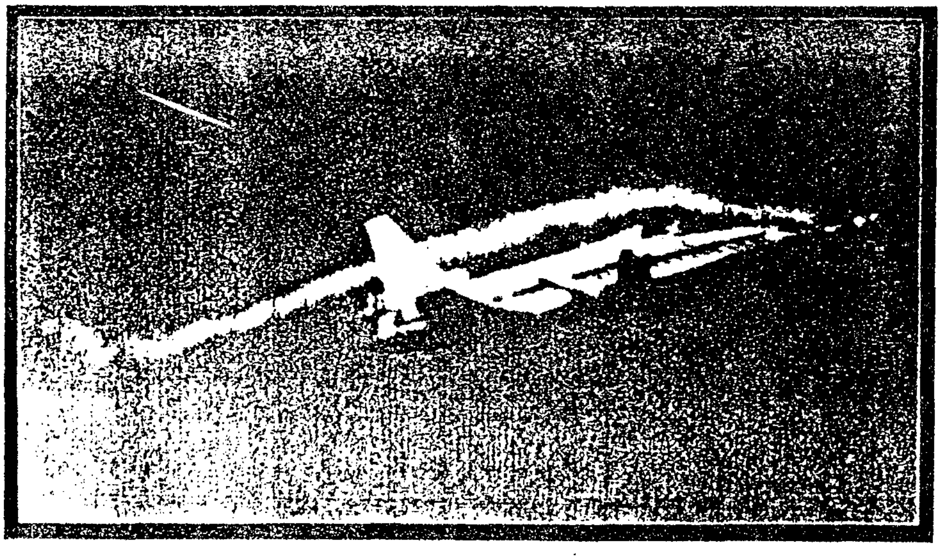

Figure 2, X-29 High AOA Flow Visualization

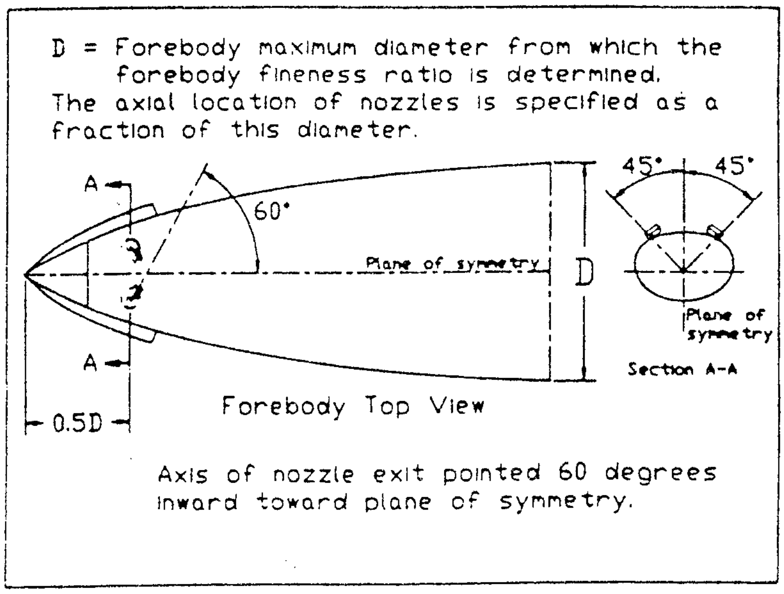

Figure 3, X-29 Forebody Nozzle Orientation

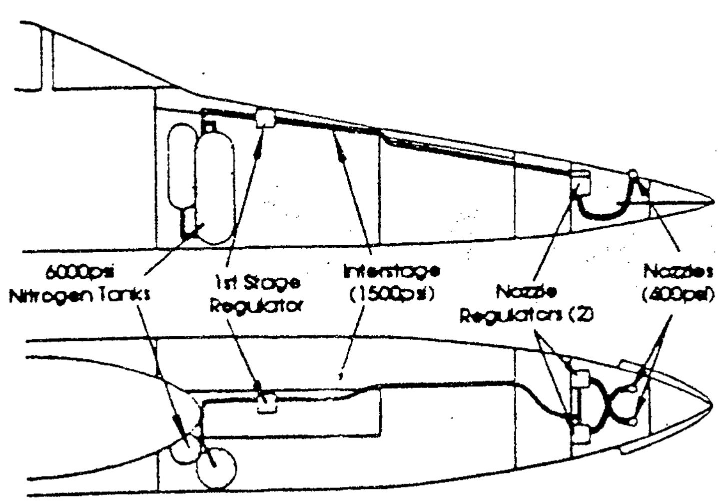

Figure 4, Vortex Control System Components

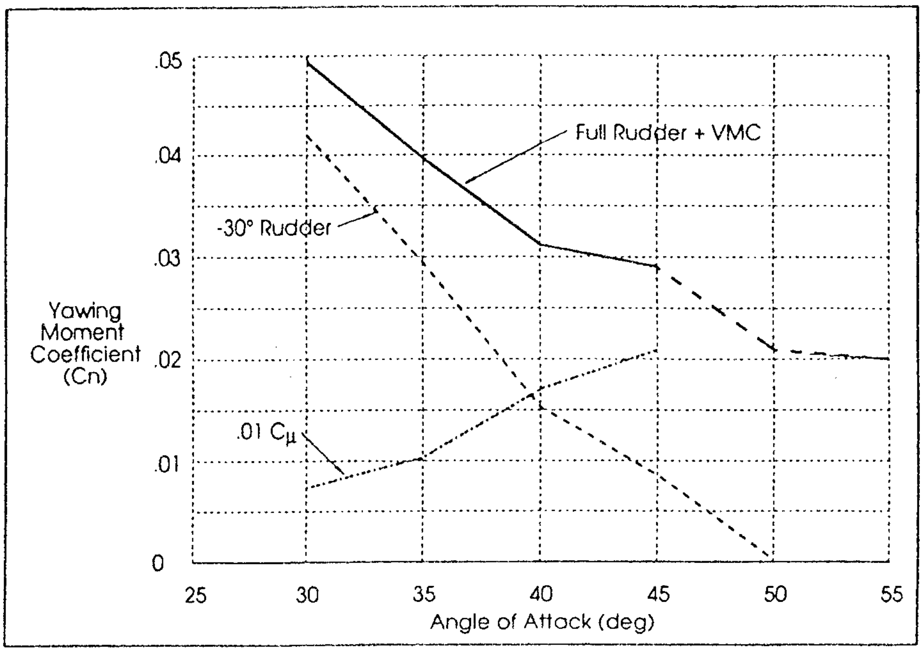

Figure 5, Vortex and Rudder Control Power

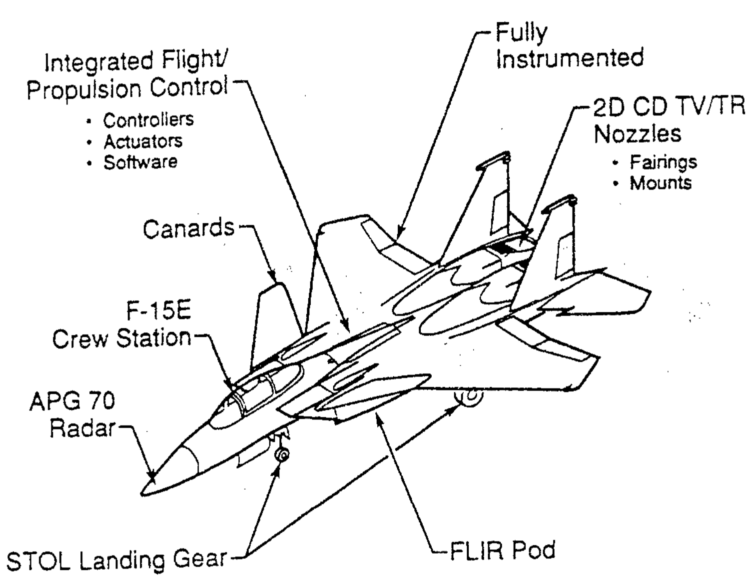

Figure 6, STOL/MTD Configuration

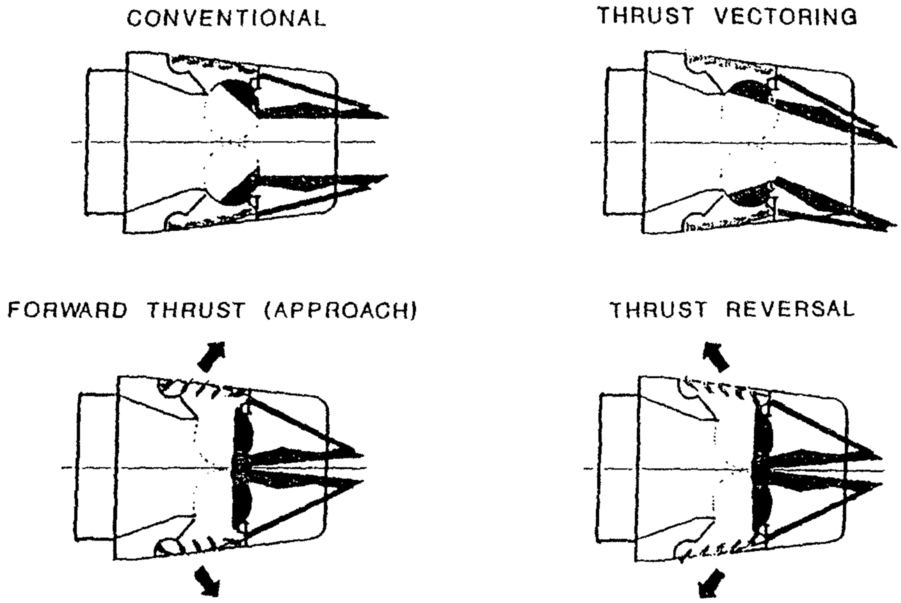

Figure 7, STOL/MTD Nozzle Modes

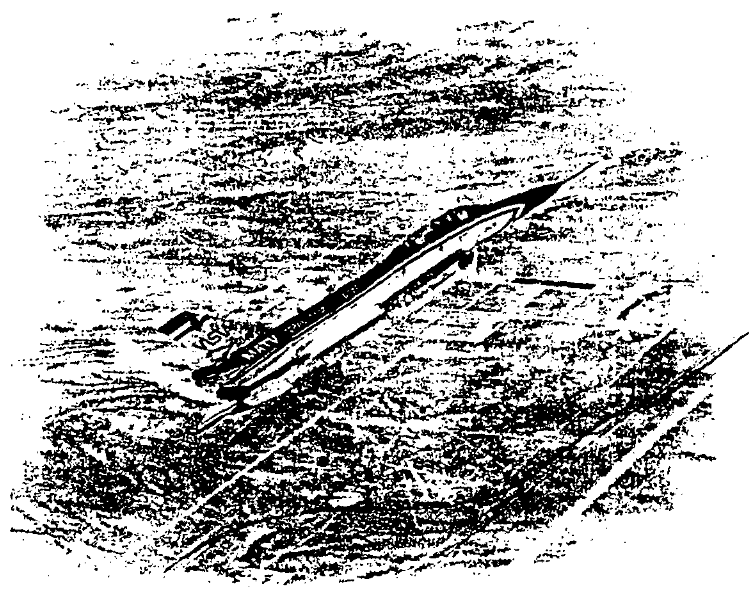

Figure 8, MATV At Approximately 80 Degrees AOA

Figure 9, MATV Configuration

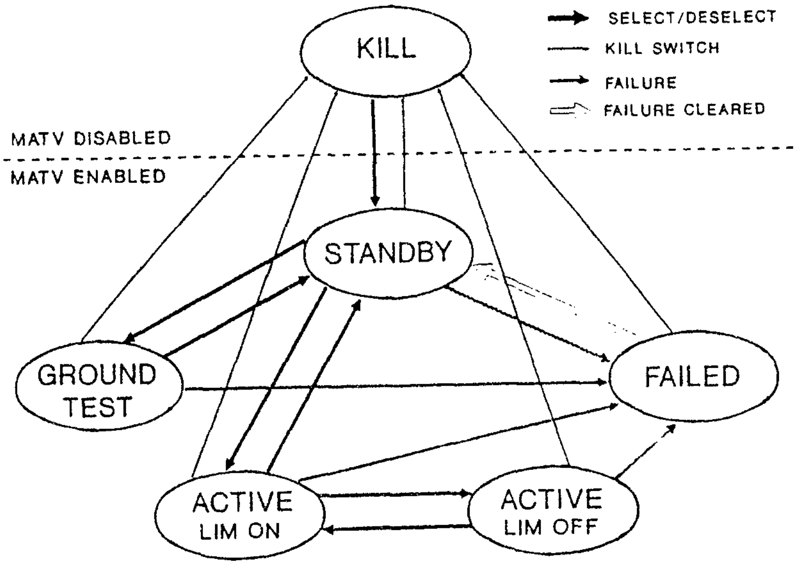

Figure 10, MATV Vector Control Modes

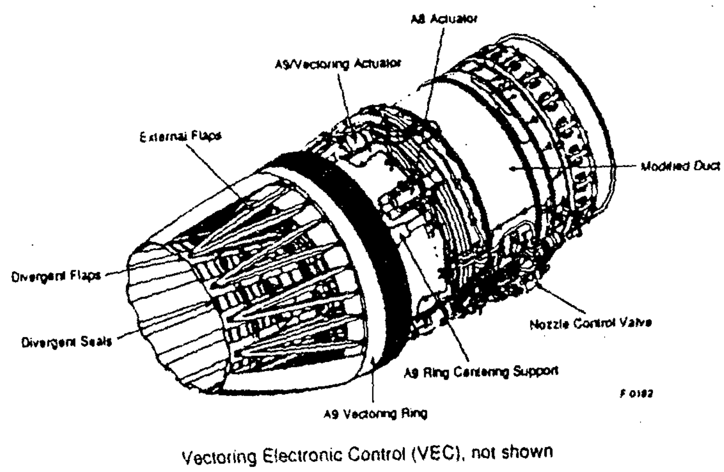

Figure 11, MATV Axisymmetric Nozzle

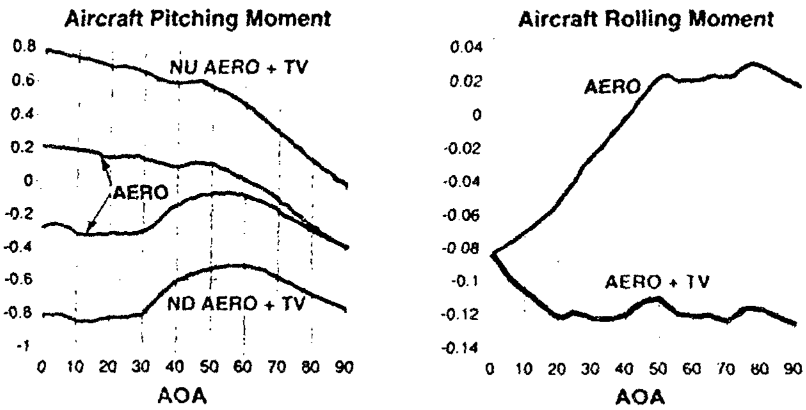

Figure 12, Vectoring Control Power

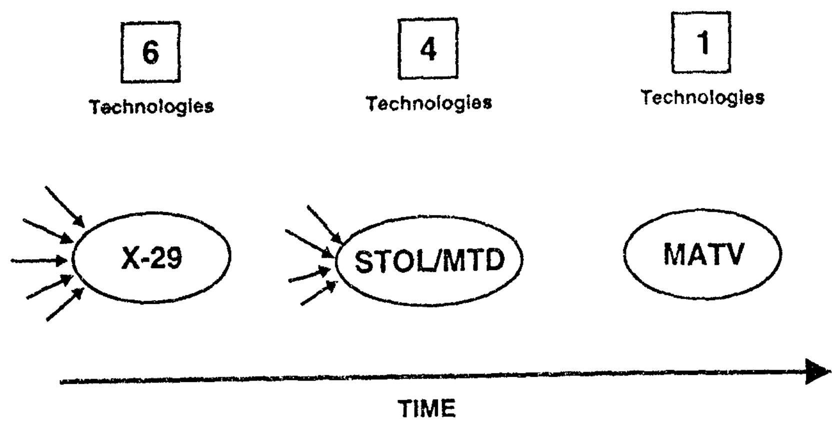

Figure 13, Number of Technologies Per Demo

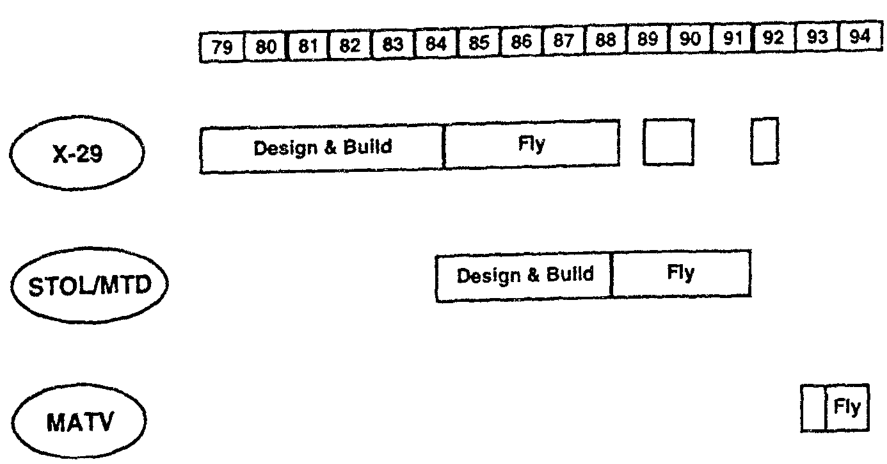

Figure 14, Flight Demonstration Schedule

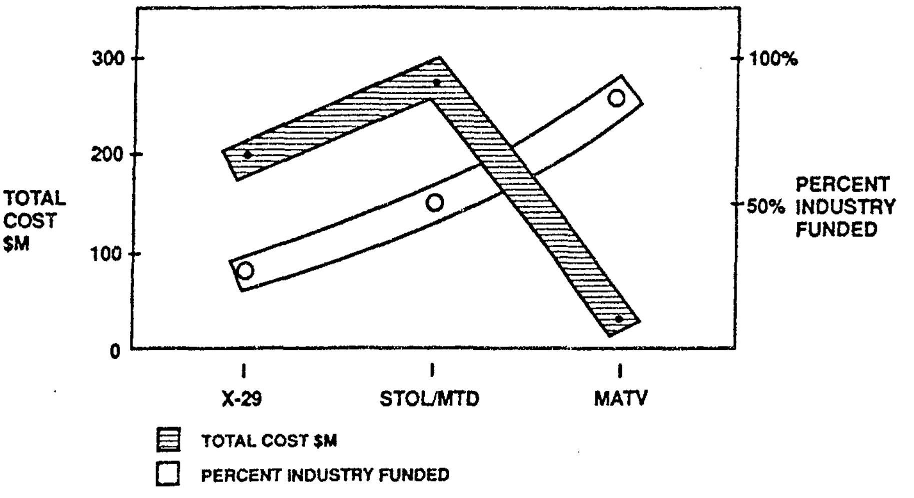

Figure 15, Demonstrator Cost and Participation

FLIGHT RESEARCH EXPERIENCE AND CAPABILITIES OF THE TU-154M IN-FLIGHT SIMULATOR FOR TRANSPORT AIRCRAFT SIMULATION

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Introduction

Flight safety and control quality of an aircraft are to a considerable extent determined by the characteristics of those systems which form up the control loop and with which the pilot is directly interacting. They include a flight-navigational data presentation system, controllers and control system which together with the airframe characteristics determine the dynamics of modern aircraft.

Recent improvements of the onboard equipment resulted in introducing on civil airplanes color electronic displays for flight-navigational data presentation, sidesticks and mini-controllers. The use of advanced digital control laws provided new handling characteristics of an aircraft and made it possible to automate certain control functions and to combine manual and automatic control modes. These means allow to get the most effective "man-machine" interface, though it is necessary while determining the optimum "man-machine" system characteristics to evaluate a great number of parameters as well as to take into account subjective factors of real flight. What is required is a complex evaluation of such factors as task performance, pilot workload, his fatigue, the adaptation level etc., with taking into consideration a possible variation of the pilot characteristics both in normal flight and in a failure one.

All the mentioned problems can be solved on the In-Flight Simulator (IFS) allowing a flexible and fast varying of characteristics of the control loop and aircraft dynamics in flight in order to directly compare different characteristic variants and determine the most effective combination of them.

In-Flight Simulator description

To conduct flight research of new control and dynamics concepts the Flight Research Institute has developed In-Flight Simulator based on the Tu-154M three-engine medium-range aircraft.

The left seat in the cabin is a research one. The conventional wheel and mechanical links have been removed from that seat. It is equipped with fittings for installation of the left and right sidesticks of different types and also of a center mini-wheel. Conventional panel instrumentation is replaced by a color electronic display and head-up flight-navigational data display. IFS is controlled by a digital fly-by-wire (FBW) system. All experimental systems of the control loop as well as IFS dynamics have variable characteristics.

Sidesticks of different forms which provide pitch, bank and, if necessary, yaw control have been investigated. All the stick controllers have force transducers. The sidesticks parameters (location, pivot point, force-displacement) are variable.

The sticks construction allows to change the stick from immovable one to controller requiring up to ± 5 cm displacement. A special block makes it possible to change the control sensitivity in flight. The mini-wheel provides similar functions. IFS is controlled by the left-seat pilot by means of a digital FBW system. The on-board computer has a high level language and it can be in real time reprogrammed in flight. The computer provides different aircraft and engine control laws for manual and automatic control. It is possible to change the IFS handling qualities in flight in order to evaluate and optimize them and also to simulate advanced airplane dynamics. Simpler control laws can be realized with the help of a triple-redundant analog FBW system. Besides, the onboard telemetry up and down links and the ground-based computer provide IFS real time advanced control laws. A number of auxiliary ground-based computers provide data analysis during flights. Experimental FBW systems (using both on-board and ground based computers) are connected to conventional actuators of the Tu-154M aircraft. When FBW is operated mechanical control system and the right column follow the surface position. This increases flight safety and provides smooth transition to conventional control in case of a FBW failure.

A color electronic head-down display and also head-up one are fed from the on-board digital mini-computer which generates in real time graphic imagery. A macro language specially developed is used for a quick reprogramming of the computer. This allows to present to the pilot in flight different formats of flight-navigational data to be directly compared or corrected so that the optimum variant could be determined.

Flight research results

We know about successful flights of the A-320 aircraft with a sidestick and positive experience of the sidestick research at our Institute. It was considered to carry out additional research of the sidestick together with a digital highly augmented control system in order to confirm a high degree of reliability and perfect handling quality. The main problems discussed are as follows:

— the influence of the sidestick type and location on pilot ergonomic rating;
— optimum force-displacement characteristics of the sidestick and handling qualities of the aircraft with a sidestick installed;
— pilot workload in flights including long duration ones;
— quickness of pilot adaptation to the sidestick and some other problems.

It was of special interest a possibility of flying with the left sidestick both in normal conditions and failure ones taking into account different qualification levels of pilots.

Thirty test-pilots of different flying experience (including those having long flight experience with maneuverable aircraft), qualification and age took part in the flight research of the sidestick to ensure objective results. The research was conducted in different weather conditions and flight regimes including visual and instrument approaches and landings.

In accordance with the objective data and the pilot ratings a sidestick is convenient for pitch and bank control, but adding yaw control disturbs the established control manner and makes control difficult for the pilot in emergency. Before the flights some of the pilots had been rather skeptical about the left sidestick. However the results showed that in this case reliable and effective control can be achieved for all regimes including approaches and landings.

Dynamic characteristics of the closed pilot-aircraft loop with different controllers were studied. For that purpose a quasi-random signal with the appropriate spectrum was sent to the surface actuators by the ground or on-board computer. The pilot's task was to stabilize the aircraft in these conditions. The frequency characteristics of pilot control show that the sidestick provides less time delays in the loop compared with the conventional column. The frequency characteristics of pilot in the sidestick control loop revealed his differentiating function which speaks of his prognostic ability.

It is necessary to point out that pilots (almost irrespective to their qualification level and flight experience) become quickly adapted to the sidestick controller. As a rule a 1—1,5 hour flight and 3—4 approaches were quite sufficient for the pilot to control the aircraft in normal conditions.

Flights of sufficiently long duration (3—4 hours) made manually by the airplane with the sidestick (conditions similar to the automatic control failure) showed that flying such a plane causes much less fatigue than when using the control wheel.

The mini-controller benefits are revealed to a greater extent when the controllers are integrated with the highly augmented control system. There was studied a digital control law using pitch, g-loads and other flight parameters in proportional path and integral feedbacks. Such a law provides a number of important features for the airplane dynamics which affect the method and quality of piloting:

— neutral stability of aircraft when speed is changing. All the pilots were positive in evaluating this feature noting that flying the plane became easier when the flight speed changed and that there was no need for control force trimming;
— possibility of setting rather simple algorithmic limitations to critical flight parameters (angle of attack, g-loads, maximum speed etc.). This prevents exceeding of limitations for short-period parameters even when mini-controller is fully deflected which is possible in stress conditions. According to the pilots this concept doesn't violate the usual control, but it makes piloting simpler and allows pilots to feel more confident and pay more attention to monitoring the systems state and observing the outside scene, which increases flight safety.

Of importance is also the concept of transition from the manual to automatic control mode and vise versa. On IFS there was positively evaluated a concept provided transition to the autopilot control mode when no force is applied to the sidestick and return to the manual control when the sidestick force is applied. The optimal delay times are 1.0 and 0.1 s correspondingly. The maximum values of the autopilot engagement/disengagement forces are taken equal to the sidestick breakout forces. In this case considerable reduction of pilot workload is achieved along with noticeable increase in control precision. In pilot's opinion this type of control must be the main control mode for civil airplanes.

At present there are no formal methods of optimizing graphic formats of the flight-navigational data presentation. The optimum format is determined by iteration, and for this participation of a great number of pilots and their experience are necessary as well as comparison of their evaluations of different formats. In order to compare different formats pilot ratings, their answers in the questionnaire, objective data of the control precision and ergonomic measures of the pilot workload are used.

IFS also can be used for investigation of some special problems of pilot-aircraft loop. As an example could be said some words about PIO investigations. In this work 4 PIO prediction techniques were selected for in-flight validation of their applicability for large aircraft:

— Ralph Smith's longitudinal axis PIO prediction technique;
— J. C.Gibson's flying qualities prediction technique;
— Hoh's bandwidth criterion;
— "pilot-in-the-loop" technique.

For in-flight investigation 4 dynamic configurations were selected. They included short-period dynamics and appropriate flight control system filters. Besides the Tu-154M inherent dynamics was considered as the base one for all kind of comparative evaluations during the ground and flight simulation.

Pilot's flight task during test flights was to keep pitch attitude error marker of head-up display within limits, denoting maximum permissible value of pitch attitude tracking error. These test flight regimes allows to perform frequency response analysis of all components of pilot-aircraft closed loop.

Some results of chosen PIO prediction technique implementation

— Ralph Smith's technique predicts only possible PIO frequency;
— Gibson's technique works quite well. Predicted results correspond to pilot ratings obtained during flight tests;
— Hoh's bandwidth criterion shows only general trend of pilot ratings;
— "pilot-in-the-loop" technique application for these configurations gives erroneous results.

Unsatisfactory results for some criteria can be possibly explained by the fact that they were developed for fighter type aircraft so they must be modified during flight investigations to be applicable for large transport aircraft.

Conclusion

The In-Flight Simulator with in-flight variation of controller type and characteristics, control laws, flight-navigational information presentation systems and dynamics is a highly effective mean of optimizing the pilot-aircraft system in investigating new control concepts for transport aircraft.

UNITED STATES NAVY SKI JUMP EXPERIENCE AND FUTURE APPLICATIONS

by

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SUMMARY

The United States Navy has been evaluating the performance benefits of using a ski jump during takeoff. The significant gains available with the use of Vertical and Short Takeoff and Landing (V/STOL) aircraft operating from a ski jump have been documented many times in the past; however, the U.S. Navy has expanded this concept to include Conventional Takeoff and Landing (CTOL) aircraft. This paper will present the results of a recent shipboard evaluation of the AV-8B aboard the Spanish ski jump equipped ship PRINCIPE DE ASTURIAS, and a shore based flight test evaluation of CTOL aircraft operating from a ski jump ramp. The analytical tools developed during the CTOL phase of testing are used to project the benefits which could be realized by combining the steam powered catapult and a "mini" ski jump ramp compatible with today's aircraft carriers.

NOMENCLATURE

AOA	-	Angle of Attack
CG	-	Aircraft Center of Gravity
CRAT	-	Catapult/Ramp Assisted Takeoff
CTOL	-	Conventional Takeoff and Landing
MIL	-	Military Thrust
Max A/B	-	Maximum Afterburner Thrust
ROC	-	Rate of Climb
STO	-	Short Takeoff
SLW	-	Short Lift Wet
V	-	Ramp Exit Airspeed (KEAS)
V_e	-	Ramp Exit Speed (kt)
VTOL	-	Vertical Takeoff and Landing
V/STOL	-	Vertical and Short Takeoff and Landing
W	-	Aircraft Gross Weight (lb)
W_h	-	Hover Weight (lb)
W/W_h	-	Hover Weight Ratio
WOD	-	Wind Over Deck

AV-8B SKI JUMP

Introduction

Flight tests were conducted aboard PRINCIPE DE ASTURIAS, a Spanish ship designed for Harrier operations with a 12 degree ski jump ramp, December 1988 to define operating procedures and limitations and document performance gains over conventional flat deck short takeoffs (STO's). A total of 89 STO's were conducted. PRINCIPE DE ASTURIAS proved to be an excellent platform for Harrier operations. The flight test program clearly demonstrated the performance gains, reduced pilot workload, and improved safety inherent in a ski jump assisted shipboard takeoff. WOD requirements were approximately 30 kt less than flat deck requirements, resulting in significant fuel savings and flight operations having less impact on ship's heading and speed. Deck run requirements were approximately 350 ft (107 m) less than flat deck requirements, improving Harrier/helicopter interoperability. Maximum payload capability for a ski jump assisted launch is up to 53% greater than flat deck capability, allowing shipboard Harrier operations to the same takeoff gross weight as shore based. The heaviest Harrier to be launched from a ship to date was accomplished during the test program (31,000 lb). The ski jump launch always produced a positive rate of climb at ramp exit, the resulting altitude gain allowing aircrew more time to evaluate and react to an emergency situation. Pilot opinion is that the ski jump launch is the easiest and most comfortable way to takeoff in a Harrier.

Background

In the mid-1970's the British aerospace community identified the significant improvements in takeoff performance for vectored thrust aircraft obtained with the assistance of an upwardly inclined (ski jump) ramp and, as a result, incorporated ramps on existing Royal Navy carriers. In 1977, the

Spanish Navy began construction of the first ship designed from the keel up to support Harrier operations. The basic ship design was modeled after the U.S. Navy sea control ship promoted by Admiral Zumwalt in the mid-1970's. A 12 degree ski jump ramp was incorporated to improve takeoff performance. Based on previous shore based ski jump testing and simulation efforts, a 12 degree ramp was found optimum for maximizing takeoff performance while maintaining aircraft structural loads within limits. The ramp profile is the same as that of HMS HERMES of the Royal Navy. Construction began in 1977 at the El Ferrol shipyard of Empresa Bazan Nacional. The ship was commissioned PRINCIPE DE ASTURIAS and delivered to the Spanish Navy 30 May 1988. Shortly thereafter, the Spanish Navy made an agreement with Naval Air Systems Command for the then Naval Air Test Center at Patuxent River, Maryland to conduct flight tests and engineering analysis required to publish an operating bulletin for AV-8B operations from the ship. Flight test objectives were to define operating procedures and limitations and document performance gains over conventional flat deck STO's.

Test Assets

Ship

PRINCIPE DE ASTURIAS can accommodate up to 36 aircraft consisting of both Harriers and helicopters. The flight deck is approximately 575 ft (175 m) long by 95 ft (29.0 m) wide. The ski jump ramp coordinates are presented in table 1. The maximum STO deck run length is 550 ft (168 m). The ship is stabilized in roll with four stabilizers. The ship has six VTOL spots. The flight deck including flight deck markings is illustrated in figure 1. A profile of the ship is presented in figure 2. The ship is equipped with SPN-35 radar for ground controlled approach, Harrier Approach Path Indicator (HAPI) and Deck Approach Projector Sight (DAPS) for glide slope information, and Hover Position Indicator (HPI) for height control. The ship has a 7,500 nautical mile range at 20 kt ship speed. The ship has a maximum speed of approximately 25 kt.

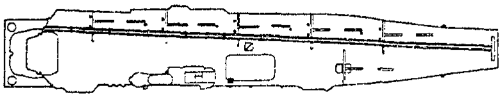


Figure 1

PRINCIPE DE ASTURIAS Flight Deck

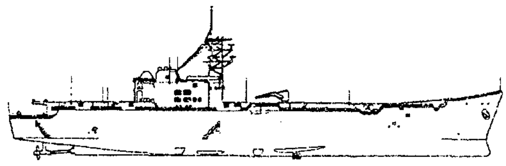


Figure 2
PRINCIPE DE ASTURIAS Profile

Table 1
Ski Jump Ramp Coordinates

Distance Along Ramp ft	(m)	Ramp Height ft	(m)
0.0	(0.0)	0.00	(0.00)
11.5	(3.5)	0.20	(0.06)
21.3	(6.5)	0.50	(0.15)
31.2	(9.5)	0.88	(0.27)
41.0	(12.5)	1.36	(0.41)
50.9	(15.5)	2.00	(0.61)
60.7	(18.5)	2.74	(0.84)
70.5	(21.5)	3.66	(1.12)
80.4	(24.5)	4.69	(1.43)
90.2	(27.5)	5.89	(1.80)
100.1	(30.5)	7.23	(2.20)
111.6	(34.0)	9.02	(2.75)
121.4	(37.0)	10.69	(3.26)
131.2	(40.0)	12.56	(3.83)
141.1	(43.0)	14.55	(4.43)
151.6	(46.2)	14.94	(4.55)

Test Aircraft

The AV-8B is a single place, single engine, tactical attack, vectored thrust, jet V/STOL aircraft built by McDonnell Aircraft Company (MCAIR). The aircraft has a shoulder mounted supercritical wing, four rotatable engine exhaust nozzles, and a lift improvement device system. The aircraft is powered by a Rolls Royce PEGASUS F-402-406A twin spool, axial flow, turbofan engine with an uninstalled sea level static short lift wet thrust rating of 21,500 lb (95,600 N). The primary flight controls consist of aerodynamic and reaction controls which are interlinked in all axes and hydraulically powered. The AV-8B is an excellent aircraft for ski jump takeoff due to its exceptional low-speed flying qualities. A three view drawing of the AV-8B is presented in figure 3.

Two aircraft were used for shipboard testing: a preproduction AV-8B which was instrumented for flying qualities and performance testing and nose and main landing gear strut positions, and a non-instrumented production AV-8B. Both aircraft were representative of production EAV-8B aircraft for the purpose of these tests.

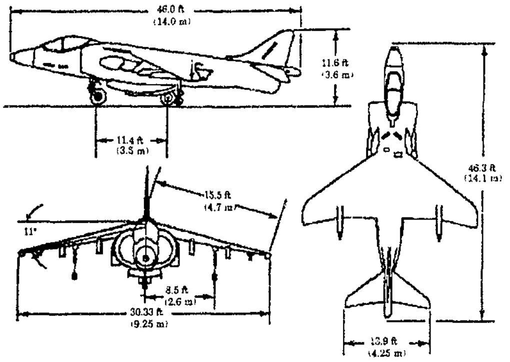


Figure 3
AV-8B Three View Drawing

Shipboard Tests

STO Launch Technique

A typical STO launch profile is illustrated in figure 4. Nozzles are positioned to 10 deg below fully aft for the deck run to reduce vibratory loads on the flaps and stabilator. The launch begins with application of full power with brake release as the tires begin to skid. The stick is guarded in the preset trim position throughout the deck run and nozzle rotation. As the aircraft exits the ramp, the pilot positions the nozzle lever to the preset STO stop. Ramp exit cues are both visual (nozzle rotation line) and physical (decrease in load factor as the aircraft leaves the ramp). After ramp exit, the pilot task is to maintain the aircraft pitch attitude achieved at ramp exit (approximately 18.5 deg) and monitor angle of attack (AOA). If AOA reaches 15 deg during the trajectory, the pilot decreases the aircraft pitch attitude as required to maintain AOA at or below 15 deg. Immediately after ramp exit, the velocity vector indicates a climb due to the upward velocity imparted by the ramp. This initial rate of climb is not a true indication of aircraft performance, and decreases to a minimum at an inflection point prior to the aircraft achieving a normal semi-jetborne climb. Prior to the inflection point, the aircraft normal acceleration is less than 1 g. The aircraft has a positive rate of climb due to the ramp induced vertical velocity, but rate of climb is decreasing due to insufficient lift. At the inflection point, the aircraft has accelerated to an airspeed at which aircraft normal acceleration is 1 g (lift=weight), and rate of climb is no longer decreasing. After the inflection point is reached, the aircraft begins a normal semi-jetborne climb (normal acceleration greater than 1 g), and rate of climb increases. At this point, the pilot gradually vectors the nozzles aft and accelerates to wingborne flight.

STO Ramp Exit Speed

STO ramp exit speed must be accurately predicted to ensure ramp exit airspeed required is obtained and landing gear structural limits are not exceeded. Ramp exit speed is a function of aircraft hover weight ratio and deck run. Tests were conducted at deck runs from 200 to 550 ft (61 to 168 m). Actual Ramp exit speeds were obtained from infrared trips which were mounted at the end of the ramp. Ramp exit speed data was reduced to an exit speed parameter and plotted against deck run. The exit speed parameter is defined as $V^2(W/W_h)$ and

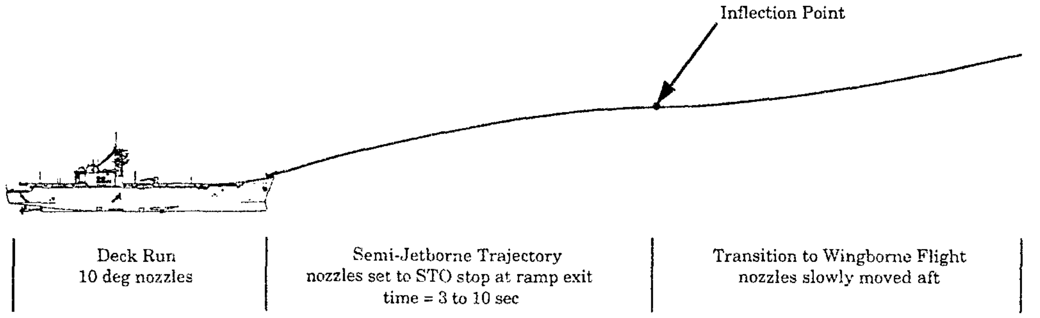


Figure 4
STO Launch Profile

its relationship to deck run is based on the dynamic relationship $V^2=2aS$ where "a" is the average acceleration and "S" is the deck run. STO ramp exit speed averaged one kt less than that of an identical flat deck launch due to the decelerating effects of the ramp. Ramp exit speed was predictable within 2.5 kt.

STO Landing Gear Structural Limits

During ski jump launch with no ship motion, loads are imparted on the landing gear due to aircraft gross weight, aerodynamic lift, vectored engine thrust, pitching moments, and inertial forces including centrifugal forces. Centrifugal forces are influenced by aircraft velocity and local ramp curvature. The primary dynamic response exhibited by the AV-8B during ski jump launch is in the aircraft heave mode. Dynamic response to aircraft pitch motion is small in comparison to heave.

STO maximum ramp exit speeds for landing gear structural limits were determined at gross weights of 26,000, 28,000, and 31,000 lb (11,793, 12,701, and 14,062 kg). Fatigue strength for 1,500 lifetime ski jump launches defined the limiting criteria for landing gear based on MCAIR analysis. Nose and main landing gear strut positions were instrumented and monitored real-time. Simulation data and previous ski jump testing indicated outrigger landing loads would not approach limiting criteria and were therefore not instrumented. Target ramp exit speed for the first launch at each gross weight was based on MCAIR simulation and was at least 10 kt below the predicted landing gear limit. The ramp exit speeds for successive launches were increased in increments of approximately three to five kt by increasing deck run until the limiting criteria were reached. A method suggested by MCAIR was used to account for ship motion. Load factor trends were incremented for sea state resulting in a shift in the aircraft gross weight vs. maximum ramp exit velocity curve for given sea states. MCAIR correlated ship motion with sea state based on ship motion studies of similar type ships by David Taylor Ship Research and Development Center. Worst case phasing of ship's pitch, heave, and coriolis effects were used to determine load factor increments due to sea state. The coriolis effect is the additional normal acceleration of the aircraft due to its increased velocity normal to the deck while it travels away from the ship's pitch center. Analytical results were verified with test data and are presented in figure 5.

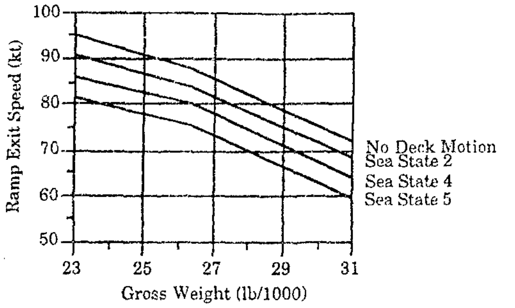


Figure 5
Landing Gear Structural Limits

STO Minimum Ramp Exit Airspeed

STO minimum ramp exit airspeed tests were conducted at hover weight ratios of 1.43, 1.52, and 1.60. The purpose of these tests was to define the minimum ramp exit airspeed required for a safe launch and to evaluate the sensitivity of reducing ramp exit airspeed when operating near the minimum. The minimum airspeed was approached by holding hover weight ratio constant while decreasing ramp exit airspeed for each successive launch. Ramp exit airspeed for the first launch at each hover weight ratio was based on MCAIR simulation and previous ski jump testing and targeted an airspeed approximately 15 kt above the predicted minimum. The ramp exit airspeeds for successive launches were reduced in decrements of approximately three to five kt by varying either deck run or WOD until the minimum ramp exit airspeed was reached. The limiting factor for ramp exit airspeed was zero rate of climb at the inflection point. Test results are presented in figure 6. Flying qualities at minimum ramp exit airspeeds were satisfactory. AOA was controllable with a maximum transient AOA of 17 deg. Lateral control was acceptable throughout the STO envelope. Longitudinal acceleration was acceptable for all launches, averaging two to four kt/sec for launches with rate of climb from 200 to 1,000 ft/min (61 to 305 m/min). The minimum longitudinal acceleration achieved during the test program was 1.5 kt/sec.

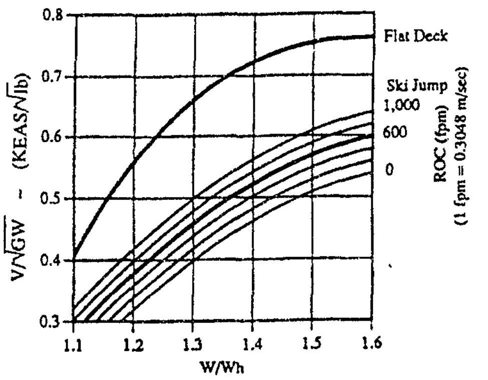


Figure 6
Takeoff Performance

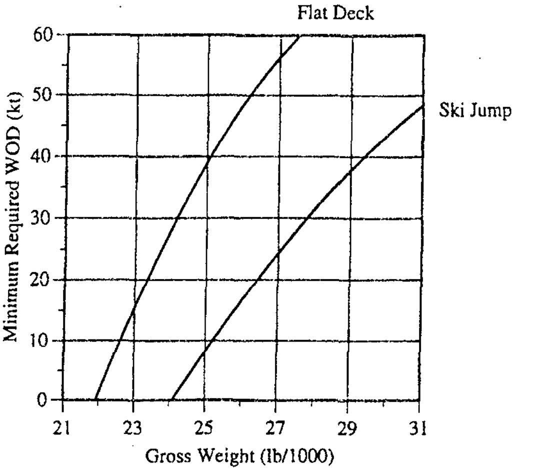


Figure 7
STO WOD Requirements
300 ft Deck Run
Standard Day, Nominal SLW Engine

Ski Jump/Flat Deck Comparison

Increased performance obtained from a ski jump assisted launch is realized through reduced WOD and/or deck run requirements and/or increased launch gross weight capability. The discussion in this section deals with the performance gains realized with the PRINCIPE DE ASTURIAS ramp. Performance gains obtained from different ramps will vary with ramp exit angle.

Ski jump launch WOD requirements are compared with flat deck requirements in figure 7. Required WOD for a ski jump assisted launch is approximately 30 kt less than a flat deck launch. Ski jump launch operations are therefore not as dependent on natural winds for launch. As a result, normal launch operations do not dictate ship's heading, allowing the ship maneuvering flexibility and decreased operating area during flight operations. Reduced WOD requirements can be appreciated in fuel savings, as the ship can steam at the speed required for minimum steerage and not dictate ship's heading, allowing the ship maneuvering flexibility and decreased operating area during flight operations and still have the required WOD for normal launch operations. Reducing ship's speed from 25 to 7 kt decreases fuel consumption by approximately 80%.

Deck run requirements for ski jump launch are compared with flat deck requirements in figure 8. Instead of launching at lower WOD, ski jump launches can be conducted at the same WOD required for flat deck launches while reducing the deck run by approximately 350 ft (107 m). The result is improved interoperability between Harriers and helicopters. On flat deck ships, if a Harrier is to launch with a significant payload then the entire flight deck is often required for the deck run. This makes Harrier/helicopter interoperability extremely difficult. By reducing the required deck run with the assistance of a ski jump, Harriers can conduct takeoff and landing operations from the forward flight deck while helicopters operate concurrently and completely independently from the aft section. Vertical landing operations to the forward deck spots provide excellent visual cues due to the ramp height, offering significant improvement over vertical landing operations to forward deck spots on a flat deck ship. The ability to operate Harriers and helicopters at the same time from the same flight deck greatly enhances the efficiency of the amphibious assault force.

Gross weight capability for a ski jump launch is compared with flat deck capability in figure 9. For a given WOD and deck run, an AV-8B can carry 3,000 to 5,900 lb more payload from a ski jump ship than from a flat deck ship. This equates to up to a 53% increase in takeoff payload capability. When operating from flat deck ships in tropical day conditions, AV-8B aircraft mission payload is limited by takeoff performance, which is not the case for operations from a ski jump ship. The efficiency of the close air support mission is therefore enhanced by a ski jump assisted launch by allowing more payload per sortie.

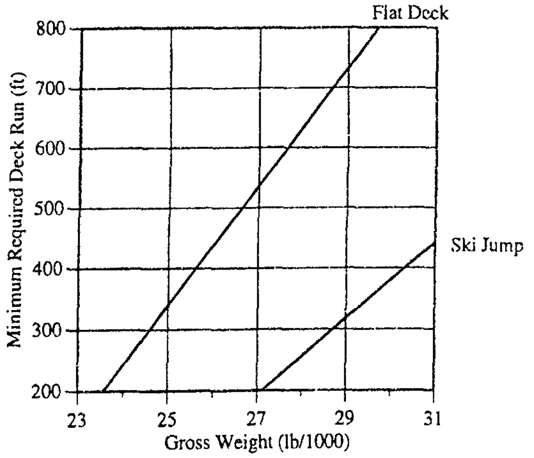


Figure 8
STO Deck Run Requirements
35 kt WOD
Standard Day, Nominal SLW Engine

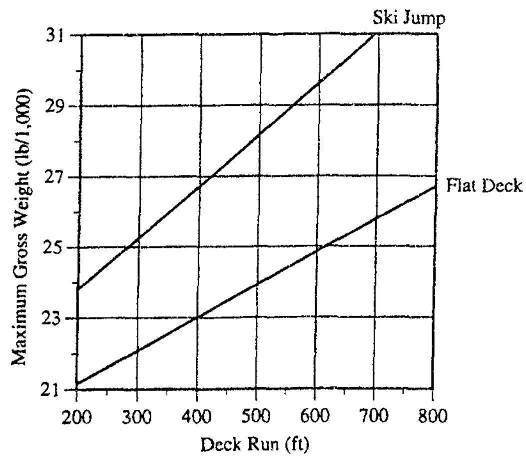


Figure 9
STO Gross Weight Capability
30 kt WOD
Tropical Day, Nominal SLW Engine

There are several safety enhancing characteristics inherent in a ski jump assisted launch. The tracking task during a ski jump launch is easier than during a flat deck launch because the tram lines are more prominent due to the height of the ramp. The ski jump launch produces no pitch-up tendencies at ramp exit and can be completely stick free for a few seconds after ramp exit. This reduces the tendency for pilot induced oscillations when attempting to capture a pitch attitude. The stick free characteristics inherent in a ski jump launch decrease pilot workload and allow more time for monitoring engine performance and critical launch parameters. The aircraft always has a positive rate of climb as it exits the ramp. The resulting additional altitude allows the aircrew more time to evaluate and react to emergency situations. The loss of an aircraft due to an emergency during a flat deck launch may be avoidable with the assistance of a ski jump. Pilot opinion is that the ski jump launch is the easiest and most comfortable way to takeoff in a Harrier.

Summary

PRINCIPE DE ASTURIAS proved to be an excellent platform for Harrier operations. The flight test program clearly demonstrated the performance gains, reduced pilot workload, and improved safety inherent in a ski jump assisted shipboard takeoff for Harrier aircraft when compared to that of a conventional flat deck. WOD requirements were approximately 30 kt less than flat deck requirements. Reduction in WOD requirements means significant fuel savings and flight operations having less impact on ship's heading. Deck run requirements were approximately 350 ft (107 m) less than flat deck requirements. Reduction in required deck run improves the Harrier/helicopter interoperability, allowing Harriers to use the forward half of the flight deck and helicopters the aft portion. Maximum payload capability for a ski jump assisted launch is up to 53% greater than flat deck capabilities, allowing 3,000 to 5,900 lb more payload. The heaviest Harrier to be launched from a ship to date was accomplished during the test program (31,000 lb). Increased payload capability allows shipboard Harrier operations to the same takeoff gross weight as shore based. A ski jump launch always produces a positive rate of climb at ramp exit. The resulting altitude gain allows the aircrew more time to evaluate and react to emergency situations. The loss of an aircraft due to an emergency during a flat deck launch may be avoidable with the assistance of a ski jump. Pilot opinion is that the ski jump launch is the easiest and most comfortable way to takeoff in a Harrier.

CONVENTIONAL TAKEOFF AND LANDING (CTOL) AIRPLANE SKI JUMP EVALUATION

Background

The U. S. Navy has also evaluated ski jump takeoff as an alternative to shipboard catapult launch for conventional airplanes. The Naval Air

Test Center conducted a ski jump takeoff test using a T-2C, an F-14A, and an F/A-18A operating from a variable geometry ski jump ramp to:

a) Evaluate the feasibility of the concept.

b) Define the operating limitations.

c) Document performance gains.

d) Verify and update aerodynamic and structural ski jump simulations.

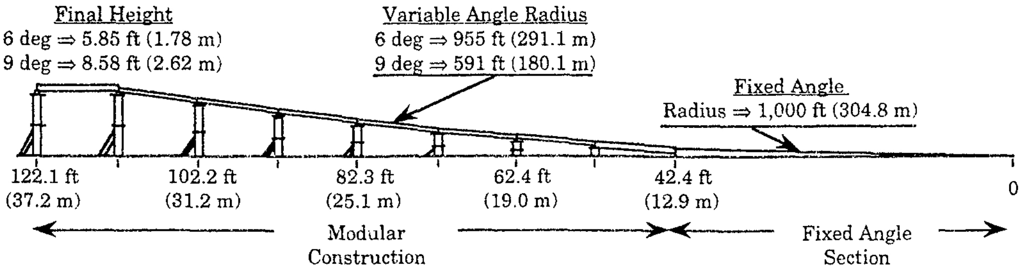
e) Propose airplane and ramp design considerations.

This section of this paper discusses the test program conducted with the F/A-18A airplane. Test results obtained with the T-2C and F-14A airplanes can be obtained from references 1 and 2. A more detailed discussion of the F/A-18A ski jump test program is presented in references 3 and 4.

Test Equipment

Ski Jump Ramp

The ski jump ramp, which was constructed at Patuxent River Maryland, was 60 ft (18.3 m) wide and 112.1 ft (34.2 m) or 122.1 ft (37.2 m) long, depending on the ramp angle. It was of modular steel construction of which the first 42 ft (12.8 m) was a fixed angle ramp with the remainder constructed of 10 x 30 ft (3.0 x 9.1 m) steel modules secured to steel pedestals. The heights of the steel pedestals was varied to give the desired ramp curvature. Figure 10 presents the general ramp arrangement and specific heights for the two ramp geometries. Leading into the ramp was a 60 ft (18.3 m) wide x 2,000 ft (609.6 m) long runway consisting of AM-2 matting. Centerline marking was two tram lines 2.5 ft (0.8 m) either side of the centerline. A modified holdback/release system was developed permitting stabilized thrust prior to the takeoff acceleration run. This system could be positioned anywhere along the runway to provide the desired ramp speed.



9 Degree Ramp Depicted

Distance Along Ramp ft (m)	Ramp Height ft (m)		Distance Along Ramp ft (m)	Ramp Height ft (m)	
	6 deg	9 deg		6 deg	9 deg
0	0	0	82.3 (25.1)	3.88 (1.18)	4.40 (1.34)
42.4 (12.9)	1.16 (0.35)	1.16 (0.35)	92.3 (28.1)	4.81 (1.47)	5.62 (1.71)
52.4 (16.0)	1.68 (0.51)	1.71 (0.52)	102.2 (31.2)	5.85 (1.78)	7.02 (2.14)
62.4 (19.0)	2.30 (0.70)	2.44 (0.74)	112.1 (34.2)	5.85 (1.78)	8.58 (2.62)
72.3 (22.0)	3.03 (0.92)	3.33 (1.01)	122.1 (37.2)	---	8.58 (2.62)

Figure 10
Ski Jump General Arrangement

Test Airplane

The F/A-18A airplane is a single-place, mid-wing, high performance, twin-engine strike fighter powered by two General Electric F404-GE-400 engines with an uninstalled thrust of 16,000 lb (71,171 N) each. The F/A-18 incorporates a digital fly-by-wire flight control system. The test airplane was aerodynamically and structurally representative of production airplanes. No modifications were made to the test airplane for the conduct of the tests. The following special flight test instrumentation installations were available:

a) Magnetic tape and telemetry system to record/transmit all required parameters.

b) Flight test instrumentation controls in the cockpit.

c) Ballast was installed to simulate the weight and CG of production equipment not installed in the airplane.

d) Radome mounted angle of sidestep vane which was displayed on the Head Up Display (HUD).

e) Retro-reflectors near the tip of each vertical tail to provide LASER tracking spatial data.

f) Landing gear instrumentation to obtain shock strut deflections and structural loads.

All build-up ground and flight tests and ski jump launch operations were conducted in the normal takeoff configuration. Table 2 details the test conditions. Two airplane gross weights were chosen to vary the thrust/weight ratio. External stores comprised two inert wingtip mounted AIM-9 (Sidewinder) and two inert nacelle mounted AIM-7 (Sparrow) missiles.

Table 2
F/A-18A Configuration Summary

Takeoff Configuration	Gross Weight lb (kg)	Field Takeoff Airspeed KEAS	Thrust/Weight
Half Flaps (30 deg)	32,800 (14,878)	146	0.52 MIL 0.76 Max A/B
	37,000 (16,783)	154	0.46 MIL 0.67 Max A/B

Manned Simulation

Extensive simulation effort was expended prior to the first ski jump takeoff. Simulation included both an aerodynamic and a landing gear loads model. The simulations not only were used to predict performance gains and structural loads, but enabled the test team to develop a build-down procedure during actual ski jump operations. Also, airplane single engine failure response characteristics and minimum safe ejection airspeeds in the event of an engine failure were established.

Early in the simulation effort, it was determined that additional performance gains could be realized by a "man in the loop" pitch attitude capture technique. Earlier simulation and all the ski jump takeoff tests with the T-2C and F-14A had been using the "stick free" technique. With these two airplanes, longitudinal trim was set to achieve the desired flyaway AOA. However, current F/A-18 flight control logic is such that a trim AOA is based on the initial stabilator trim position prior to the takeoff run. This AOA /trim schedule is shown in figure 11. Initial simulation runs at the higher ramp exit airspeeds permitted initial trim settings providing stick free flyaways at 12 deg AOA. However, as the ramp exit airspeed was reduced, the initial trim position had to be reduced to keep peak AOA's within limit (17 deg AOA true) during the initial rotation phase following ramp exit. This resulted in trim AOA's during the flyaway somewhat below any optimum for use during a ski jump takeoff. A pilot pitch capture technique was investigated which resulted in a significant decrease in the takeoff airspeed of approximately 15 kt below the stick free predictions. The technique was to allow the pitch attitude to increase during the initial rotation following ramp exit and peak at approximately 18 deg, at which time nose down pitch rate was generated as the flight control system attempted to acquire the commanded trim AOA. As the pitch attitude dropped to 15 deg the pilot commanded aft stick to maintain 15 deg pitch attitude. A target capture pitch attitude of 15 deg was chosen as the HUD pitch ladder is incremented every 5 deg and at zero rate of climb, a 2 deg AOA margin below the limit AOA was provided. During the flight test program, both the stick free and pitch capture techniques were evaluated.

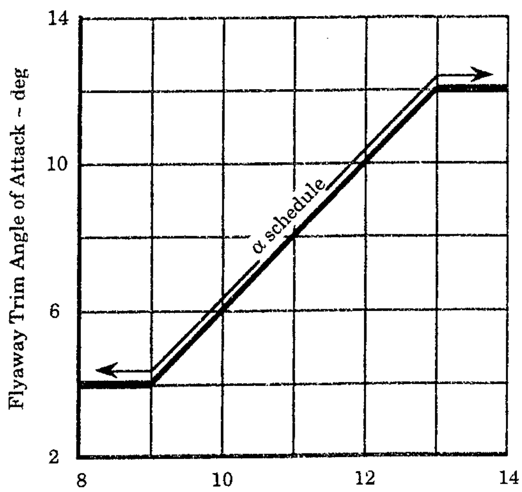


Figure 11
Trim Angle of Attack vs.
Initial Stabilator Trim Position

Single Engine Airspeed Considerations

The reduced takeoff airspeeds attainable with ski jump operations are significantly below minimum controllability airspeeds in the event of a single engine failure. Manned simulation allowed the test team to determine single engine airspeed boundaries and develop/employ aircrew procedures in event of an engine failure. Predicted F/A-18A minimum ski jump takeoff airspeeds were as much as 40 kt below dynamic single engine control airspeeds. Ski jump operations in this region mandated ejection should an engine failure occur at or shortly after ski jump ramp exit. F/A-18 safe ejection boundaries were established during simulation. With but one exception (32,800 lb with Max A/B on the 6 deg ramp), safe ejection was possible at ramp exit airspeeds below the predicted two engine minimum takeoff airspeeds. For this one condition, testing was conducted only down to the safe ejection airspeed. For all tests, ejection was mandatory below an airspeed of 120 kt.

Build-Up Test Operations

Prior to initial ski jump takeoffs, extensive build-up ground tests were performed. These included:

a) Acceleration performance: Following thrust stand calibration, normal field takeoff tests were performed to equate ground roll and speed to airplane gross weight and thrust setting. The results provided ground roll requirements to provide the desired ramp speeds.

b) Abort capability: The abort capability and pilot procedures were defined during simulated aborted takeoffs with the additional requirement of the pilot taxiing around the ski jump ramp (ramp simulated in position). During the takeoff ground roll at the desired groundspeed, the pilot retarded one engine to idle. After 1 sec, to simulate reaction time, the pilot retarded the other engine to idle and made aggressive lateral/directional inputs to the right on the runway. From these tests an abort location and speed could be determined. These data were provided to the pilot for each test event.

c) Single engine-committed to takeoff: Once past the abort capable point, the airplane is committed to ramp takeoff. A single engine failure is the most critical from a standpoint of keeping the airplane within the 60 ft (18.3 m) width of the ski jump runway and ramp. As with the abort capability testing, engine failure during takeoff ground roll was simulated; however, the pilot task was to maintain runway centerline. The maximum lateral deviation recorded was 6 ft when using Max A/B. If an engine failure had occurred past the abort capable point, the airplane was controllable within the width of the runway and ramp.

Test Results

General

A total of 91 ski jump takeoffs were obtained with the F/A-18A operating from both the 6 and 9 deg ramps. Significant reductions in takeoff ground roll up to 66% with corresponding takeoff airspeed reductions of 64 kt were achieved. With the proper longitudinal trim set prior to the takeoff, a "hands off" takeoff during rotation and flyaway following ski jump ramp exit was possible. However, additional performance gains were obtained using the pilot pitch attitude capture technique described earlier.

Performance Gains

As the ski jump takeoff exit airspeed was decreased, the minimum rate of climb during the flyaway slowly decreased. The minimum rate of climb as a function of ramp exit airspeed for the 9 deg ramp is shown in figure 12. The minimum ski jump takeoff airspeed tested was limited by zero rate of climb during takeoff. The minimum takeoff airspeeds achieved during tests are presented in table 3. Excellent correlation of flight test data with simulation predictions is apparent

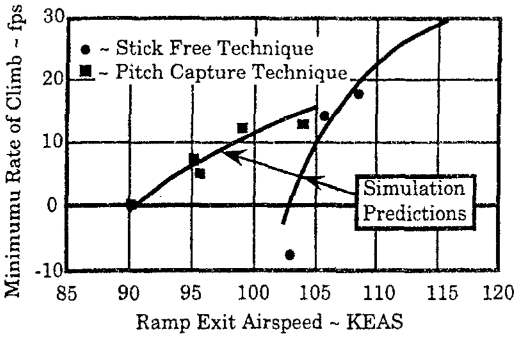


Figure 12
Minimum Rate of Climb during
Ski Jump Takeoff
9 Degree Ski Jump
F/A-18A Airplane
37,000 lb (16,783 kg) ~ Max A/B

Table 3
Ski Jump Minimum Takeoff Airspeeds

Thrust	Gross Weight lb (kg)	Minimum Takeoff Airspeed KEAS		Minimum Ground Roll ft (m)	
		6 deg ramp	9 deg ramp	6 deg ramp	9 deg ramp
MIL	32,800 (14,878)	102	98	1,075 (328)	850 (259)
	37,000 (16,783)	110	106	1,400 (427)	1,250 (381)
Max A/B	32,800 (14,878)	100	82	640 (195)	385 (117)
	37,000 (16,783)	99	90	700 (213)	575 (175)

NOTE: Minimum airspeed criteria: Proximity to zero rate of climb for all test points except 32,800 lb (14,878 kg) with Max A/B on 6 deg ramp which was limited by operation within safe ejection boundaries.

With the reduction in the ski jump takeoff airspeed was a corresponding reduction in the takeoff ground roll. F/A-18A ski jump reduction in takeoff distance for takeoff ground roll is presented in figure 13. The reduction is distance is related to the airplanes flight manual performance data for the test day conditions. The maximum reduction in takeoff ground roll relates to the minimum takeoff airspeed, whether dictated by zero rate of climb or single engine safe ejection boundaries. For any takeoff where minimum ground roll is required and the takeoff trajectory is not critical, the lowest airspeed is necessary. Reductions in takeoff distances are summarized in table 4.

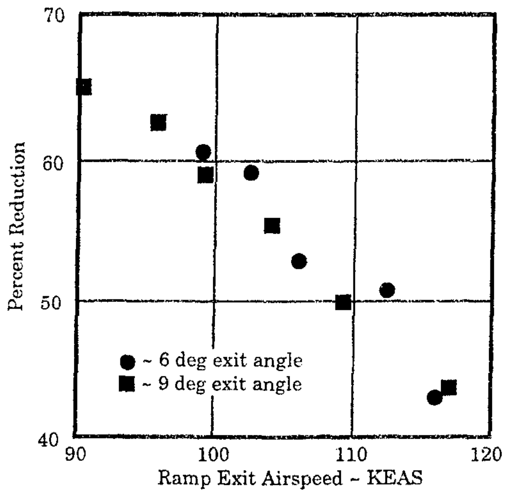


Figure 13
F/A-18A Reduction in Takeoff Ground Roll
during Ski Jump Takeoff
37,000 lb (16,783 kg) ~ Max A/B

Table 4
Comparison of Reduction in Takeoff Distance
F/A-18A Ski Jump

Thrust	Gross Weight(lb)	% Reduction in Takeoff Ground Roll	
		6 Deg Ramp	9 Deg Ramp
MIL	32,800 (14,878)	51	51
	37,000 (16,783)	51	55
Max A/B	32,800 (14,878)	49	62
	37,000 (16,783)	61	66

Ground Handling and Flying Qualities

The ski jump takeoff commenced when the modified holdback/release was activated. In both MIL and Max A/B thrust takeoffs, the initial acceleration was smooth with only a slight tendency towards pilot "head-jerk" at release. Although acceleration was more rapid in Max A/B, especially at the lower gross weight, the pilot had sufficient time to make pre-abort checks of engine performance. The airplane was not readily disturbed in its directional track by irregularities in the AM-2 matting; any small deviations were easily controlled ± 2.5 ft (± 0.8 m) of runway centerline. No significant longitudinal airplane response (pitch oscillation, nosewheel bounce, etc.) was encountered after holdback release. Nosewheel lightening was experienced prior to going onto the ramp; however, it was not objectionable and did not affect directional control. The abort capability point within ± 50 ft (15.2 m) was recognized by the pilot visually and reinforced by scanning the INS display for the predetermined ground speed for abort. Once beyond the abort point and committed to takeoff, the pilot was able to monitor engine performance and maintain centerline tracking. An increase in normal acceleration of 2 to 4 g characterized the entry onto the ramp, with more onset rate perceived on the 6 deg ramp than the 9 deg ramp. Using the 6 deg ramp, a rapid and abrupt g-onset was encountered, feeling to the pilot as though the airplane had rolled over a small obstacle. Entry onto the 9 deg ramp was smooth with predictable g-onset building rapidly and without the "thump" associated with the 6 deg ramp. Duration of elevated g on the ramp was short, lasting 1/2 to 3/4 sec. The dynamic landing gear interface with the ramp allowed for predictable and satisfactory flying qualities upon ramp exit.

The inclination of the ramp established the initial pitch attitude off the ramp. Longitudinal trim settings, accurate to within ± 0.5 deg, produced comfortable, initial positive pitch rates of 6-8 deg/sec. The trim setting was adjusted to obtain a peak pitch attitude of 18 ± 2 deg at less than the AOA limit of 17 deg. Pitch rates damped to zero or slightly positive during stick free takeoffs or were arrested to zero by pilot flight control input during pitch capture takeoffs. The airplane flew an arc with normal acceleration beginning at 0.25 g and increasing to 1 g over a 4 to 5 sec time frame. The 15 deg pitch capture was easily accomplished within ± 0.5 deg using longitudinal stick inputs of less than 2 inches (5 cm) and usually required only one stick input. No tendencies for longitudinal PIO were experienced during the pitch capture. The AOA peaked shortly after the peak pitch attitude and peaked a second time when the pilot captured 15 deg of pitch then smoothly decreased as the airplane accelerated.

Lateral control throughout the ski jump test program was excellent, even with a crosswind component. After departing the end of the runway, the airplane would yaw smoothly into the relative wind and little or no control input was required to maintain wings level attitude.

The F/A-18A digital flight control system eliminated any adverse flying qualities following takeoff from the ramp. The HUD information is sufficiently accurate for VMC and IFR conditions and would provide more than adequate information for night operations. The accurate and repeatable longitudinal trim system enhanced predictability for the ski jump takeoffs. All these factors made the F/A-18A ski jump takeoff, stick free or pilot-in-the-loop, easier than a field takeoff.

Structural Loads

Significant structural loads are imposed on an airplane during ski jump ramp transit. The stringent structural design requirements of US Navy carrier based airplanes provided the necessary strength for ski jump operations. The principle area of concern was landing gear loads. The desire to conduct initial ski jump takeoffs close to normal field takeoff airspeeds posed a dilemma in that the maximum loads were incurred during the first ski jump takeoffs. In general, main gear loads showed good agreement with simulation predictions; however, higher nose gear loads were obtained. A significant random variation in nose gear loads was experienced due to nose gear dynamics encountered prior to the start of the ramp. These nose gear dynamics were attributable to the unloading of the nose gear during the acceleration run and the uneven surface of the AM-2 matting runway. Most notable to the pilot during ramp transit is the incremental normal acceleration. Peak incremental accelerations measured at the airplane CG are shown in figure 14. Accelerations experienced by the pilot were higher.

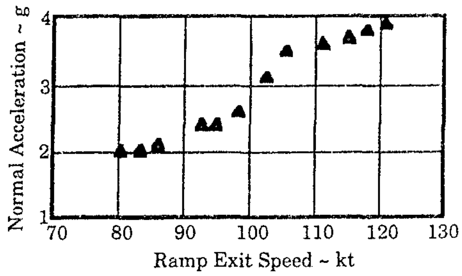


Figure 14
Maximum Normal Acceleration
During Ramp Transit
9 deg Exit Angle

A circular radius of curvature ramp, as tested, is not the optimum curvature profile for a ski jump ramp. Figure 15 depicts F/A-18A nose and main landing gear loads along the curvature of the ramp. High nose gear loads were encountered only during a small portion of the ramp. Ideally, landing gear loading should be equally distributed throughout ramp transit. This would permit attaining the desired ramp exit angle, ramp angle being the dominant factor in performance gains, using a minimum ramp size and still keeping the loads within limits. Simulation is the perfect tool to evaluate different ski jump ramp profiles to optimize nose and main landing gear loads.

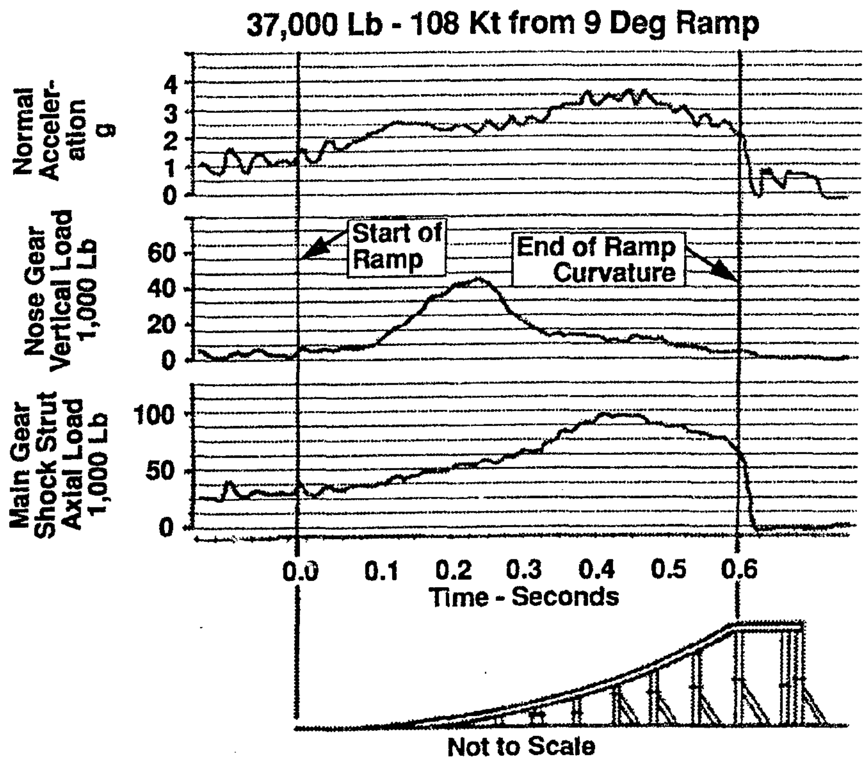


Figure 15
F/A-18A Nose and Main Landing Gear
Loading During Ramp Transit

Summary of CTOL Airplane Ski Jump Testing

Ski jump takeoff operations with current conventional fixed wing airplanes are possible. The significant performance gains, as exemplified by a 66% reduction in takeoff ground roll clearly demonstrates the potential of the ski jump concept. From a ground handling and flying qualities standpoint, a ski jump takeoff is an easier maneuver than a normal field takeoff. Longitudinal trim can be set to permit a stick free takeoff; however, additional performance gains were realized by the pilot using a pitch capture technique. Structural loads during ramp transit were well within the design limits of the test airplane.

CATAPULT/RAMP ASSISTED TAKEOFF

Introduction

The beneficial use of ramp assisted (Ski Jump) takeoff has been proven operationally by the British Navy, US Marine Corps and, more recently, by the Spanish and Italian Navies for AV-8B Harrier V/STOL aircraft. The US Navy test program described earlier in this paper demonstrated the feasibility of using Ski Jump to greatly reduce land-based takeoff distance requirements for CTOL aircraft as well. The analytical tools developed and validated during the US Navy CTOL program have been used to investigate potential benefits which might be derived from the use of Ski Jump for shipboard CTOL aircraft launch operations. A cross-section of operational US Navy carrier-based aircraft (F/A-18A, E-2C, A-6E, EA-6B, S-3A, F-14A) have been analyzed in conjunction with a modified mini-ramp geometry and steam catapult combination (Catapult/Ramp Assisted Takeoff (CART)). Aircraft performance, flying qualities, structural dynamics and piloting requirements were considered in determining possible required WOD reduction or allowable aircraft takeoff gross weight increase. Analytical results are presented which show potential reduction in WOD of from 5 to 35 kt for operational aircraft gross weights while keeping 1) maximum landing gear loads well below design limits and 2) minimum endspeeds above minimum aircraft control speed. The potential impact on aircraft carrier operations and possible operational problem areas are also discussed.

CRAT Concept and Ground Rules

The ski jump concept uses a ramp to rotate the aircraft flight path from horizontal to a positive climb angle at forward speeds less than those which are normally required to rotate the aircraft aerodynamically. The "early" rotation and lift-off provides an initial ROC and altitude margin which allows the aircraft to accelerate to flight speed while in a partially ballistic trajectory. A reduction in takeoff distance is achieved primarily as a result of lift-off speeds which may be considerably less than the stall speed of the aircraft.

CRAT uses the same concept as CTOL Ski Jump but replaces the free ground roll acceleration with a steam catapult assisted acceleration and the large ramp is replaced with a much smaller ramp due to deck space limitations. The lift-off speed reduction is applied to a reduction in catapult endspeed requirement for launch. In this case, takeoff distance is not reduced as it was in the previous CTOL Ski Jump effort but benefit is derived from:

1) Reduced WOD required for launch;

2) Increased takeoff gross weight at the conventionally required endspeed;

3) Some combination of 1) and 2).

For ease of analysis, the geometry of the "fixed" portion of the ramp used in the previous CTOL Ski Jump test program was used for analytical evaluation. The geometry is presented in figure 16 and represents the first 42.4 ft (12.9 m) of the ramp shown in figure 10. It has a reference radius of curvature of 1,000 ft (305 m), a departure angle of approximately 2.1 degrees and a maximum height above the flat deck of 13.875 in (35 cm).

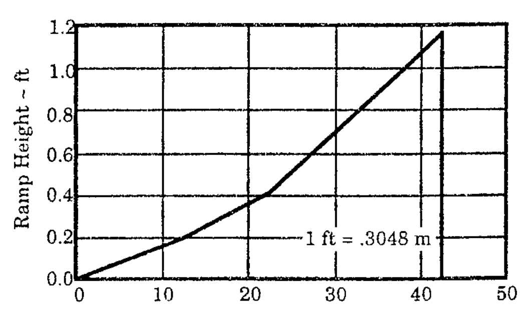


Figure 16
Mini-Ramp [42 ft (12.8 m)] Geometry

During a CRAT launch, the aircraft is assumed to leave the catapult (tow bar release) and immediately transition onto the ramp. Initial aircraft attitude, velocity, landing gear stroke, etc. are determined by the catapult stroke dynamics. Any stored energy in the landing gear due to strut compression during the catapult stroke will be released while the aircraft is on the ramp and resulting rotation is additive to that induced by the ramp. For the following analysis, each aircraft was assumed to enter onto the ramp with nominal end-of-catapult landing gear compression and aircraft pitch attitude (see table 5). Catapult endspeed was parametrically varied to evaluate performance benefits.

Table 5
Nominal Aircraft End-of-Catapult Conditions

Aircraft	Landing Gear Compression % Compressed		Pitch Attitude
	Nose	Main	deg
F/A-18A	80.7	75.0	-0.18
E-2C	0.0	83.5	1.35
A-6E	100.0	95.7	2.44
EA-6B	77.5	93.5	5.37
S-3A	96.0	84.5	1.10
F-14A	97.2	87.2	-2.13

"Minimum" Criteria Definition

The minimum launch airspeed for conventional aircraft catapult launch within the US Navy is defined as the minimum equivalent airspeed at the end of the catapult stroke for which the aircraft can safely fly away. Specifically, the minimum launch speed is set by a combination of related criteria which are described in reference 5 and are summarized here. The minimum launch airspeed is the highest of the following:

1) Stall Speed: The stall speed of the aircraft in the takeoff configuration or the speed at which stall warning first occurs if the warning does not significantly intensify as stall is approached.

2) Minimum Satisfactory Flying Qualities Speed: The speed below which the high AOA flying qualities of the configuration (e.g., damping, control response, etc.) become unsatisfactory.

3) Minimum Level Acceleration Speed: The speed at which sufficient thrust excess is available to provide at least 1 to 1.5 kt/sec of longitudinal acceleration.

4) Minimum Engine Inoperative Speed: The minimum airspeed for which there is sufficient lateral/directional control to counter an engine failure immediately following the catapult power stroke or for which single engine maximum rate of climb is attainable.

5) Minimum Rotation/Sink-off-the-Bow Speed: The speed below which aircraft pitch rotation is not sufficiently rapid or dynamic pressure is not great enough to provide enough lift (vertical acceleration) to arrest sink and establish level or climbing flight within some maximum acceptable amount of altitude loss; past experience indicates that this acceptable sink-off-the-bow is 15 to 20 ft (4.6 to 6.1 m).

The minimum conventional catapult end airspeed is typically defined by a combination of more than one of the preceding criteria over the takeoff gross weight range of a given aircraft. The operational minimum catapult end airspeed is set 15 kts higher than the previously defined minimum to allow for the negative effects of atmospheric disturbances, deck motion and non-optimum pilot technique, and to diminish (if not entirely remove) the probability of any sink-off-the-bow during normal launches.

Current practice for shipboard (AV-8A/B) ski jump operations is to define minimum launch speed such that the rate of climb during the flyaway does not become negative and available longitudinal acceleration does not become less than 1.5 kt/sec. Additionally, the obvious criterion that flying qualities must remain satisfactory down to the launch speed is also enforced. These criteria (zero minimum rate of climb, 1.5 kt/sec minimum acceleration and satisfactory flying qualities) were also used successfully to safely establish the minimum ramp endspeed for the CTOL Ski Jump program described earlier in this paper

Criteria for minimum endspeed for CRAT launches are not so clearly defined. Consider the possible flyaway trajectories of figure 17. When a ramp of any inclination is used to impart noseup rotation and rate of climb to a launching aircraft, the flyaway trajectory may be categorized into one of three classes. At higher speeds, comparable to conventional (flat deck) launch endspeeds, the trajectory exhibits positive rate of climb throughout (see trajectory 1 on the figure). As endspeed is decreased, the minimum rate of climb during the flyaway decreases until trajectory 2 is achieved with the rate of climb decreasing to zero but never becoming negative. This is equivalent to the minimum definition used for the previous CTOL programs. Finally, as endspeed is further decreased, the minimum rate of climb becomes increasingly negative and there is some minimum altitude (or maximum sink) achieved before rate of climb begins to increase (trajectory 3).

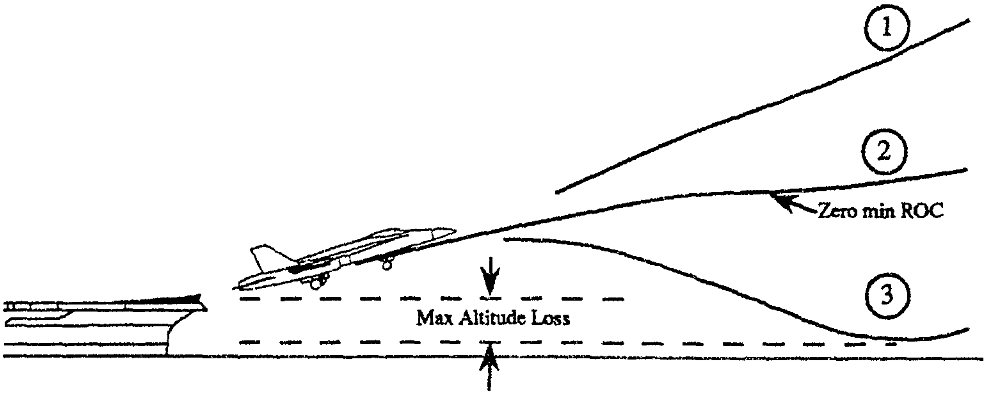


Figure 17
Possible CRAT Flyaway Trajectories

The likely candidate criteria for setting minimum endspeed are either 1) zero minimum rate of climb or 2) maximum allowable altitude loss. Zero minimum rate of climb has been proven for existing ski jump operations (both V/STOL and CTOL) and has the added benefit of always providing the pilot with a reassuring positive rate of climb. Maximum allowable altitude loss, on the other hand, is most like the current criteria for setting minimum endspeed for conventional catapult launch. Piloted flight simulation and perhaps even flight test is required to adequately choose one criterion or some compromise of the two (e.g., maximum rate of sink). Of course, conventional catapult launch criteria 2), 3), and 4) from above must still be satisfied. The analytical results which follow include potential performance improvements for both zero minimum rate of climb and maximum allowable altitude loss trajectories.

Analytical Results

The three degree of freedom (longitudinal, vertical and pitch dynamics) digital simulation model which was developed and validated during the CTOL Ski Jump program was used to analyze CRAT trajectories for a representative group of operational Navy aircraft. Table 6 list the aircraft configurations which were analyzed, including gross weights, thrust levels and flap settings. The models for each aircraft included nonlinear aerodynamic and thrust characteristics, nonlinear landing gear strut load and damping characteristics, and complete control system dynamics (see reference 6).

The analysis proceeded as follows. First, a conventional flat deck launch was simulated for each configuration at the minimum catapult endspeed and maximum altitude loss between 10 and 20 ft (3.1 and 6.1 m) was noted. These trajectories were used as a reference for comparison with the predicted CRAT launches. The ramp geometry of figure 16 was then simulated at the end of the catapult and the launch trajectories were recomputed for successively decreasing catapult endspeeds starting with the flat deck minimum and decreasing in 2-3 kt increments. Minimum rate of climb and altitude at zero rate of climb were recorded until the maximum altitude loss equaled or exceeded that for the flat deck launch. In all cases, nominal end of catapult conditions (landing gear strut compression, aircraft pitch attitude and CG height above deck) were assumed. Typical results are shown in figure 18 for the 46,000 lb (20,866 kg) F/A-18A with Max A/B Thrust. In this case, the flat deck minimum endspeed is 149 kt and the altitude loss at this speed is approximately 16 ft (4.9 m). With the ramp simulated, 16 feet of altitude loss occurs at an end speed of 129 kt providing a reduction in required catapult end airspeed of 20 kt. If the minimum were to be defined by zero minimum rate of climb instead of altitude loss, the minimum endspeed would be 137 kt providing a 12 kt reduction. Absolute minimum end airspeeds for all of the simulated configurations for flat deck launches with 15 to

20 ft (4.6 to 6.1 m) of sink and CRAT launches with comparable sink and zero minimum rate of climb are tabulated in table 7. Endspeed reduction potential for each of the minimum criteria (sink or zero rate of climb) is compared in figure 19. The results of table 7 and figure 19 indicate that minimum catapult end airspeed (and therefore required WOD) can be reduced by anywhere from 5.5 to 34.0 kt depending on the aircraft/configuration. If zero minimum rate of climb is used as a criterion, minimum endspeed reduction is decreased by a third to a half in most cases.

Table 6
Nominal Aircraft Configurations

Aircraft	Gross Weight lb (kg)	Thrust Setting
F/A-18A	46,000 (20,866)	MIL
	46,000 (20,866)	Max A/B
	52,000 (23,587)	MIL
	52,000 (23,587)	Max A/B
E-2C	53,000 (24,041)	MIL(1)
	53,000 (24,041)	MIL (2)
A-6E	46,000 (20,866)	MIL
	58,600 (26,581)	MIL(3)
EA-6B	50,000 (22,680)	MIL
	58,600 (26,581)	MIL
S-3A	44,000 (19,958)	MIL
	52,500 (23,814)	MIL
F-14A	59,000 (26,762)	MIL
	59,000 (26,762)	Max A/B
	69,800 (31,661)	Max A/B

Notes: 1. 10 degree flap setting
2. 20 degree flap setting
3. With loaded Multiple Bomb Racks

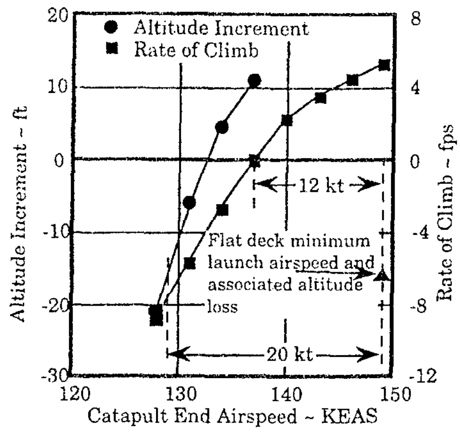


Figure 18
Altitude Loss and Minimum Rate of Climb vs. Endspeed
CRAT Launch of 46,000 lb (20,862 kg) F/A-18A with Max A/B

The last column of table 7 indicates the minimum control speed for each of the configurations. This speed is determined from engine out control capability or aerodynamic stall speed of each configuration, whichever is most critical. The table shows that the minimum end airspeed with the ramp and using the altitude loss criterion is significantly below the minimum control speed for only the F-14A Max A/B cases. Therefore, the wind over deck reduction potential for these case may be limited by minimum control speed restrictions. If the zero minimum rate of climb criterion is used, all of the predicted endspeeds are greater than the corresponding minimum control speed. Table 8 summarizes the predicted maximum nose and main gear reaction loads and limit loads for each configuration for all speeds up to the current flat deck minimum launch speeds. In all cases the predicted loads are well below the limit loads.

Table 7
CRAT Endspeed Summary

Aircraft	Configuration Wt ~ Thrust lb (kg)	Minimum Flat Deck Airspeed KEAS	Minimum Ramp Airspeed KEAS — Altitude Loss: Absolute KEAS	Altitude Loss: D kt	Zero Minimum ROC: Absolute KEAS	Zero Minimum ROC: D kt	Minimum Control Airspeed KEAS
F/A-18A	46,000 (20,862) ~ MIL	152.0	138.5	-13.5	144.0	-8.0	120.0
	46,000 (20,862) ~ Max A/B	149.0	129.0	-20.0	137.0	-12.0	130.0
	52,000 (23,583) ~ MIL	164.0	150.5	-13.5	155.5	-8.5	120.0
	52,000 (23,583) ~ Max A/B	161.0	141.5	-19.5	150.0	-11.0	130.0
E-2C	53,000 (24,036) 10 deg flap	122.0	115.0	-7.0	122.0	0.0	97.0
	53,000 (24,036) 20 deg flap	108.0	102.5	-5.5	108.0	0.0	97.0
A-6E	46,000 (20,862)	115.0	105.5	-9.5	110.5	-4.5	* 105.0
	58,600 (26,576)	144.0	134.5	-9.5	138.5	-5.5	* 120.0
EA-6B	50,000 (22,676)	119.0	110.0	-9.0	114.0	-5.0	* 107.0
	58,600 (26,576)	129.0	119.0	-10.0	122.0	-7.0	* 120.0
S-3A	44,000 (19,955)	104.0	93.0	-11.0	102.0	-2.0	88.0
	52,500 (23,810)	115.0	106.0	-9.0	110.0	-5.0	88.0
F-14A	59,000 (26,757) ~ MIL	122.0	99.0	-23.0	111.0	-11.0	+ 88.0
	59,000 (26,757) ~ Max A/B	122.0	92.0	-30.0	105.0	-17.0	+ 103.0
	69,800 (31,655) ~ Max A/B	135.0	101.0	-34.0	112.0	-23.0	+ 103.0

* - 2 engine stall speed
\+ - Mid-Compression Bypass open, locked rotor, 10 deg sideslip

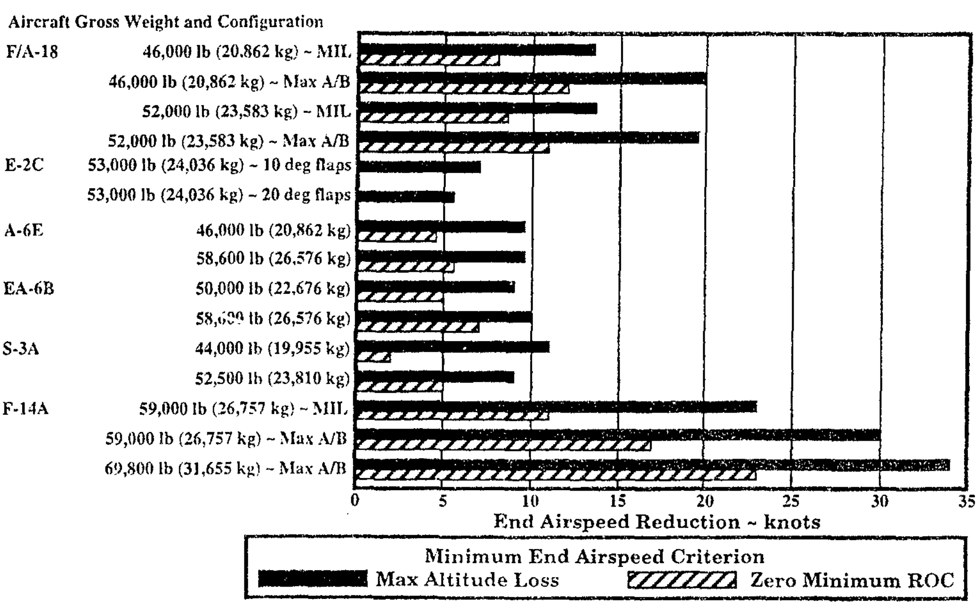


Figure 19
CRAT Endspeed Reduction Potential

Table 8
CRAT Landing Gear Load Summary

Aircraft	Configuration	Landing Gear Reaction Load ~ 1,000 lb (kN)			
	Wt ~ Thrust lb (kg)	Nose		Main	
		Maximum	Limit	Maximum	Limit
F/A-18A	46,000 (20,862) ~ MIL	53.2 (236.6)	80.0 (355.9)	48.3 (214.9)	77.0 (342.5)
	46,000 (20,862) ~ Max A/B	50.6 (225.1)	⇓	46.4 (206.4)	⇓
	52,000 (23,583) ~ MIL	66.7 (296.7)	⇓	61.9 (275.3)	⇓
	52,000 (23,583) ~ Max A/B	64.5 (286.9)	⇓	59.6 (265.1)	⇓
E-2C	53,000 (24,036) 10 deg flap	19.4 (86.3)	81.0 (360.3)	65.8 (292.7)	109.0 (484.9)
	53,000 (24,036) 20 deg flap	14.4 (64.1)	⇓	46.4 (206.4)	⇓
A-6E	46,000 (20,862)	41.9 (186.4)	64.0 (284.7)	38.4 (170.8)	88.0 (391.4)
	58,600 (26,576)	47.9 (213.1)	⇓	54.3 (241.5)	⇓
EA-6B	50,000 (22,676)	35.6 (158.4)	132.0 (587.12)	67.8 (301.6)	137.0 (609.4)
	58,600 (26,576)	41.7 (185.5)	⇓	70.4 (313.2)	⇓
S-3A	44,000 (19,955)	36.5 (162.4)	80.0 (355.9)	29.0 (129.0)	105.0 (467.1)
	52,500 (23,810)	38.3 (170.4)	⇓	36.2 (161.0)	⇓
F-14A	59,000 (26,757) ~ MIL	58.8 (261.6)	70.0 (311.4)	41.0 (182.4)	100.0 (444.8)
	59,000 (26,757) ~ Max A/B	58.8 (261.6)	⇓	42.2 (187.7)	⇓
	69,800 (31,655) ~ Max A/B	65.1 (289.6)	⇓	52.5 (233.5)	⇓

Operational Considerations

While the preceding simulation results indicate the strong potential for reducing WOD requirements for catapult launch from an aerodynamic performance viewpoint operational factors must still be considered. For example, is there sufficient usable space in front of existing catapult installations to accommodate a ramp of the required length? Should ramps be positioned in front of all catapults or just the bow catapults? If ramps are positioned in front of the waist catapults, what is the effect on bolter performance/characteristics and safety? Should operational launch speed be based on the minimum altitude criterion plus 15 kt excess, the zero minimum rate of climb criterion or some other criterion? These questions, as well as I'm sure others, must be answered before CRAT becomes an operational reality.

Summary

In summary, non-real time simulation has indicated the potential to reduce WOD requirements for current US Navy carrier-based aircraft by as much as 35 kts using a combined catapult/ramp assisted launch. Maximum landing gear reaction loads remain well within acceptable limits and minimum airspeeds experienced are above the minimum aircraft control speeds. Based on the non-real time simulation, pilot-in-the-loop simulation followed by land-based demonstration flight test is expected to validate the concept. If the flight test demonstration is successful, ramp shape, size, placement and construction will be optimized and the feasibility of carrier-based flight test will be investigated.

RELEASE

The conclusions concerning benefits of CRAT are the opinions of the authors and do not necessarily reflect those of the Naval Air Systems Command.

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FLYING TEST-BED STUDIES ON AUTOMATED FLY-BY-WIRE CONTROL SYSTEMS FOR ADVANCED HELICOPTERS

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The missions and environment of modern helicopters are being constantly widened and becoming more complicated, whereas the pilot work caused by the processes of handling and control of various subsystems are increasing. Therefore the control systems being developed should provide improvements of helicopter stability and controllability sufficient to decrease and restrict the psycho physiological loads on the pilot within permissible limits consistent with a prolonged normally tolerated load.

One of the possible directions in solving the above problem and improving the effectiveness of new-generation helicopters is the development of fly-by-wire control systems (FBWCS) incorporating controllability augmentation subsystems (CAS). The implementation of highly automated FBWCS (fig. 1) provides, together with electronic systems of navigation-flying data display, the possibility to make good use of small-size and light side control sticks (SCS) rather than standard wheels and central sticks. These systems enable convenient integration of the pilot and the helicopter control loop, make it possible to increase the response speed of this loop (fig. 2) and reduce the level of the pilot work load, which results in a considerable improvement in the helicopter controllability, especially in maneuvering flight conditions, and the handling accuracy in flight conditions close to steady. The application of the FBWCS with the SCS improves also the ergonomic characteristics of the cockpit clearing the space in front of the instrument panel, opens up fresh opportunities for a rational arrangement of data display systems and the entire cockpit, increases flight safety and survivability of the helicopter, especially in combat environment.

In the practice of our country flight tests of handling qualities of the helicopter equipped with the SCS were first conducted in LII in 1989. In the course of these researches it has been revealed that in hover and translation flight conditions the handling of the Mi-8 and Mi-6 with the FBWS and SCS presents no difficulties even without automation aids, does not require any special skill and training and, according to the pilots' assessment, is more convenient as compared with the handling using a conventional cyclic-pitch stick.

Subsequently a considerable amount of flight researches (more than 100 flights) have been carried out to evaluate the effectiveness of the FBWCS with the controllability augmentation system (fig. 3) and low speed side sticks for controlling the cyclic and accumulated main rotor pitch. Test pilots from various organizations of the aviation inustry and Air Forces took part in the studies, the helicopter was piloted in instrument and visual flight conditions. The characteristics of the SCS-equipped helicopter angular and translational control loop on pitch, rank and altitude channels were evaluated in hover and translation flight conditions.

A comparison of spectral densities of control forces (movements) and the helicopter angular motion, especially in hover mode and movements near the ground, when flying with the SCS and standard cyclic-pitch stick, indicates that the side stick provides the pilot with a more wide spectrum of control actions, which he can realize without an increase in his own work loads and additional psycho physiological strains.

Assuming that the limiting values of the pilot's working frequency range are defined by the maximum speed of control movements, we can estimate the allowable amplitudes of stick deflections versus frequency (Fig. 2). It is show that the region of allowable side stick movements is considerably wider than that for the conventional control stick. In controlling the helicopter angular motion and flight speed, the configuration of the movable side control stick with low force gradients (0.15—0.20 kg/mm) has received the highest scores.

The simplest control loop algorithms incorporating the side control stick (Fig. 1) feature the availability of smoothing filters, non-liner elements with dead band and amplifying links with variable, in general case, gear ratios. For relieving the SCS a parallel trimming circuit with an integrating element is provided in the control loop.

An analysis of transient processes in the helicopter angular movement using the SCS and corresponding flight estimates in hover and translation flight conditions on instrument and visually has shown that the helicopter with the SCS is stabilized sufficiently easy in steady conditions, and in performing separate evaluations and maneuvers a sufficiently accurate metering of pilot's actions is provided. The pilot can evaluate the helicopter controllability more readily than using the conventional stick in all main conditions, both in instrument and visual flights. However the instrument flight requires an increased pilot attention, especially at landing approach.

The necessary level of handling qualities for the helicopter with the FBWCS is provided by a proper selection and application of the controllability augmentation subsystems (fig. 3) and by the optimization of their control algorithms. The feasibility and effectiveness of variable-structure CAS algorithms were evaluated in flight researches and in the process of computer aided simulation. Studied were the control algorithms with reference and predictive models, as well as the algorithms of automatic system structure change-over depending on flight modes and pilot's actions. Fig. 4 illustrates the regions of the SCS structure change-over versus the helicopter angular rate and the reference model. In flight studies the helicopter controllability was evaluated using SCS and CAS in two main configurations which enable attitude and angular rate control. The comparison of these configurations (fig. 5) has shown that the angular rate system offers advantages in speed of response at large (more than 10—12) changes of the helicopter attitude over an operational frequency range (up to 1.4—1.6, c-1). This makes the system of such configuration preferable for maneuvering flight conditions. The altitude control system allows the pilot to keep the helicopter attitude with a sufficient accuracy and to control the pitch at high response speed with insignificant changes (within 10—12) over a wide frequency range, which appears to be preferable for applying this configuration in the conditions of general helicopter navigation.

The feasibility of using digital systems in FBWS raised the question of a more complete utilization of resources and special features of digital technology, namely, high computational productivity, possibility to store data bulks, etc.

Analogous means of control laws realization imposed stringent restrictions on the control algorithms used, which were usually reduced to simple functional dependencies of the control vector at each moment of time upon the estimate of the state vector at that moment of time. Additional information concerning the parameters of the controlled object, environment or the nature of controlling actions was usually used for correcting gear ratios or for control structure change-over. The on-board digital systems make it possible to widen the scope of algorithms used, select among several possible solutions, such which correspond a specified criterion.

One of the new and promising approaches is the concept of control with a predictive model. The essence of the concept is that in parallel with the real controlled object its mathematical model is functioning in the computer, which is used for defining the object motion during a certain future interval of time. By comparing the predictions with a specified future trajectory or limitations, defined are the required control actions as well as the necessity to draw the pilot attention in order to prevent the controlled object from overstepping the restrictions.

Investigations of predictive systems carried out at LII for various problems of helicopter control have revealed that this concept is promising. It is evident that the predictive systems will find the most extensive application in solving the problems on flight safety and in a number of complicated tasks (for example, provision of low-level flights).

The results of flight studies on the fly-by-fire control system incorporating the CAS and SCS allow us to conclude, that the FBWCS is a highly convenient aid of helicopter control. The application of the FBWCS with the CAS and side control sticks makes it possible to improve significantly the helicopter controllability, relax psycho physical tension of the pilot during the handling and most naturally integrate him with the control loop. In further studies it is reasonable to investigate the problems of redundancy and flight safety in performing all operational flight modes of general and special application in visual and difficult weather conditions.

Together with the development of highly automated control systems, the problem of ensuring flight safety in case of automatics failure is still pressing especially the determination of the helicopter dynamic characteristics.

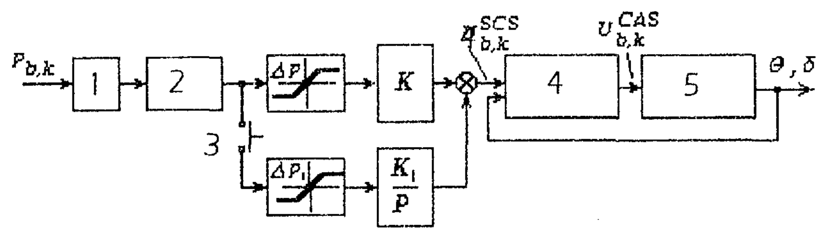
Thus, flying test beds enabled simulation of helicopters with different characteristics of stability and controllability over a wide range of their variations, while the flying test bed Mi-6 provided the capability of varying the control system characteristics (stick force gradient, the magnitude of friction and backlash in the control circuit).

Parametric studies aimed at evaluating the controllability of helicopters of various model were carried out channel-by-channel. In this case if, for example, the longitudinal controllability was being investigated, the simulation system of two other channels was adjusted to the characteristics of the reference helicopter in a corresponding flight regime. The controllability was evaluated in a series of flight regimes for which the initial flight conditions were hovering near the ground and horizontal flight with cruise speed. When evaluating the lateral controllability, in addition to hovering over a point and stabilizing application of lateral and directional control, forward right and left sideward movements were made, as well as left and right turns by 90 deg. relative to the reference course.

In the second case (initial regime is level flight with cruise speed) a series of flights in conditions necessary for the pilot to assess the controllability of proposed helicopter models were performed.

The pilots' evaluations were tape recorded directly in flight after completion of the series of flight regimes for each model to be tested. Apart from the detailed evaluation of the proposed model stability and controllability, the pilots gave an integrated assessment on three-mark scale: "good", "acceptable", "unacceptable".

As a result of the investigations carried out, there were obtained domains of combinations of dynamic stability parameters versus the effectiveness of longitudinal, lateral and yaw control channels, which provide different levels of the helicopter controllability — from "good" to "acceptable". It is shown that at the same stability characteristics it is possible to change the helicopter stability from unacceptable to good marks by varying the control effectiveness. At the same time there is a domain of combinations of the dynamic stability parameters, where the poor controllability cannot be changed by selecting the control effectiveness. Under this circumstance it is evident that the characteristics of dynamic stability can be normalized irrespective of the control system characteristics, but the characteristics of the control system should be highly rated by the pilot. Similar results have been obtained in the course of investigating the lateral stability and controllability.



Fugure 1. Block-Diagram of Test Fly-by-Wire Control System with a Two-Channel Strain-Guge Side Control Stick and Controllability Augmentation to be Used for Angular Motion Control

1 - SCS
2 - Filter
3 - Trim Button
4 - CAS
5 - Servoed Helicopter

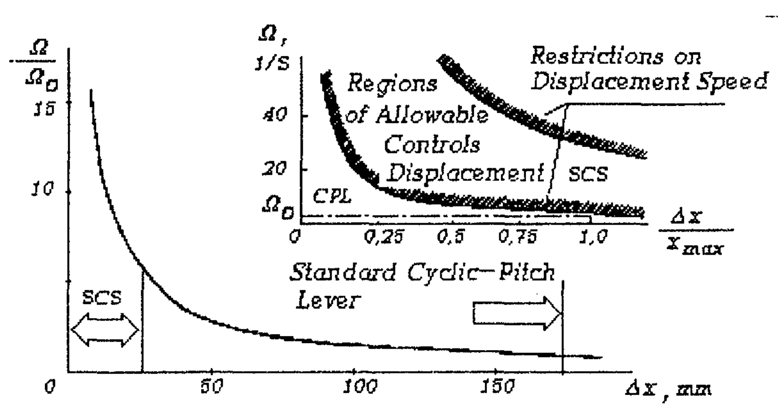


Figure 2. Change in Limiting Width of Spectra of Controlling Pilot Actions Versus Amplitude of Controls Displacement

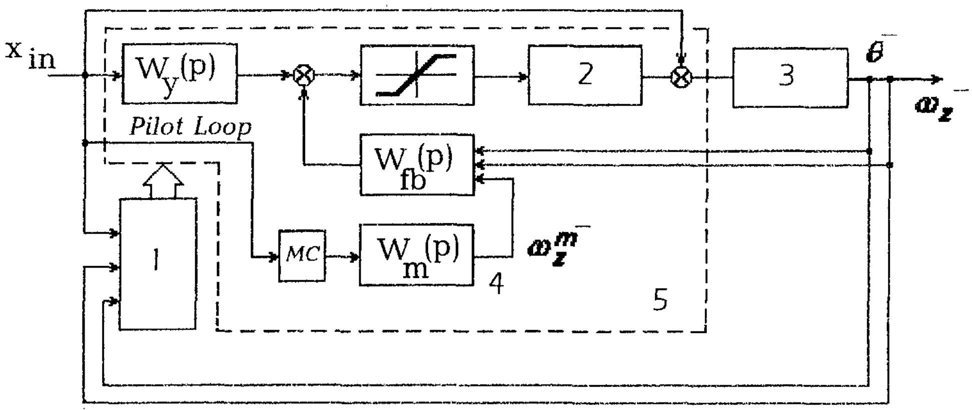


Figure 3. Block Diagramm of the Control Augmnetation System for Maneuverable Helicopter

1 - Automatic CAS Structure Change-Over
2 - Servo Drive
3 - Helicopter
4 - Reference Model
5 - CAS

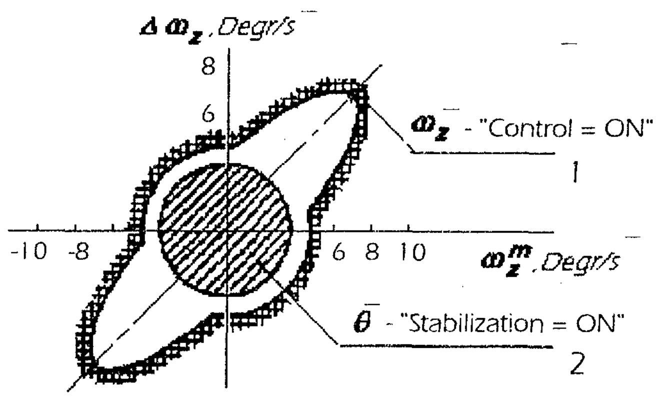


Figure 4. Domain of CAS Structure Change-Over as a Function of Flight Conditions and Pilot Actions

1 - Control;
2 - Stabilization.

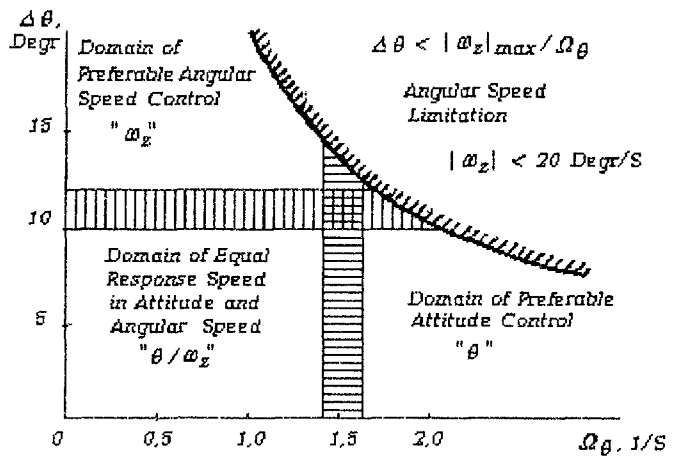


Figure 5. Domain of Applying SAS Algorithms of Various Types Depending on Flight Modes (Amplitude and Loading of Angular Motion Spectra)

ENDORSING TACTICAL OPERATION OF A FAA CERTIFIED HELICOPTER

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ABSTRACT

The acquisition of an "off the shelf" civil certified helicopter for military applications has the potential to reduce cost significantly. However, civil certification standards in accordance with the Federal Aviation Regulation (FAR) Part 27 or 29 does not ensure that the helicopter will have the necessary handling qualities and mission survivability to safely conduct the intended military mission. Compliance of specific issues to military specifications may have to be established to ensure that requirements not covered by the FAR are satisfied before license to conduct tactical operation is granted.

This paper provides a review of the FAR certification process and of the various military specifications such as MIL-H-8501A Helicopter Flying and Ground Handling Qualities, MIL-F-83300 Military Specification for Flying Qualities of Piloted V/STOL Aircraft and ADS 33-C Handling Qualities Requirements for Military Rotorcraft. This paper also presents the flight test programme that was developed to determine the tactical flight envelope for the Canadian Forces Utility Transport Tactical Helicopter (CFUTTH) based on an extension of the baseline FAR 29 certification of the BELL 412 helicopter. Additional flight testing to augment the operational capability of the aircraft is outlined. Concluding remarks and recommendations based on lessons learned throughout this programme are offered.

Introduction

On 9 September 1992, a contract was awarded by the Canadian Department of National Defense to Bell Helicopter Textron of Canada (BHTC) for the purchase of 100 BELL 412CF to fulfil the role of Canadian Forces Utility Tactical Transport Helicopter. This project was valued at $ 1,291 Billion canadian dollars and included a full motion mission simulator subcontracted by BHTC to CAE, a Canadian worldwide leader in simulator technology. The intended roles for the new helicopter was Utility Transport and Combat Support (Search and Rescue).

The contract for the BELL 412CF included the provision of certification of airworthiness in accordance with Part 29 of the Federal Aviation Rules (FAR) covering the certification of transport category helicopters. However, as this paper explains, this civil certification did not ensure that the helicopter had the necessary handling qualities and mission survivability to safely conduct the intended military mission. Nor did the civil certification addressed the proof of compliance of certain aircraft systems. Thus, in many cases, compliance to military specifications had to be established to ensure that requirements not covered by Part 29 of the FAR were satisfied before a clearance to conduct tactical flight was granted.

This paper, while praising the overall benefits of acquiring a civil certified aircraft for a military role, presents the gaps between civil and military certification and describes the experience gained by the Aerospace Engineering and Tests Establishment (AETE) during the conduct of a flight test program that made the issuance of a tactical flight clearance for the BELL 412CF possible.

The Military Helicopter

To better understand the intricacies of the procurement and certification of military helicopters, it is convenient to identify the three major components of a military helicopter: the airframe, the power plant and the avionic systems. Requirements for the airframe generally relate to the role and the environment of operations. For example, in the case of flight deck or offshore operation, the airframe component is expected to provide a large cargo area and a certain type of landing gear. The power plant component requirements relate to performance in terms of all-up-weight, hover ceilings, and numerous other related criteria. While the requirements of the airframe and powerplant components are often similar for a number of military and civilian roles, the avionic systems component is directly related to the military role for which the rotorcraft is intended.

Project Life Cycle: The Traditional Military Approach

The traditional project life cycle for the acquisition of a military rotorcraft involves the preparation of one critical document: the Statement of Requirements (SOR). This document provides a definition of the features and the capabilities for all three of the major components that the rotorcraft must possess to be able to fulfil its intended mission. This document can easily turn into a wish list containing unrealistic technical expectations and initial feasibility studies based on the requirements can often reveal high research, development and integration costs. To a certain extent, the SOR can "make or break" a project since without exception, in the traditional military approach, the aircraft design is started on the drawing boards.

The SOR will "make" the project if it is well balanced in terms of features, capabilities, cost and risk associated with the technical challenges it implies. The SOR will break the project if the technical challenges not only contain a high level of risk, but also if the cost to meet the features and the capabilities is high.

Project Life Cycle: The Modular Approach

Another approach in the design of a military helicopter is to use existing airframe and power plant components from a FAA certified helicopter and to integrate the avionic component with "off the shelf" equipment thus adapting the aircraft to the military role. All data used in the certification process of the airframe and powerplant can be credited and the FAA certification of the new aircraft can be obtained by minimal testing of the components of new avionics systems for which civil certification is required (navigation aids, radios, etc). FAA certification need not be concerned with systems that are military in nature such as chaff and flare dispensers and armoured protection.

It is however important to recognize that while the civil certification process takes you a long way to ensuring suitability of an aircraft it does not guarantee that the intended military mission can be safely conducted. This does not imply that a civil certified aircraft can not be acceptable for certain tactical missions, it merely implies that the civil certification process does not provide the necessary data to verify that it is.

The concept of flying and handling qualities are very important to understand the difference in scope between civil and military certification. The next sections will discuss flying and handling qualities and will present one civil and three military standards.

Flying Qualities versus Handling Qualities

The difference between flying and handling qualities is an important consideration when comparing the civil and military standards. As these terms are used intensively in the following presentation of helicopter standards, a short explanation of the difference is in order.

The difference between flying and handling qualities can best be described by figure 1. The flying qualities of an aircraft are based on its agility, stability and manoeuvrability which are characteristics of an aircraft that are independent of the mission or the environment. These characteristics are derived from stability derivatives, power margins and control systems and they can be obtained using standard flight test techniques. When the aircraft is evaluated for an intended mission task in the intended conditions and environment, the notion of pilot workload becomes important. This notion is central to the handling qualities rating of an aircraft. An aircraft with desirable flying qualities may have undesirable handling qualities for certain mission tasks and environment because the pilot workload required to accomplish the mission is unacceptable.

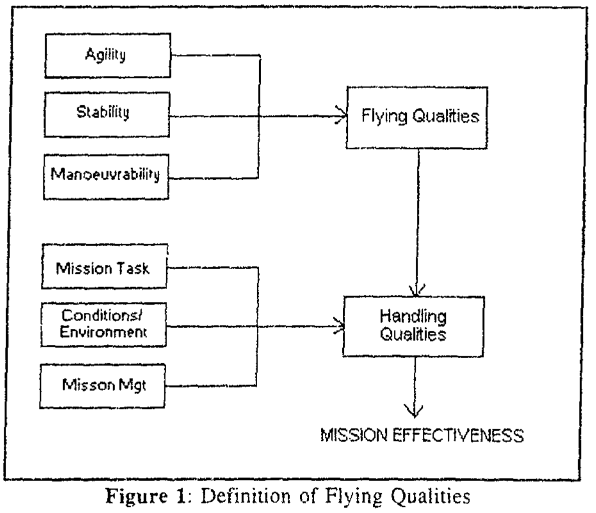

Figure 1: Definition of Flying Qualities and Handling Qualities

Standards for Helicopters

While the military standards are often too detailed, burdensome, and if invoked for a program will certainly incur increased costs, one must realize that they have been developed based on years of military operational experience and program management. Consequently, while they may have evolved to the point where they are not practical for some military programs, the technical and operational problems which they are intended to avoid may still need to be addressed. It is the very difference in military and civilian operational roles which has forced the evolution of two distinct certification process. But it is the most prohibitive cost of military certification that has caused many military forces to reevaluate their process and determine if a better way can be found by following their civilian brethren.

The difference in the certification process used for commercial and military helicopter can be appraised by exploring various norms used today:

a. FAR 29 - Certification of Transport Category Helicopter;

b. MIL-H-8501A - Helicopter Handling Qualities Specification;

c. MIL-F-83300 - Flying Qualities of Piloted V/STOL Aircraft; and

d. ADS 33-C - Handling Qualities Requirements for Military Rotorcraft.

The FAR Part 29 Certification of Transport Category Helicopter

The FAR Part 29 specification is applicable to multi-engine transport category helicopters intended to carry more than 9 passengers. The FAR 29 criteria are intended to provide a certain level of safety to the would-be paying passengers of these helicopters. For instance, specific safety requirements are found for aircraft performance while experiencing one engine failure (n-1 engine operation). Other examples of safety requirements are door locking mechanisms, fireproofing materials and fire extinguishers in engine compartment, master gear box endurance (30 minutes without oil), personnel egress, etc.

The flying qualities criteria of Part 29 cover static and dynamic stability and manoeuvrability. The static stability ensures that the flight control displacement versus trimmed airspeed follows a positive gradient (though a 10 percent negative displacement is allowable under special conditions) whereas the dynamic stability criteria ensures that oscillations will dampened within one cycle when they are short in period (less than 5 seconds). The manoeuvrability criteria ensures that enough margin exists for the cyclic and directional control in all flight regime.

Part 29 of the FAR is the subject of an FAA publication referred to as the Advisory Circular 29-2 (AC 29-2) which provides more details on certification requirements. Whereas the FAR 29 is the governing document, the AC 29-2 is the document that certification engineers use. For instance, the earlier mentioned criteria for static stability and manoeuvrability are only found in the AC 29-2, not in the FAR 29.

A major distinction between FAA certification and military specification compliance is in the demonstration process. For military specifications compliance, the procuring agency generally determines how compliance will be demonstrated as part of the contract. For FAR certification, it is up to the aircraft manufacturer to present a certification plan to the FAA which is a point by point description of how compliance is going to be demonstrated. Options available to the manufacturer include demonstration based on similarity to a previously certified model, analysis, simulation, and flight test. Needless to say and rightly so, flight test is sometimes the most expensive mean of demonstration and other methods will be attempted first when possible.

When dealing with the procurement of a Part 29 certified aircraft, the procuring agency has no say in the means of demonstration used in the certification process. Decisions on demonstration are only between the FAA and the manufacturer regardless whether the procuring agency is purchasing one aircraft or one hundred. In addition, the data collected for certification remains property of the manufacturer and the procuring agency is not automatically entitled to it. The only data available to the procuring entity by right is the data contained in the type certificate and the aircraft flight manual.

MIL-H-8501A - Helicopter Handling Qualities Specification

The MIL-H-8501A specification was originally published in November 1952 and revised in September 1961. It covers the requirements for flying qualities and some relevant ground handling qualities for military helicopters.

The flying qualities requirements contained in MIL-H-8501A cover the longitudinal, lateral and directional axes. Specifications for static and dynamic stability are found in addition to control forces requirements.

The requirements that go beyond the flying qualities requirements found in the civilian certification are the control response criteria and vibration characteristics. Control response criteria dictate the rate of attitude change resulting from control inputs. Vibrations characteristics provides maximum values of vibration that must not exceed by the aircraft.

Although MIL-H-8501A provides additional flying qualities requirements not found in the Part 29 of the FAR, it falls short of addressing handling qualities requirements as it does not cover the concept of operational mission and pilot workload.

MIL-F-83300 - Flying Qualities of Piloted V/STOL Aircraft

This specification was published in December 1970 and superseded MIL-H-8501A. The objective of this specification is that no limitations on flight safety or on the capability to perform intended missions result from deficiencies in flying qualities.

This specification includes flying qualities requirements and emphasizes that these requirements be met during the intended operational missions. The concept of operational flight envelope, defined in terms of airspeed, altitude and load factor ranges, is also included to recognize that operational missions occur in environmental conditions likely to impact the pilot workload.

A three-level rating system for flying qualities is used in this specification. The scale goes from level 1 for adequate flying qualities to level 2 for adequate flying qualities with increased workload or degradation in mission effectiveness to level 3 for a safely controllable aircraft with excessive pilot workload or inadequate mission effectiveness. These flying qualities ratings are very much similar to handling qualities ratings since the concept of mission effectiveness and pilot workload are included.

The requirement for the manufacturer to document aircraft normal states and failure states is also part of MIL-F-83300. This ensures that flying qualities requirements are met not only in cases where all aircraft systems are perfectly functional, but also in cases where some aircraft systems have failed.

MIL-F-83300 was the first step towards recognizing that mission effectiveness and pilot workload are germane to helicopters intended for military operations. This standard lead the way for the most modern specification developed for military helicopters described next.

ADS 33C - Handling Qualities Requirements for Military Rotorcraft

The product of a fifteen year effort by the US Army assisted by the work of the Flight Research Laboratory of the National Research Centre of Canada, ADS 33-C is the most recent norm for military rotary wing aircraft. It has been in use since 1988 for the development of the Comanche aircraft.

There are two sections of interest in ADS 33-C. Section 3 contains quantitative requirements and criteria that are deemed to be applied in the design of the rotorcraft such as flight control bandwidth and phase delay criteria. Section 4 defines flight test manoeuvres called "mission task elements" designed to evaluate the handling qualities of the rotorcraft. These manoeuvres are derived from the intended operational role of the helicopter. They require a high level of precision or aggressiveness which in either case will cause pilot gain to increase thus highlighting obvious aircraft limitations based on control input frequency. These handling qualities limitations may be revealed by pilot-induced oscillations (PIO), rotor overspeed or droop, or simply by performance stagnancy or degradation when the control input frequency increases.

Parameters and corresponding tolerances are defined for desired performance (level 1 on the Cooper Harper scale) and acceptable performance (level 2 on the Cooper Harper scale). These parameters and tolerances are expected to be defined by the procuring entity. This is to ensure that an aircraft is judged using standards that are in line with its intended role. For instance, a heavy cargo transport helicopter would not be judged using the same standards as a light attack helicopter.

Our experience has shown that minimal importance can be given to the absolute level of performance used in the definition of the manoeuvres. In other words, handling qualities assessment using ADS-33C can be seen as a tool rather than a pass/fail test. Indeed, ADS 33-C pits the aircraft against a set of representative tactical flight regimes using well defined workload requirements that make flight tests repeatable and comparable between several evaluation test pilots. Thus, aircraft limitations revealed by pilot-induced oscillations (PIO), rotor speed excursions and performance degradation can be documented and tests are repeatable.

The Canadian Forces Utility Tactical Transport Helicopter (CFUTTH)

A contract was awarded by the Canadian Department of National Defense to Bell Helicopter Textron of Canada (BHTC) on 9 September 1992 for the purchase of 100 BELL 412CF. The project was valued at $ 1,291 Billion canadian dollars and included a full motion mission simulator subcontracted by BHTC to CAE, a canadian worldwide leader in simulator technology. The intended roles for the new helicopter were as follows:

a. Utility Transport; and

b. Combat Support (Search and Rescue).

The first aircraft delivery was scheduled for early 1995 to replace the aging fleets of Kiowas (BELL 206 equivalent), Single Hueys (BELL 205 equivalent) and Twin Hueys (BELL 212 equivalent).

The BELL 412CF is a twin-engine helicopter equipped with a four blade, soft-in-plane rotor system. The 1600 shaft-horse-power powerplant consists of a twin-pack PT6T-D made by Pratt and Witney Canada. The control system includes a three-axis Honeywell Dual Digital Automatic Flight Control System (DDAFCS) which provides Stability Augmentation System (SAS) and attitude retention. Coupling capabilities using altitude and airspeed sensors and through navigation aids are also available thanks to a flight director integrated to the DDAFCS.

Other than several military specific mission kits, the major difference between a BELL 412CF and previous 412's lies in the Avionic Management System built by Canadian Marconi. This system allows access to the communication and navigation systems using two fully redundant Computer Display Units (CDU's).

The first flight of the BELL 412CF occurred on 30 April 1994. FAA certification in accordance with FAR 29 was issued in February 1995 and deliveries to the squadrons started in April 1995.

Since the program made use of the modular approach, military specifications were not invoked and thus demonstration of compliance to these specifications by the contractor were not required. This not only saved time but reduced program costs. The spirit of these specifications, however, were not ignored and were used as guidelines to determine the "delta" of tests required to be performed by the AETE in order to ensure that the BELL 412CF was capable of safely fulfil its intended military role.

Consequently, an interim tactical flight clearance was issued in less than 90 days after delivery of the first aircraft to AETE.

Flight Testing Requirements

The "delta" flight test requirements for the CFUTTH program follow in order of priority:

a. Tactical Flight Clearance;

b. NVG;

c. Mission kits without FAA Supplemental Type Certificate (STC);

d. Stores clearance;

e. Limited navigation systems proof of compliance;

f. Limited communication systems proof of compliance;

g. Limited performance proof of compliance; and

h. Electromagnetic Compatibility and Interference (EMC/EMI) baseline documentation.

Only these topic areas were covered for this flight test program since they were determined to be the only relevant areas not covered by FAR 29 certification or for which getting data was important to ensure that the aircraft was capable of conducting its intended Canadian military role.

The EMC/EMI baseline was determined important since past experience has proven that modification and addition of new mission kits will occur in future years. The EMC/EMI baseline would therefore reduce future program costs.

A period of 7 to 8 months is estimated to complete all of the above CFUTTH testing.

Tactical Flight Clearance

A tactical flight clearance without a flight envelope is meaningless. The tactical flight envelope of a helicopter is based on two parameters: altitude and speed. For efficient tactical flight, one needs to flight low and fast. There are two main limitations involved in the determination of the tactical flight envelope: aircraft survivability during the failure of a flight critical component and crew workload.

The tactical flight clearance of the CFUTTH required testing in three major areas:

a. Preliminary flying qualities and handling qualities testing;

b. Determination of a tactical flight envelope based on failures; and

c. Determination of a tactical flight envelope based on crew workload.

The above testing was conducted for the clean configuration only (no external mission kits) in clear visual conditions (no night vision imaging systems). The aircraft was not instrumented. When data requirements existed, standard aircraft instrumentation and measuring tapes attached to the flying controls were used. The lack of precision in the data was not objectionable since the collected data was used to confirm aircraft stability and control margins rather than document the baseline.

Preliminary Flying Qualities and Handling Qualities Testing

These tests were the first contact of the test team with the aircraft. The main objective of the preliminary flying qualities testing was not to gather baseline data – because of the lack of instrumentation as discussed previously – but rather to allow the test team to become familiar enough with the aircraft flying qualities to carry on with the rest of the evaluation. As such, these tests were labelled Qualitative Evaluation even though some quantitative data were collected.

The flying qualities of the aircraft were assessed using the following standard flight test techniques:

a. Apparent Longitudinal Static Stability;

b. Trim Changes With Power;

c. True Longitudinal Static Stability;

d. Steady Heading Sideslip;

e. Turns On One Control (Lateral Cyclic and Pedals);

f. Dynamic Longitudinal Stability;

g. Lateral-Directional Dynamic Stability;

h. Control Response Testing;

i. Manoeuvre Stability; and

j. Low Airspeed Testing.

Flying qualities tests were flown to the greatest extent possible of weight and centre of gravity (CG). Special attention was given to worst case combination of weight and CG documented by the manufacturer. Flying qualities tests were flown in all DDAFCS modes: SAS "OFF", SAS "ON"and in attitude retention mode. The results of the flight test techniques showed that the aircraft was generally acceptable in all areas and that testing of handling qualities could be pursued.

Handling qualities were assessed using ADS-33C as a tool rather than a pass/fail test. The following most representative mission task elements were selected from ADS-33C as they were considered the most suitable for the aircraft role:

a. Precision landing;

b. Precision hover;

c. Pirouette;

d. Rapid sidestep;

e. The acceleration and deceleration;

f. Hover turn;

g. Rapid slalom;

h. Deceleration to dash;

i. Bob up and bob down;

j. Vertical remask; and

k. Slope landing.

The mission task elements were conducted by three separate test pilots who rated the aircraft using the Cooper-Harper Handling Quality Rating Scale. Conducting the manoeuvres as defined in the ADS 33-C lead the CFUTTH to be rated acceptable (level 2) according to the specification. The benefits of this testing were not limited to the end result but rather to the discovery of several aircraft characteristics in the area of rotor governing, power margin, and control response that could only be seen during mission tasks involving significant workload.

The Preliminary Flying Qualities and Handling Qualities Testing phase required 16 sorties for a total of 38 flying hours over a period of three weeks. It was conducted before the first aircraft was delivered to the military thus no real impact on the project timeline resulted.

Determination of a Tactical Flight Envelope Based on System Malfunctions

When considering critical aircraft component failures, two factors come to mind: the probability that a failure occurs and the consequence of the failure occurring. There are tradeoffs between these two factors: if the probability is high, then the consequence should be minor; conversely, if the consequence of a failure is high then the probability of it occurring better be low. When the probability is unknown, one has no choice but to investigate the consequence thoroughly.

Preliminary analysis of critical aircraft component failures revealed that the only system requiring investigation was the DDAFCS since the data on the probability of occurrence of this type of failure was not available to the test team. What needed to be determined was the speed and altitude envelope that would make a safe recovery of the aircraft possible should a failure occur.

Technical review of the DDAFCS system allowed the test team to devise an in-flight simulation procedure that could be used at a safe altitude to evaluate the consequence of an DDAFCS malfunction. A time and space positioning system allowing 2 ft precision in position and 2 ft/sec. in velocity was used to determine the flight path deviation caused by the failures at airspeeds from 40 to 100 kts in level flights and during banked turns. Analysis of results lead to the determination of a minimum safe altitude and maximum airspeed for tactical operations.

The Determination of a Tactical Flight Envelope Based on System Malfunctions phase required 8 sorties for a total of 16.3 flying hours were required for this testing.

Determination of Tactical Flight Envelope Based on Crew Workload

The tactical flight envelope based on crew workload was determined through the conduct of simulated mission scenarios within the previously determined tactical flight envelope. The aim was to have the crew conduct all the flying, navigation, communication and mission management tasks associated with a tactical mission in normal and degraded modes for failures of non flight critical mission systems (radio, CDU and GPS failures).

An incremental build-up approach was employed. Testing started with ground based cockpit and systems assessments, followed by in-flight evaluation of individual systems, and finally by an integrated evaluation under simulated mission conditions. The in-flight evaluation of individual systems was conducted at non tactical altitudes prior to an evaluation at tactical altitudes. The evaluation included the following parts:

a. ground based evaluation of cockpit and crew/system interface;

c. low level 250 ft AGL navigation using GPS and Doppler Navigation System (DNS);

d. tactical navigation using GPS and DNS; and

e. integrated mission scenarios of combat mobility and fire support coordination mission.

The test criterion for this evaluation was that the crew members must not be compromised in their ability to perform critical aircrew duties, including lookout, obstacle detection and avoidance, and monitoring of systems malfunction annunciation. The crew communications and situational awareness were not to be objectionably degraded and there was to be no interference with the safe operation of any flight critical systems or components.

Our test showed that the crew workload was acceptable within the tactical flight envelope determined during the system malfunction simulation. No further restriction on minimum altitude and maximum airspeed had to be imposed.

The Determination of Tactical Flight Envelope Based on Crew Workload phase required 16 sorties for a total of 35.5 flying hours were required for this testing.

Conclusion

The procurement of a FAA certified aircraft provided a fast and effective approach to fulfil our requirements for the utility tactical transport helicopter. Less than three years was required to go from contract award to the an operational helicopter in the tactical environment. A total of 40 sorties (89.8 flying hours) was required to generate the data in supplement of the FAA certification for tactical operations. Additional testing for unique military requirements, specification compliance and baseline is estimated to 8 months.

Recommended Approach for Aircraft Acquisition Program

The following simplified steps are recommended as an approach to aircraft acquisition based on our experience as the flight test agency for the CFUTTH program:

a. First step: identify specific operational requirements for the aircraft;

b. Second step: determine where FAA certification falls short of military requirements: what data are missing? The military specifications can be consulted for guidance in the preparation of a flight test program; and

c. Third step: include the additional data requirements in the acquisition contract. For example, stipulate that the manufacturer is to provide the tactical flight envelope or proof of compliance data where deemed necessary.

In this approach, the onus is on the manufacturer to deliver an aircraft which is specification compliant and includes the additional requirements over and above FAA certification.

Recommendation for a Qualitative Evaluation

A qualitative evaluation or "preview" of the aircraft prior to letting contract can provide enough data in advance to demonstrate suitability for the military role by focusing on items over and above FAA certification requirements. The information gathered from the qualitative evaluation can aid in determining proof of compliancy issues as well as the scope of additional testing required above and beyond FAA certification. The qualitative evaluation must include handling qualities testing while performing mission representative tasks. Our experience showed that 38 hours of evaluation provided a significant amount of data with no impact on the project schedule.

AIRCRAFT TECHNICAL OPERATING CAPABILITIES ENSURING AND EVALUATION EXPERIENCE

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Aviation development for the last 15 years showed the great significance of aircraft technical operating capabilities (TOC — flight safety, reliability, testability, and maintainability) ensuring at all stages of life-cycle, besides high aircraft performance, to achieve economical efficiency of operations, high level of safety, and despatch reliability.

It could be confirmed by a classical optimization interconnections chain (fig. 1). Indeed, a growth of requirements to the composition and quality of the tasks to be solved by an aircraft and economic efficiency of its operations complicates the aircraft and its systems' design, and influences the failures rates and effects. To maintain the required reliability and safety levels there should be taken measures to enhance the system components reliability, redundancy, warning and indicating means, and special safety ensuring systems (both, forewarning the critical flight modes, and reimbursing the hazardous conditions).

On the one hand, all these measures affect directly the aircraft life-cycle cost (through the systems' costs, aircraft weight, etc.), and on the other hand they stipulate new requirements to the aircraft maintenance system (AMS) and operating test equipment. These new requirements cause problems of maintainability and testability that have to be solved; development of new or improvement of existing ground support equipment (GSE), on-board test equipment and flight data recorders (FDR); ground automated test equipment (ATE) and non-destructive inspection (NDI) techniques; elaboration of more effective maintenance programs to reveal hidden failures of redundant components and prevent aircraft systems' failures evident to the crew. This helps to ensure required reliability and safety of the aircraft, though it affects considerably the despatch reliability and operating cost due to increase in labour spent for the troubleshooting in complicated redundant systems, probable human errors, and problems with reducing the "re-test OK" rates in complex aircraft systems.

Thus, aircraft performance enhancement results in life-cycle cost rise and leads to a new cycle of working out advanced requirements to all the aircraft characteristics. That is why the idea of systematic or complex approach in TOC ensuring and evaluation appears to be very attractive and having become a prevalent one in the aviation community for the last period of time.

The Lietno-Issliedovatelsky Institute named after M.M.Gromov (Russian abbreviation M.M.Gromov LII) or the Flight Research Institute has gained for the past 30 years great experience in developing and successful implementing the methodology for ensuring and integrated evaluation of TOC at the aircraft design, testing, certification, and operation.

The Institute's 4th Division responsibilities include development of requirements, regulations and methodological basis, expertizing and certification in following fields of activities (fig. 2):

— methodical support and participation in aircraft flight safety and systems operation safety assessment, failure mode and effect analysis, testability and maintainability analysis, and certification of these characteristics;
— evaluation of available warning devices and indicators completeness in abnormal flight situations, methodical support of accident investigation;
— nondestructive testing, investigations into failed or damaged components and assemblies;
— flight data and cockpit voice recorders (FDR/CVR) evaluation/certification, software development for automatic flight data processing and "quick-look" analysis; testing, development, and verification of flight data processing software;
— development and implementation of computerized TOC evaluation methods at all phases of aeronautical product development and operation;
— ground and flight test procedure for the on-board automated test and diagnostics systems, and their certification;
— methods for the flight data retrieval during aircraft accident cause investigations;
— methods of rational maintenance programs development and certification;
— organizational and methodical support of maintainability testing, GSE and aircraft ground equipment (AGE) evaluation and certification;
— ground power source electric current quality verification;
— methods and procedures for GSE testing, methodical support and participation in rational GSE sets development for the aircraft classes and particular aircraft types.

Initially Division was founded as a Reliability Laboratory to resolve problems emerged in the 60th in our aviation due to an increasing failure rate of more complicated aviation hardware. However the integrated approach to the TOC consideration gained its end and four research Laboratories were established into the Division engaged in flight safety, reliability, maintenance and operating test problems.

At the beginning safety problems were being solved by standardization of safety procedures and accident investigation techniques based on the FDR data. Later the "quick-look" analysis systems were developed and implemented for the flight crew actions monitoring at the critical flight stages in order to prevent the flight standards and techniques violations.

It is common knowledge that air accidents occur mainly due to an unfavorable combination of several faults of the air transportation system. The combination may include events of different nature and a combination of six factors results in an accident, on an average. In this case, as aviation equipment has improved, the share of aircraft system failures causing an accident has been continuously decreasing, and human factor influence has been increasing. For the last 35 years there were lost over 300 gas turbine engined aircraft in air accidents: 82% occurred due to operation faults, and among them 75% were caused by crew errors. And most crew error caused accidents occurred with no faults of the aircraft systems or air-traffic control (ATC) service reported.

If we take all crew error caused accidents as 100%, then the crew faults in piloting will make about 90%, but crew errors combined with ergonomic and other factors — a little more than 10% (fig. 3).

To prevent the errors indicated above, LII in collaboration with TsAGI and GosNII GA have developed and inculcated special techniques for "quick-look" or express-analysis of flight data recorded by a standard aircraft FDR in operation.

Processing and automated analysis of data are carried out on special ground-based facilities Luch-74 and Luch-84, as well as on standard IBM PCs provided with specialized interfaces. More than 200 airports in Russia are equipped with such systems. The FDR data analysis has helped prevent many accidents since the crew errors were revealed in good time. Besides, flight data express-analysis promoted improvement of the Flight Manuals recommendations for the Tu-154, Yak-42 and other aircraft.

At the end of the last year the Russian — American Working Group on assessing our air safety system drew attention to the system indicated above as the most significant component which could be employed as a base for decreasing human factor effect upon flight safety of future aircraft [1].

Another important area of flight safety investigations is an economically substantiated combination of measures aimed at improving safety. Many safety systems (TCAS II, GPWS. etc.) are expensive and not always effective. A great role is played by ways and means of flight crew training, as well as by economic and social conditions under which the air transportation system is operated.

The USSR disintegration and subsequent changes in the social and economic spheres have brought about the emergence of new factors affecting flight safety, including:

- destruction of the established normative technical and legal base of aircraft development, testing and operation; and unjustified imitation of foreign regulatory approaches to these processes;
- wash out of the industry's research Institutes role in providing methodical and scientific support of Design Bureaux (references to foreign experience are not competent, since our Design Bureaux have never developed their scientific, engineering and research facilities up to levels of big foreign aviation corporations just because of industry's Institutes existing, and our Design Bureaux are not capable of handling appropriate questions);
- aggravation of the situation in the industry due to lack of funding: poor runways conditions; lack of hangars for maintenance; lack of checking and diagnosing equipment; irregular deliveries of spares for engines; shortage of aircraft parking area (resulting in congestion and damage of aircraft by service devices, etc.) and run-up grounds (which increases FOD rate); obsolete forms and technologies of aircraft maintenance;
- increase of dependence of flight crews on ground services and ATC service, which leads to a growth of air accidents caused by overloading, improper fueling (insufficient fueling, non-observance of emergency and en-route fuel allowances), captains' refusals to go to an alternate airfield (where one could stay for ever or fly back robbed);
- decrease of training flying hours and simulation training time; negative effect of paying pilots depending on their flying hours which makes it possible to exercise pressure to bear upon aircraft captains (flying inadequately prepared aircraft; overloading, etc.); a decrease of the number and quality of inspector flights.

This is just a general review of flight safety problems for our Institute to be partially responsible for as the leading organization in TOC area of the Russian aviation industry.

Reliability problems study began with development of a theoretical basis for designing aircraft with the flight mission completion success probability prescribed, standardization and inculcation of reliability assessment at design, test and operation stages. Then there were worked out applications of reliability theory to aircraft certification in respect of evaluation of probabilities and consequences of different failure combinations of complicated aircraft systems.

At present two interconnected directions of reliability investigations may be indicated:

— feasibility study of alert levels of aircraft reliability and measures to provide them;
— analysis of physical causes of items failures to select measures of improving reliability.

The first type of research is aimed at improving the normative base of providing civil and military aircraft reliability and customer support system as regards aircraft delivery and acceptance conditions, guarantees and warranties, provision of spares, etc., which is to be dwelt upon further. The point is a transition from the centralized system of developing measures to eliminate all the system failures revealed in operation to a system of economically substantiated ways of checking reliability and warning of only dangerous and affecting the aircraft operation efficiency faults, with faults effects and their prevention measures costs being comparable.

The second type of research is directed towards flight and bench research to evaluate the influence of structural and external factors on aircraft reliability. Thus, for the last years our Division's specialists have done work to study such factors' influence upon engines, avionics units and airframe components, which I would like to deal with in more detail.

Aircraft operation and investigations show that the major factor influencing reliability, avionics, in particular, is vibration. As analysis of a great amount of information on home-made and foreign aircraft, concerning avionics malfunction causes, indicates that vibration makes up 27—30 % of failures and malfunctions.

Avionics vibration may be caused by, for example, aerodynamic stalls and air-flow fluctuations in going over an aerodynamic surface. It will occur in cases when their setting angles are very close to the limit values due to insufficient accuracy or errors made while the wing was assembled or their maladjustment in aircraft operation.

To study this factor there were conducted, with use of a production maneuvering aircraft, flight investigations of the influence of the outboard wing leading edge slat and aileron setting angles deflections upon the level and spectrum of avionics vibration. The latter was recorded on the port wing tip and avionics, mounted in the behind-the-cabin section without shock absorbers and on the shock absorbers, as well as in the compartment. As a result, there were obtained. for every place of avionics location, two-factor regression dependencies connecting the levels of the vibratory loads and setting angles deflections of the outboard wing slat (X1) and aileron (X2), and indicating that the outboard wing slat setting angle changes produce a greater influence upon the vibration in all the avionics zones of an aircraft. With this angle deflection increased by 2 degree, the root-mean-square value of vibroacceleration (RVA) in the avionics zones increases almost 1.5 times (fig. 4).

The regression dependencies obtained can be used to forecast vibration levels in the avionics location zones for the range of the leading-edge slat and aileron angles deflections studied, as well as to check the wing assembly accuracy and angles of surfaces setting adjustment in operation. Similar results and practical recommendations were obtained as regards propellers, landing-gear wheels (in respect of their balancing and setting angles), bladed wheels of aeroengines, etc.

The work on test equipment development started with scientific and methodical support provision for creating ground automated check systems to improve the combat capabilities of military aircraft and the first national flight-data magnetic recorder still in use. Then, in the late 70s there were put into service ground-airborne flight data acquisition, automated processing and analysis systems, with the faults of the system operation and crew actions being documented. These systems were used to check the piloting techniques mentioned above.

Nowadays, the research in this area covers a wide range of problems, each deserving a special discussion, so the main ones are as follows:

— creation of stand facilities for investigating and testing airborne and ground automated test equipment;
— improvement of evaluation techniques and provisions of aircraft system testability, digital systems included;
— improvement of NDI methods and test techniques for failed equipment;
— development of techniques of retrieving the flight data information from the damaged magnetic recorders media;
— improvement of standards and stand facilities to ensure safety of the flight data crash-recorders in accidents;
— elaboration of advanced conceptions of airborne data acquisition systems, their new products testing and certification.

In the area of maintenance improvement the Division started with evaluation of aircraft in service, unification of their servicing points connections and GSE, of the maintenance schedules for different aircraft types to provide common bases and enhance the combat capabilities. Later on, the Institute made a great contribution to aircraft testing and certification in respect of maintainability and GSE characteristics and adoption of on-condition maintenance in our aviation.

Maintainability includes such characteristics as accessibility of maintenance objects, their replaceability, interchangeability, etc. These properties are directly tied to the structure parameters, volumetric configuration and they are laid down at the development stage. Analysis of maintainability deficiencies, revealed through testing and operation of different types of aircraft, shows that a selection of an effective "configuration parameters — maintainability characteristics" interrelation has been a difficult task in developing several generations of aircraft. Even forth-generation aircraft have drawbacks, the relative number of which is over 80%. We may mention low accessibility of individual units, assemblies and avionics which were arranged without considering the rate of maintenance events and reliability of maintenance objects. For example, there are well-known problems with oil-filling of the cooling turbine of the MIG-29. when to provide access for refilling, usually requiring 30 minutes all in all, it took about 8 man-hours of additional removing work.

With reference to the rational configuration selection of the aircraft compartments, the mean time to repair (MTTR) and its elements must be specified concerning the system components to be arranged in them. Our specialists have developed methods of forming quantitative requirements to the MTTR. These methods have been approved in practice and may be employed in the integrated data bases of aircraft computer-aided design systems, specifying the indicated values for systems and their components on the base of the prescribed requirements to the aircraft as a whole. At the same time, there are questions of local configuration influence (near-by ducts, braids, structure components, neighboring assemblies, etc.) on the replacement downtime arise, which require further development.

In the past years it has been of significance to improve technical and economic characteristics of maintenance, to introduce advanced types of GSE, to inculcate new techniques of maintenance programs development and certification, and to improve aircraft operation documentation.

It should be pointed out that under the conditions of hard competition at the air transportation market development of the maintenance program which provides for optimization of system primary maintenance processes (PMP), maintenance and repair tasks to reduce the aircraft operational costs and to maintain its airworthiness and competitiveness.

The experience of the aircraft operations shows that one of the dominant ways to reduce maintenance cost when ensuring the necessary level of flight safety is wide implementation of the on-condition maintenance in order to keep the inherent aircraft reliability and airworthiness at minimum labour and material expenses on maintenance. This principles in the world practice have received the name of the RCM (Reliability-Centered Maintenance). RCM-analysis principles, first developed by F.S.Nowlan [4] have greatly evolved. Our country's experience in elaborating aircraft operation and certification documentation, in respect of maintenance operating capabilities and safety of aircraft systems, emphasized the necessity of further movement in this direction, of developing proper techniques based on RCM technology.

LII and Design Bureau worked-out methodology which was used to develop the Il-96-300 airplane and some other aircraft maintenance programs, is allows formal analyzing of the influence of possible systems and components failure modes on safety, despatch reliability, and economical efficiency that provides the substantiated PMPs and maintenance tasks selection in order to keep the inherent levels of reliability and airworthiness at the aircraft operations.

Development of the maintenance programme for the Il-96-300 aircraft was carried out based on the above mentioned methodology from the early stages of the aircraft design with the scientific and engineering support and practical participation of the specialists from the leading research Institutes of the aviation industry and civil aviation. The Design Bureau of the S.V.Ilyushin AK has organized the special department in the "IL" Product Support Division whose specialists for a year and a half have performed the analysis of all basic airplane system.

All the system components without division into "significant" and "non-significant" as it is prescribed by well-known document ATA MSG-3 [5] were analyzed. Generally, about 800 types of maintenance objects (assemblies, subassemblies, parts, and components) were considered. The volume of the substantiating documentation is about 3,000 pages, and the analysis general man-hours finally exceeded nearly 30,000. The analysis results permitted reduction of the IL-96-300 maintenance volume by 30% compared to that of the IL-86 and showing the compliance with the world level. To continue this job actual maintenance working hours and downtime were evaluated in the airplane testing; maintenance programme and other operational documentation were certified. Activities were provided for:

— substantiation of the design estimations and analysis results obtained from the above-mentioned methodology;
— evaluation of the possibility to accomplish all the scheduled and basic unscheduled maintenance tasks prescribed by the Maintenance Program, which was performed by practical accomplishing the maintenance tasks in accordance with the technological instructions outlined in the Maintenance Manual;
— substantiation of the impossibility of the improper mounting (de-mounting) and installation of the system components through the evaluation of the design features, the availability and quality of necessary markings and placards;
— evaluation of the operational documentation statements formulation unambiguity and clearness.

Most important directions of evolving the methodology developed are:

— improvement of the optimization methods for maintenance tasks intervals;

— improvement of the software for maintenance program development automation in the framework of aircraft computer-aided design;
— introduction of the national methodical aids in the world practice within the next revision of ATA MSG-3, which there has been reached an agreement on with the MSG-3 Team Manager, ATA Director for Maintenance and Material Steven R.Erickson;
— improvement of methodology to be used in other industries; the experience gained indicates that it can be applied to creating most complicated systems (cars, ships, locomotives, etc.) with a cyclic operation, allowing a formal description of system potential failures and associated risks to be made.

Recently of topical interest has been the work, started earlier, to investigate human factor influence on maintenance effectiveness. The studies conducted showed that one third of air incidents with civil aircraft are caused by technicians' errors. For 100,000 flights there are 4—5 technicians errors, and 70% of them are due to the lack of aircraft maintainability. These problems are becoming urgent owing to lack of qualified aviation specialist and development of their training techniques. For example, American specialists have just published a computer manual on human factor in maintenance, worked out with the multimedia technology.

Summing up the above-said we can turn to fig. 5 which reflects, through the dynamics of the main indications, the results of the work in TOC area and scientific and engineering potential in the industry as regards the questions under discussion.

At the end of this short review of our Institute's experience in the area of providing and evaluating TOC during over 30 years of our Division's history I cannot but deal with the work which is indispensable for future development of a TOC evaluation system for advanced aircraft. This system is to provide standards in the form of two-level documents (fig. 6):

1) normative and legal provisions — laws and other state regulatory acts;
2) technical regulatory provisions — standards, industry and inter industry normative and methodical documents (general requirements, manuals, techniques, specifications, etc.). And also should be the third group of working documents (certificates, licenses, contracts, reporting forms, etc.).

TOC ensuring and monitoring functions include both technical and managing activities. The last is deals mostly with the first level documents, and the technical — mostly with the second.

Normative and legal documents must provide implementation of the following economic measures and organizational principles:

- differentiation of working conditions for government orders and independently funded developments with the government's order being made profitable, and of first priority;
- selection of a contractor on a competition basis to fulfil the government order (with enterprises of all forms of ownership participating); legitimate establishment of tax privileges to be granted to enterprises engaged in adopting new advanced technologies and products, provision of support to effective "small businesses" in aviation;
- government measures in regulating the aircraft development and manufacture, deliveries of raw materials, components and materials in respect of providing, in the first place, aircraft reliability and safety;
- inclusion in the contracts and orders clauses defining the enterprise's responsibility for providing both performance and TOC, that may be implemented as an extended system of supplier guarantees and warranties;
- guarantees by the government, when its order is being accomplished, of keeping up the agreed maximum prices of raw materials, and products during the period of the contract, with the government audit and control over the materials usage;
- penalty measures and sanctions, or other forms of responsibility in case of non fulfillment or break-down of the government's order accomplishment;
- government bodies' just general regulating activity as regards the enterprises' work to provide the aircraft quality and its approval for operation, with the role of non-governmental bodies of manufacturers and operators growing based on interrelated laws/regulations, requirements, methods and contractual obligations;
- setting up of special mixed organizational structures (committees) financed by operators and suppliers, with the government participating, which could control the provision of aeronautical products performance and TOC when aircraft are developed and operated.

Some of the legal documents indicated already exist or existed before and are being reviewed now (Air Code, Quality Provision Laws, Law of Products and Services Certification). Other documents have to be prepared anew. Now national aviation regulations (AP) are being developed on the analogy with the American Federal Aviation Regulations (FAR). National aviation rules must be based on two principles:

1) a considerable reduction of the amount of standards and regulations (similar to the approach of the European committee — JAA);
2) use of the existing effective normatives and regulations concerning aircraft operation and TOC.

Fig. 7 offers an AP structure which will meet the national requirements and JAA conception, the only question that has to be considered is whether there should be separate parts (AP-91) and (AP-121) or both combined in one document (AP-OPS) like the European Requirements JAR-OPS comprising three parts: Part 1 — Aircraft Operations. Commercial Air Transportation; Part 2 — Aircraft Operations. General Aviation, and Part 3 — Helicopter Operations. Commercial Air Transportation. The structure given should not cause any objections on the part of FAA specialists when we sign an Bilateral Airworthiness Agreement, as the main points of FAR are sure to be taken into account. And we may count on our country to be integrated easily in the world community in respect of cooperation in the areas of aircraft operation and TOC.

As to the second group of documents (ref. fig. 6), they must provide elaboration of several general requirements (in particular, final development of the existing General Requirements for Civil Aircraft TOC [2]; manuals (Manual for Maintenance Program Development based on RDK-E Manual [3] and ATA MSG-3) for designers and operators: methods and specifications — on the analogy with the world documents ATA, AEA and ARINC Specifications. Here belong the Manuals containing interpretations and procedures of AP usage, conforming to the Orders and Manuals of FAA.

In addition there must be further developed the Regulation on Civil Aviation Products Development and General Conditions of Aircraft Delivery. The latter may be combined in one document with General Requirements to Airlines Suppliers. Competition-based selection of a contractor and government's contract fulfillment should be regulated by separate documents (e.g. Orders of government or industry levels) and should include the appropriate orders and legal acts.

The third-group documents must be reasonably adopted in compliance with the world practice, with the existing certificates being included in them (TC-Type Certificate; AC-Airworthiness Certificate; PC-Production Certificate; SCA-Standard Certificate of Airworthiness; APL-Aircraft and Powerplant License and Standard Contracts agreements).

These activities will conform, on the whole, to those adopted by FAA and JAA, whose experience has been borrowed as shown in fig. 6 (in brackets there are indicated their organizational structures — FAA, JAA, PCT, NBAA, AIA, ARINC, MRB, FOEB, FSB; normative and methodical documents — FAR, JAR, as well as ATA and ARINC Specifications (SPEC), Manuals and Guides (GUIDE), Requirements (REQ), and ATA MSG-3).

Acknowlegements

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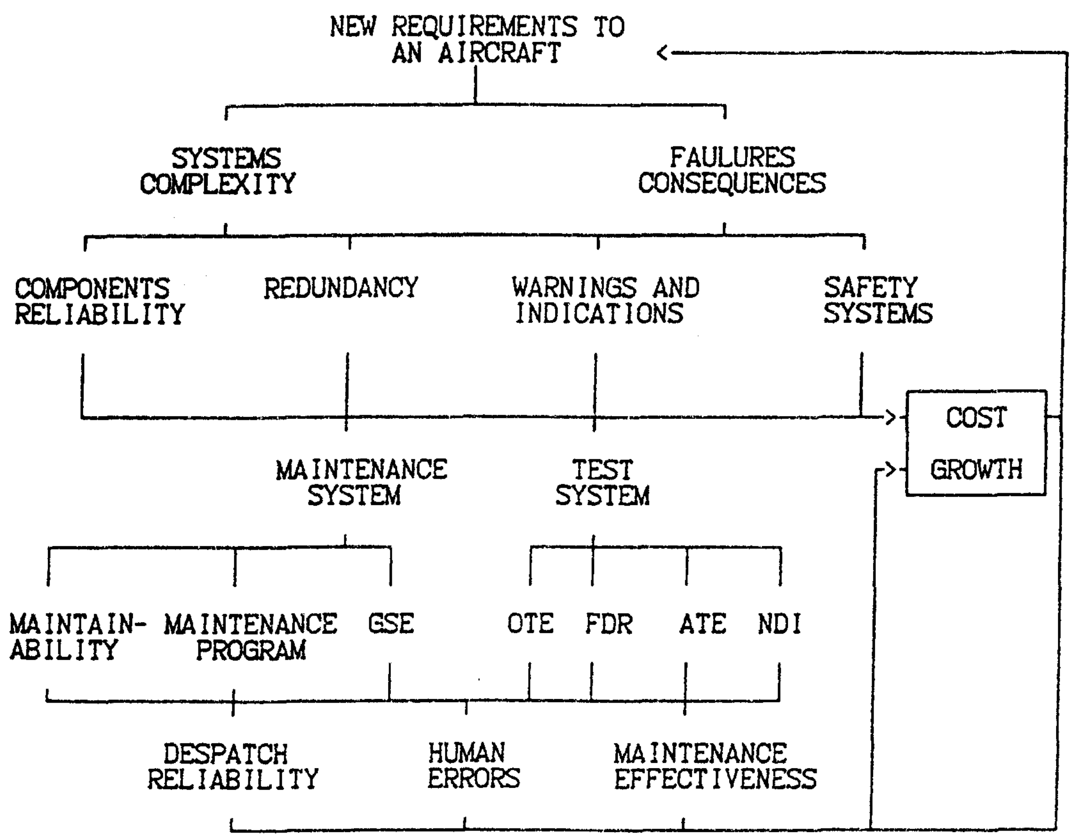


Fig. 1. Technical operating capabilities effect upon aircraft performance optimization.

TECHNICAL OPERATING CAPABILITIES (TOC) DIVISION OF THE
M.M.Gromov LII

TOC include:

FLIGHT SAFETY, RELIABILITY, MAINTAINABILITY AND TESTABILITY, CHARACTERISTICS OF GROUND SUPPORT AND TEST EQUIPMENT, ON-BOARD TEST AND FLIGHT DATA RECORDING EQUIPMENT

MAIN FUNCTIONS:

* DEVELOPMENT OF METHODOLOGY AND STATE AVIATION INDUSTRY ACTIVITIES COORDINATION IN THE FIELD OF AIRCRAFT TECHNICAL OPERATING CAPABILITIES

* TOC LEVELS ANALYSIS AND PROGNOSIS, REGULATORY DOCUMENTS DEVELOPMENT AND EXPERTIZING

* FUTURE TECHNOLOGIES STUDY, AIRCRAFT TOC LEVELS AND GSE/ATE/FDR/CVR TESTING AND CERTIFICATION

* METHODICAL ASSISTANCE FOR THE DESIGN BUREAUS, EXPERTIZING, CERTIFICATION, OPERATIONS EXPERIENCE ANALYSIS, RECOMMENDATIONS DEVELOPMENT

Fig. 2. Main Institute's responsibilities in the field of aircraft technical operating capabilities.

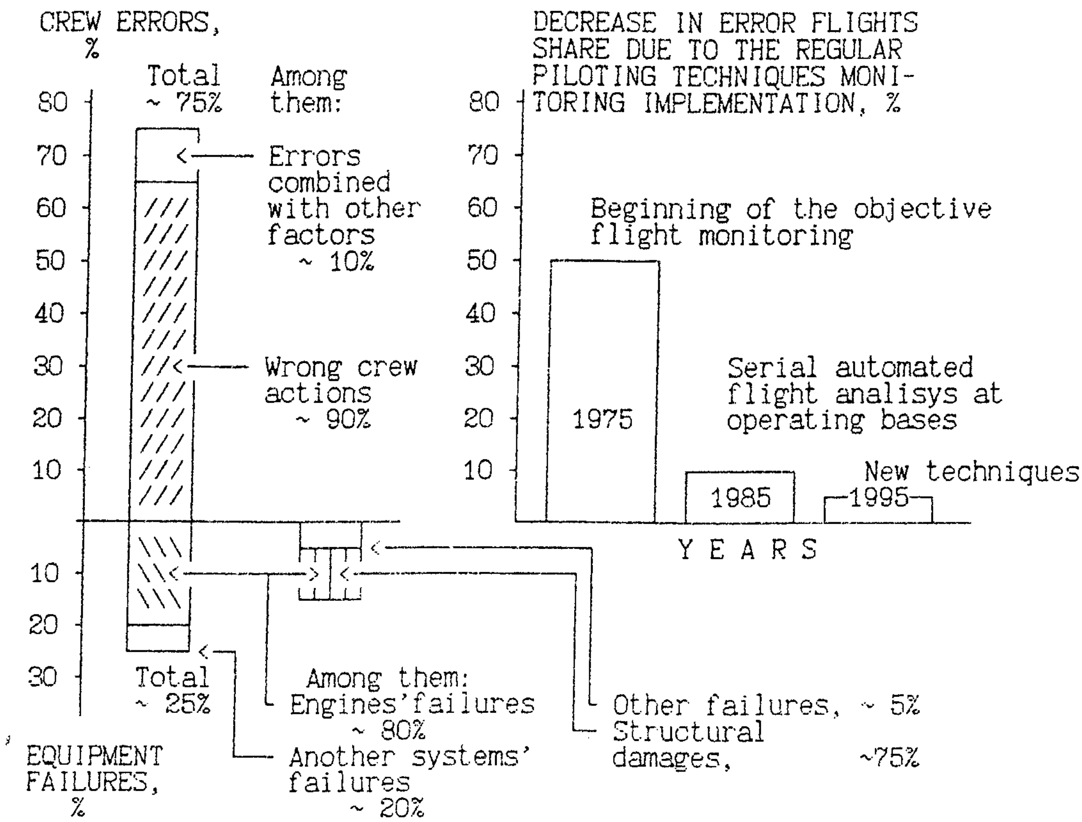
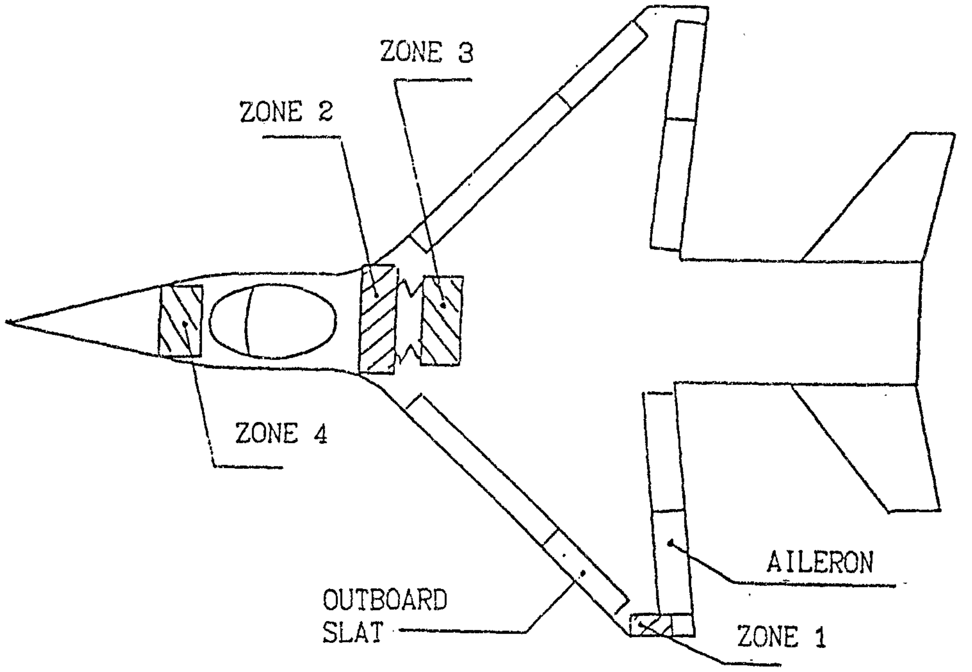


Fig. 3. Basic causes of accidents and objective flight monitoring effectiveness.



AIRBORNE EQUIPMENT VIBRATIONS REGISTRATION ZONES

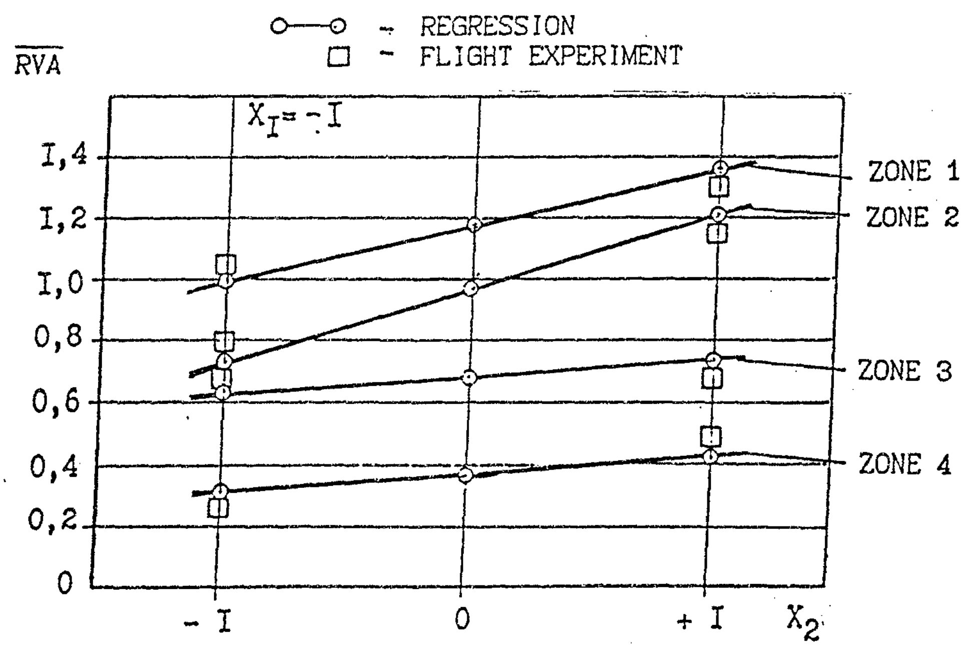
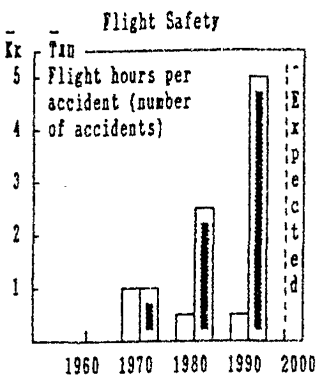


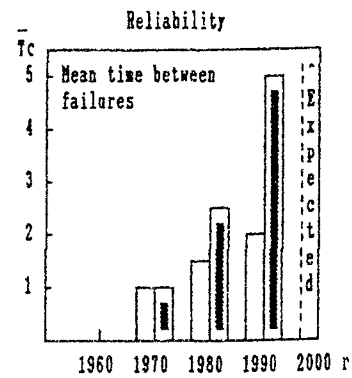
Fig. 4. Regression dependences for relative vibroaccelerations (RVA) vs trim angles of aileron (X2) and outboard slat (X1)

LII OBJECTIVES AND FUNCTIONS:

* Development of methodology and tools for TOC ensuring
* Future technologies study, aircraft TOC levels and GSE/ATE/FDR/CVR testing and certification
* Coordination and methodical assistance for the Design Bureaus, expertizing and recommendations development



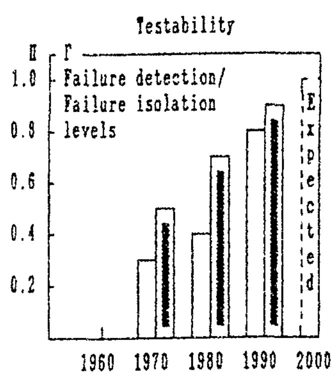
1968 - Establishment of the Flight Safety Research Laboratory



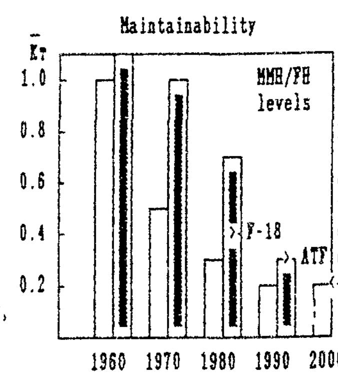
1962 - Establishment of the Reliability Research Laboratory

WORKS BASIS AND ORGANIZATION:

* Operating aircraft fleet monitoring
* Testing of all the types of aircraft
* Specialized Divisions for each system evaluation
* Division for systematic TOC study, which includes Safety, Reliability, Maintenance and Test Systems Laboratories



1968 - Establishment of the Test Equipment Research Laboratory



1972 - Establishment of the Maintenance Systems Research Laboratory

LEGEND: ▯ - civil aircraft; ▮ - military aircraft

Relative values showed

Fig. 5. Main directions and results in the field of aircraft technical operating capabilities improvement.

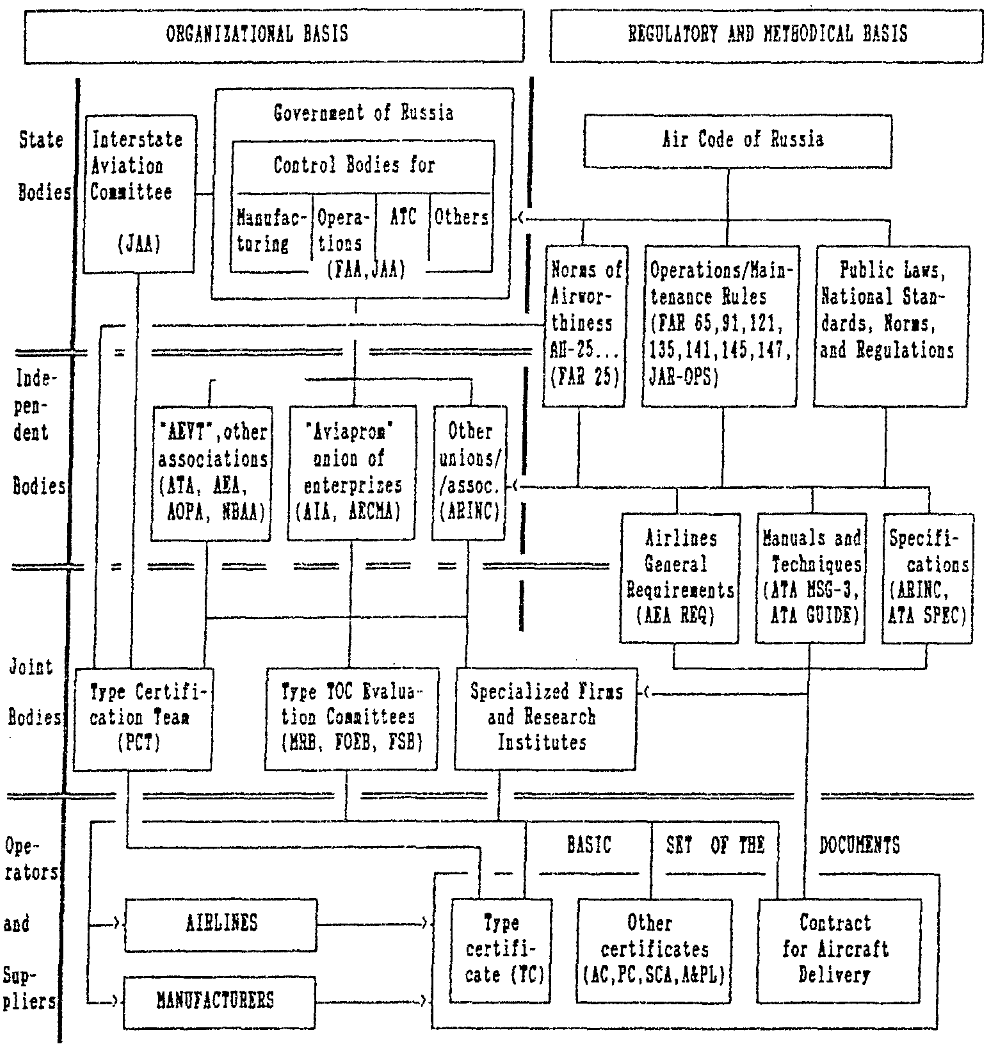
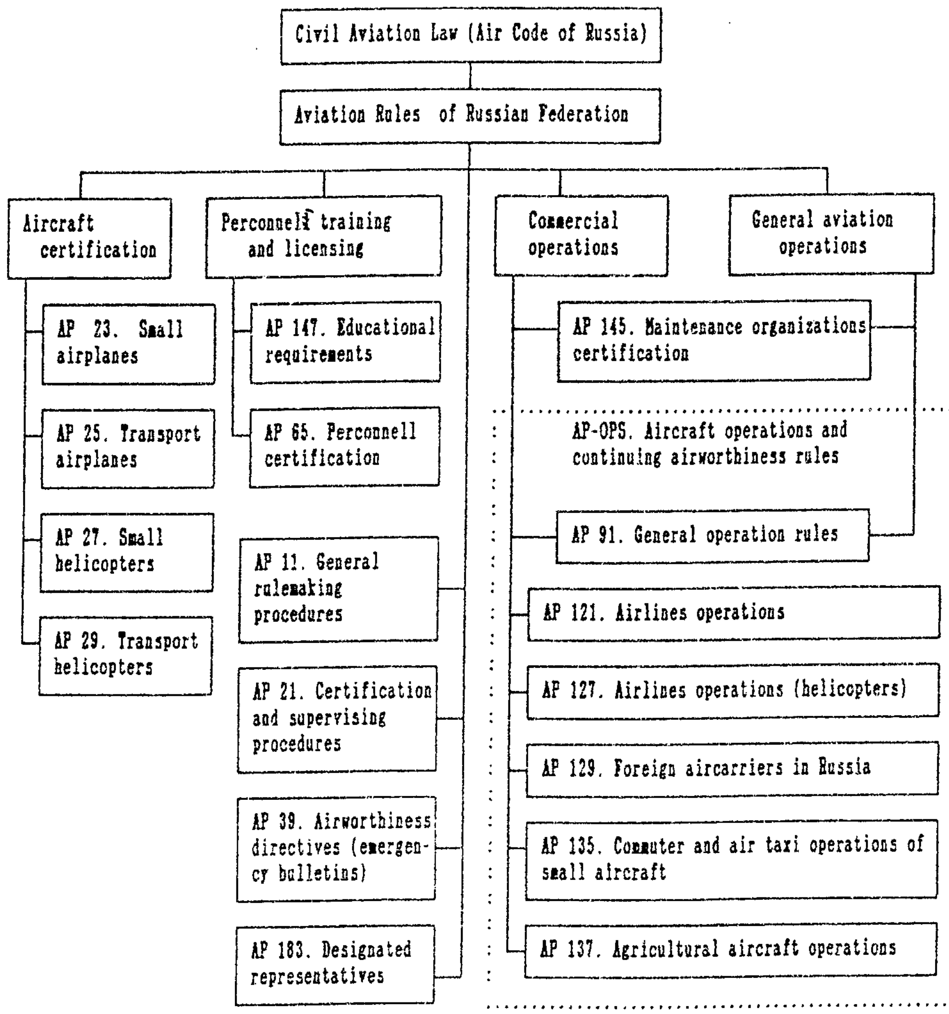


Fig. 6. General scheme of aircraft technical operating capabilities (TOC) ensuring and evaluation system (taking into account the international practice).



NOTE: In order to show proposed general AP structure the titles of the parts are given briefly.

Fig. 7. General structure of the main aviation rules (AP) for civil aircraft development and operation.

EVALUATING FIXED WING AIRCRAFT IN THE AIRCRAFT CARRIER ENVIRONMENT

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Overview

Operating fixed wing aircraft from today's modern aircraft carrier is a demanding task. Evaluation of aircraft/ship compatibility, both during the concept development phase and Engineering and Manufacturing Development (EMD) ground and flight tests presents the evaluation team with unique challenges. The capabilities and characteristics of high performance carrier based tactical aircraft must be quantified for the catapult launch and subsequent flyaway, and the carrier approach and arrested landing tasks. Catapult launching involves determining the minimum safe launch airspeeds while maintaining acceptable flight characteristics in this low altitude, high angle of attack (AOA) regime. Approach and landing requires the slowest possible approach airspeeds while retaining the performance and handling qualities needed for precision glide slope control. Defining the lowest catapult launch and landing airspeeds reduces wind over deck (WOD) requirements, resulting in reduced ship's operating speed and increased operational flexibility. The tight physical confines of the flight and hangar decks, in conjunction with the large number of other aircraft, support equipment, and personnel dictate unique design requirements which must be considered in the earliest design stages of a new airplane. This paper addresses the shore based and shipboard ground and flight tests which are conducted to assess the flying qualities, performance, and structural suitability of an airplane in the aircraft carrier environment.

The Aircraft Carrier Flight Deck Layout

The flight deck layout of today's modern aircraft carrier is shown in figure 1. Two steam powered catapults are located forward (bow catapults) and two catapults are located amidships on the port side (waist catapults). Retractable Jet Blast Deflector (JBD) panels are located aft of each catapult. The centerline of the landing area is angled relative to the ship's centerline, permitting simultaneous catapult launch operations from the bow catapults and arrested landing operations. Four arresting gear cables, connected to arresting engines are located in the landing area. The first is approximately 170 ft (51.8 m) from the stern with approximately 50 ft (15.2 m) between each arresting gear cable. Visual glide slope information is provided to the pilot by a Fresnel Lens Optical Landing System (FLOLS). Aircraft are moved between the flight deck and the hangar deck by four elevators.

Catapult Launch

Evaluation of the catapult launch environment of an airplane covers many disciplines. These areas include:

a) Compatibility with the catapult accessories.

b) Exhaust gas recirculation/reingestion and the thermal/acoustic environment when operating at maximum power in front of the JBD's.

c) Tolerance of the engines to ingestion of steam emitted from the catapult during the power stroke.

d) Structural integrity during the catapult power stroke.

e) Minimum catapult launch airspeeds and characteristics during the rotation and flyaway phases.

f) Shipboard catapult launch operations such as waist catapult operations, lateral/directional trim requirements for asymmetric external stores and crosswinds, etc.

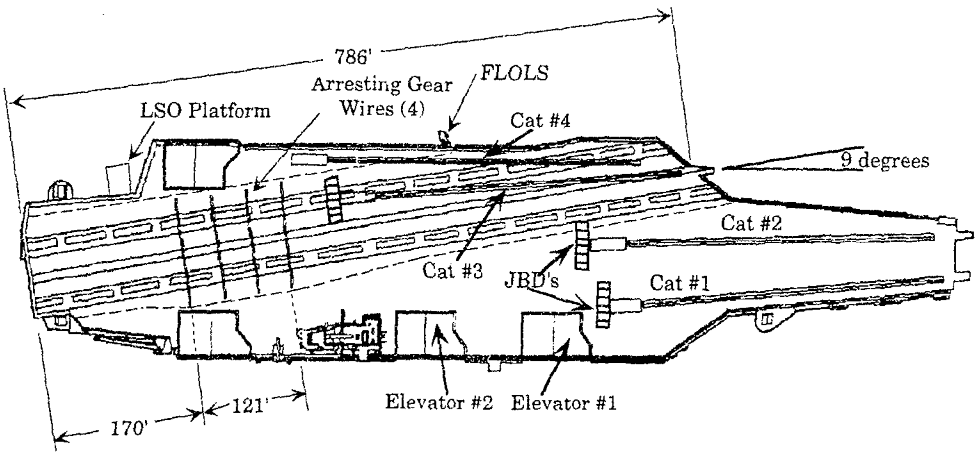

Figure 1
Plan View of Flight Deck
NIMITZ Class Aircraft Carrier

Catapult Accessories

Catapult accessories are the items of hardware necessary to attach the airplane to the catapult. A typical setup is shown in figure 2. Items considered are:

a) Ease of installation of the repeatable release holdback bar onto the nose gear. The operation of this bar does not rely on breakable "dumb bell" elements to release the aircraft, but incorporates an internal strain element and locking rings to release the aircraft at a predetermined load. The holdback bar coupling to the nose gear is unique for each aircraft type.

b) Tracking of the launch bar tee head and holdback bar in the catapult nose gear launch guide rails.

c) Mating of the launch bar tee head with the catapult spreader.

d) Clearance between the airframe and external stores and above deck obstructions such as the catapult shuttle, catapult control station, etc.

e) Holdback bar dynamics following release due to the sudden release of high strain energy.

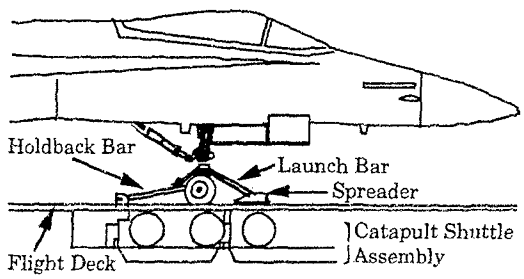

Figure 2
Aircraft/Catapult Hookup

Jet Blast Deflectors

Exhaust gas recirculation and reingestion can occur when an airplane is operating at maximum power levels when positioned in front of the JBD. Reingestion of exhaust gas can cause excessive temperature rise in both the compressor and turbine sections, resulting in damage to the engine. Ingestion of exhaust gas by an airplane positioned behind the JBD can also result in damage to it's engine. Impingement of the exhaust plume on the JBD panels can result in local hot spots which can cause premature warping and cracking. JBD operations also produces a severe acoustic environment. Shore based tests are conducted using a shipboard representative JBD installation. Initial testing consists of placing the aircraft in front of the JBD. The position is varied from the minimum to the maximum engine tailpipe to JBD distances representative of shipboard JBD/catapult combinations Military and afterburner thrust (if equipped) runs are conducted for approximately 30 seconds. Subsequent testing consists of placing the test aircraft aft of the JBD as shown in figure 3. The airplane placed in front of the JBD is the most critical from a thermal and acoustic standpoint. That is currently the F-14A. The acoustic and thermal environment is monitored using microphones and thermocouples mounted on the aircraft and in the vicinity of the JBD. Pole mounted instrumentation provides jet blast velocities and temperatures in the flow field beside and behind the JBD. Generally, the wind over deck during shipboard operations tends to alleviate any recirculation, reingestion, or thermal problems. However, if an airplane has demonstrated a tendency to have excessive exhaust gas ingestion, a shipboard test program may be warranted to define a wind over deck envelope which reduces the ingestion to acceptable levels.

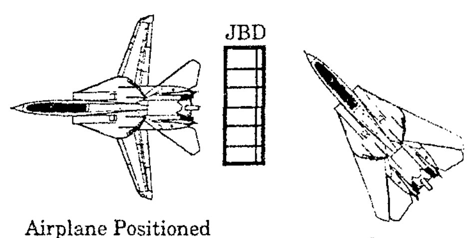

Figure 3
Jet Blast Deflector Testing

Steam Ingestion

Steam catapults typically emit launch steam above the deck during the launch. The design of the engine inlets and the proximity of these inlets to the catapult shuttle frequently cause this above deck steam to be ingested into the engine(s) of the airplane being launched. The result is that the engine is forced to operate at off-design conditions and instabilities can occur. These instabilities can take the form of minor pressure fluctuations within the compressor or the afterburner and could result in blowout, compressor stall, or engine flameout.

The primary method of determining susceptibility to engine stall is to conduct shore based catapult launches from a degraded catapult. The catapult is intentionally degraded by removing plugs in the aft most plate of each piston assembly. This allows steam in the cylinders to travel forward of the aft face of the piston, bypass the catapult cylinder sealing strip as the shuttle assembly lifts the sealing strip during the power stroke, thus allowing the steam to exit above deck around the catapult spreader. This steam leakage produces conditions that are more severe than those encountered in the actual shipboard environment. The airplane is launched a sufficient number of times (about 30 launches) to reasonably ensure that no instabilities are encountered. An appropriate number of additional launches will also be required if the engine is equipped with an afterburner. Testing is confined to those days when the surface winds are less than 10 knots and ± 20 deg relative to the catapult centerline. Telemetered engine performance parameters are monitored to ensure continued satisfactory engine performance.

Structural Requirements

A typical catapult launch structural envelope is shown in figure 4. This figure shows the longitudinal acceleration (N_X)/launch bar load/maximum gross weight boundaries.

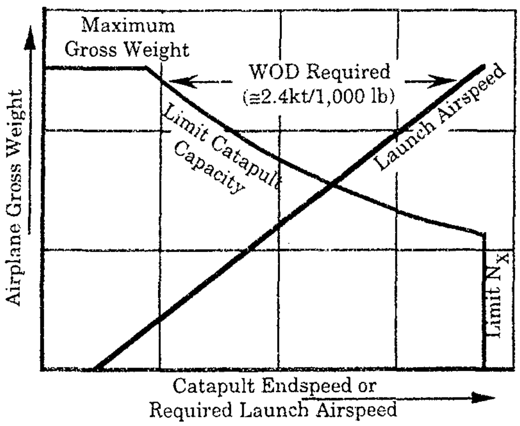

Figure 4
Typical Airplane/Catapult
Structural Envelope

The N_X and limit launch bar load limits are design numbers which are defined by the mission requirements and maximum performance capabilities of the catapult types from which the airplane is to operate. The maximum gross weight is an airplane design factor based on a 10% growth factor of the basic operating weight of the design. Shore based structural testing consists of increasing the catapult end speed until either the limit N_X or launch bar load is reached. Catapult tests involving a new airframe are initially conducted with full internal fuel loads only. As testing proceeds, additional external and internal stores are carried until all weapon stations have demonstrated adequate strength for catapult launch to the limits of the basic airframe. Most launches are conducted with the airplane on center; however, off center launches with the main landing gear up to 24 inches (0.61 m) offset from the centerline position are performed to evaluate structural loads resulting from yaw accelerations and airplane directional characteristics during and following launch. The airplane and suspended stores are extensively instrumented to monitor strains and accelerations for all critical structural areas. An airplane catapult launch structural demonstration program may require up to ten loading configurations to adequately test the structure/functional integrity during catapult launch.

Figure 4 also shows the detrimental effect of unplanned weight growth on catapult launch operations. Unplanned weight growth can be the result of overly optimistic estimates during the initial design phase and corrections to deficiencies discovered during full scale testing. As the weight grows, the maximum catapult endspeed is reduced while the required launch airspeed is increased. The increased wind over deck required to launch the airplane must be provided by increased ship's speed.

Catapult Launch Minimum End Airspeeds

The most extensive test program relating to catapult launch is the determination of the minimum catapult launch airspeeds. From an operational point of view it is desirable that a minimum catapult launch end airspeed be defined. This minimum airspeed is the slowest equivalent airspeed achieved at the end of the catapult power stroke at which the airplane can safely takeoff. Establishing the lowest safe launch airspeed has the following advantages:

a) Decreases the wind over deck required for launch, thus decreasing the ship's speed and increasing the operational flexibility of the aircraft carrier and it's support group.

b) Decreases the loads imposed on the airframe increasing service life.

c) Decreases the amount of energy imparted to the airplane resulting in conservation of fresh water and fossil fuel/core life.

The catapult launch minimum end airspeed is defined by a set of related criteria. Although these criteria generally have interrelated effects, the following addresses each factor separately:

a) Proximity to or warning of stall: The stall airspeed/angle of attack defines an absolute minimum. The required safety margin is dependent upon the characteristics of the airplane under consideration. If stall warning (generally in the form of artificial stick/rudder shaker, airframe buffet, and/or wing drop) occurs at some angle of attack (AOA) below a true aerodynamic stall and the warning does not increase in intensity as the airspeed is decreased to the stall, then the AOA corresponding to stall warning will likely define the minimum end airspeed.

b) Flying qualities/characteristics at high AOA: Frequently an airplane may exhibit adverse flying qualities or characteristics at high AOA, yet at airspeeds well in excess of the stall airspeed. The pilot must then determine the minimum airspeed/maximum AOA at which the airplane characteristics/flying qualities remain acceptable. Examples of limiting characteristics include: buffet, wing rock, wing drop, pitch up tendency, nonlinear stick force gradient, and unacceptable lateral/directional characteristics.

c) Proximity to the airspeed at which thrust available equals thrust required or "lockpoint": For practical purposes, the minimum launch airspeed should be at least 8 kt above the lockpoint. Pilots have indicated that the minimum level of longitudinal acceleration at which he has the sensation of accelerating or climbing is equivalent to 1 kt/sec. This level of acceleration must be available even though this airspeed may be more than 8 kt above the lockpoint. This acceleration capability must be available at the minimum end airspeed. This minimum launch airspeed may become the dominant factor at higher ambient temperatures due to the decreased thrust available with increasing temperature. The maximum catapult launch gross of an airplane may be limited as a function of ambient temperature or the minimum launch airspeed may be increased to put the airplane on a more favorable position on the thrust required curve. Longitudinal acceleration characteristics can also be improved by reducing drag, such as using half flaps instead of full flaps or by the use of afterburner on airplanes so equipped. However, the use of reduced flap settings will increase the minimum launch airspeed thus increasing wind over deck requirements, and the use of afterburner greatly increases fuel usage during takeoff impacting mission radius.

d) Airplane rotation requirements and subsequent sink off the bow: The use of nose gear catapulting has resulted in reduced aircraft pitch attitudes in order to optimize the catapult tow load path from the launch bar, through the nose gear, and into the primary aircraft structure. The pitch attitude at the completion of the catapult power stroke is generally well below that providing the optimum flyway attitude and AOA. A lift deficiency exists during the period of time required to rotate the airplane. This causes the airplane to generate a sink rate which results in sink off the bow until airplane performance/aerodynamics provides sufficient vertical acceleration to establish level flight and subsequent flyaway. The post launch rotation requirement to achieve the flyaway attitude will frequently cause the minimum launch airspeed obtained to be higher than that predicted exclusively from proximity to stall or adverse flight characteristics. For a given airplane end airspeed, sink off the bow will vary with time required to rotate, average lift deficiency during rotation, and excess lift and thrust at the flyaway airplane attitude. Airplane center of gravity (CG) sink off the bow of 20 ft (6.1 m), as measured from the static position on the deck (CG vertical height), is considered the maximum acceptable.

e) Failure of one engine on a twin engine aircraft during launch: Two factors must be considered if an effort is to be made to establish a minimum end airspeed at which an airplane can remain airborne after losing one engine during launch. Foremost of these is the single engine minimum control airspeed (V_{MC}) at which sufficient control authority is available to counter the yawing forces. Secondarily, is whether the single engine rate of climb performance of the airplane is sufficient to permit safe flyaway. The single engine minimum control airspeed will establish an absolute minimum launch airspeed. If only a small increase in minimum end airspeed is required to improve single engine rate of climb performance enabling single engine flyaway, it should be a consideration in establishing the minimum end airspeed. The use of afterburner, if available, should significantly improve single engine performance, but will necessitate an increase in the minimum launch airspeed to provide single engine control.

f) Automatic flight control response: The incorporation of digital, fly-by-wire flight control systems into more recent aircraft models has eliminated the need for pilot programmed flight control inputs to attain a predetermined rotation and flyaway response. Current systems are implemented such as to achieve a desired flyaway trim AOA. However, flight control response due to pitch rate feedback during the highly dynamic conditions during the first several seconds following catapult shuttle release may result in flight control surfaces reaching their physical limits. If any of the primary flight control surfaces reach full deflection during the rotation or initial flyaway phases, the minimum end airspeed is then limited by this criterion.

Test Considerations

A considerable amount of time and effort is expended during shore based build-up to generate prerequisite data prior to tests aboard ship. Careful consideration is given to all the factors governing the minimum end airspeeds so that the results are applicable to the entire range of Fleet operating conditions. These factors include the high lift configuration (half or full flaps), external store loadings, CG positions, longitudinal trim requirements, and thrust (Military or afterburner).

Since the intent of determining the minimum airspeed is to define the lowest launch airspeed, the highest lift configuration is tested. With airplanes having more than one flap setting, the maximum flap deflection is suggested. However, this decision has to be tempered with the possibility of reduced nose up pitch authority which could result in increased time to rotate to the flyaway attitude, thus increasing sink off the bow. Additionally, there is an increased chance of reaching control surface limit deflections. The higher flap setting also results in more drag, thus decreasing longitudinal acceleration. External stores are selected to cover the range of anticipated gross weight, CG, and drag conditions expected during operational use. Forward and aft CG positions are tested to evaluate rotation characteristics and to define longitudinal trim requirements to be set prior to launch.

Shore based build-up flight tests are conducted in each of the high lift, external store, and CG position conditions. Classical flight test techniques are used to define the longitudinal/lateral/directional characteristics at high AOA up to stall, static/dynamic single engine control airspeeds, and thrust available and required. Shore based catapult launches are conducted at the predicted minimum end airspeed to investigate trim requirements, flyaway characteristics, and pilot technique. Shore based catapult launches are preliminary in nature because the airplane remains in ground effect and, of course, there is sink off the bow. All of these shore based tests enable prediction of the catapult launch minimum end airspeeds. The final judgment comes aboard ship.

Testing at Sea

The shipboard tests are conducted in a tightly controlled environment. Tests are conducted with steady winds from dead ahead and minimal deck motion. The catapult is maintained at a constant thermal state to ensure repeatability of catapult end speeds during subsequent launches. A calibrated boom anemometer is installed on the bow to provide accurate wind speed and direction. Noncritical external store loadings and CG's are tested initially. Initial launches are conducted well in excess of the predicted minimum airspeed (approximately 25 knots). Upon recovery following the launch the airplane is refueled and external stores expended prior to recovery are reloaded to re-establish the desired gross weight and CG. The catapult end airspeed is reduced in suitable decrements; initially 5 knots and then 3 knots as the predicted minimum end airspeed is approached. The initial reductions in catapult end airspeed are achieved by reducing the catapult end speed and as the predicted minimum end airspeed is approached, the catapult end speed is maintained constant and the wind over deck is lowered by reducing ship's speed Airplane performance parameters; such as sink off the bow, rotation characteristics, flight control response, longitudinal acceleration, etc. are monitored and analyzed by the engineering test team via telemetered instrumentation. Catapult launch end airspeed is thereby reduced until one of the previously mentioned criterion are reached. This sequence of catapult launch tests are repeated for each critical gross weight, CG position, and external/internal store loading until the operational envelope has been defined.

In general, no minimum end airspeed criterion is the determining factor throughout the operational gross weight range of an airplane. An airplane may be V_{MC} limited at lighter gross weights, sink off the bow limited at medium gross weights, and longitudinal acceleration limited at high gross weights and ambient temperatures. Figure 5 represents these three different criteria.

It is important to note that the minimum catapult launch end airspeeds are the lowest airspeeds that an airplane can be safely launched. However, these airspeeds are determined under optimum conditions. These conditions include day Visual Meteorological Conditions (VMC), a nonpitching deck, steady winds monitored by a calibrated anemometer, skilled aircrew trained in the optimum technique, gross weight and CG accurately known, catapult performance closely monitored, and end speed corrections made for ambient temperature and barometric pressure.

In view of all these considerations, operational catapult launch operations are conducted at a recommended airspeed 15 knots above the minimum launch airspeed.

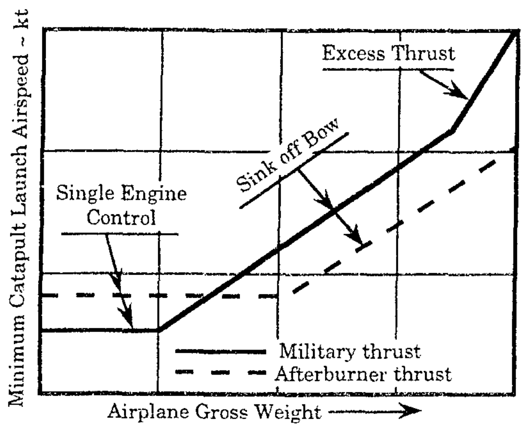

Figure 5
Factors Defining Catapult Launch
Minimum End Airspeeds

Although the shipboard test program to define catapult launch minimum end airspeeds involves the bulk of operations, other catapult launch tests are required. These include:

a) Waist catapult operations to assess the effect of the additional flat deck run forward of these catapults on the rotation characteristics and subsequent sink off the bow.

b) Lateral/directional trim requirements for asymmetric store loadings.

c) Crosswind launch operations, from both the bow and waist catapults, to determine lateral/directional trim requirements for crosswind components up to 15 knots.

d) Sensitivity of rotation characteristics and associated sink of the bow to improperly set longitudinal trim.

e) Light gross weight/low catapult end speed launches to evaluate the potential for degraded nose gear stored energy imparted pitch rates due to low catapult launch bar loads at the end of the power stroke.

Carrier Approach and Landing

The aircraft carrier approach and landing task is the most demanding task in aviation. The requirement is to maintain precise glide slope control to land in an area ± 20 ft (± 6.1 m) of the angled deck centerline and where the distance from the first arresting gear cable to the last cable is only 120 ft (36.6 m). Control of both AOA and airspeed is demanded to remain with the structural limits of both the airplane and the arresting gear engines. This must be accomplished during both day and night operations and in all types of weather.

A typical VMC landing pattern for an aircraft carrier is presented in figure 6. Terminal glide slope information is provided to the pilot by a Fresnel lens optical landing system (FLOLS). For most recoveries, the glide slope is set for 3.5 deg. The approach is monitored onboard the ship by a Landing Signal Officer (LSO). The location of LSO and FLOLS is shown in figure 1.

Shore Based Tests

Extensive shore based approach and landing tests are conducted to determine the suitability of an airplane for carrier approach and recovery prior to initial sea trials. These tests include:

a) Structural integrity during landing and arrestment.

b) Optimum approach AOA and associated airspeeds.

c) Bolter and waveoff performance and characteristics.

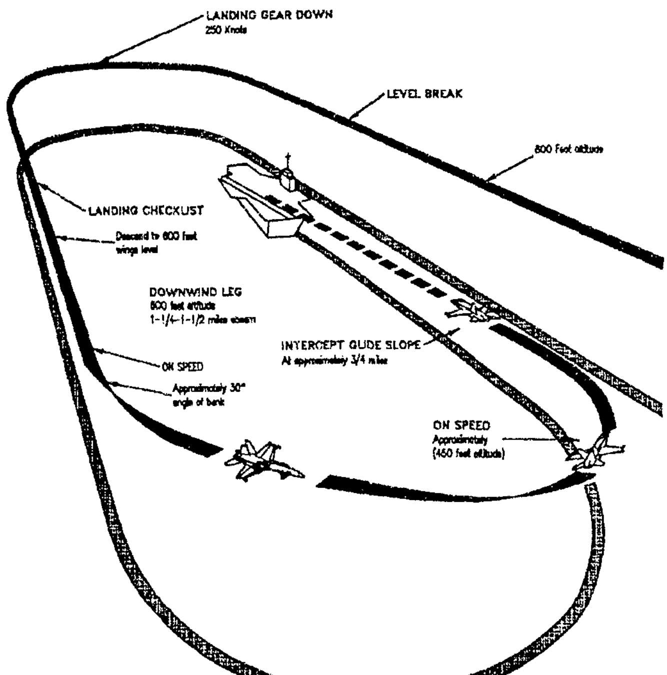

Figure 6
Typical VMC Landing Pattern

Structural Tests

Landing an airplane aboard an aircraft carrier imposes severe loads on the landing gear and airframe. A flared landing is not performed. Immediately after landing, and sometimes before, the decelerating forces of the arresting engine are encountered. Last second glide slope and lineup corrections when encountering the turbulence induced by the ship's structure in combination with ship's motion can cause high airplane touchdown speed or rolled/yawed attitudes. Shore based arrested landing tests are conducted to evaluate structural integrity when landing in the many types of conditions possible aboard the carrier. These conditions are:

a) Maximum arresting gear engaging speed: This condition produces the maximum arresting hook loads and longitudinal decelerations and are conducted at the limit design condition of the airplane.

b) Rolled and yawed attitude at touchdown: This type of landing represents a last second lineup correction. The target attitude for both roll and yaw at touchdown is 5 deg. Landings are conducted with the roll and yaw in the same direction and also in the opposite direction.

c) Free flight arrestment: Occasionally an arresting hook will engage the arresting gear cable prior to main landing gear touchdown. This could happen with an inclose pitchup attitude change or during a waveoff. This type of arrestment is called a free flight. High nose gear landing loads are obtained upon touchdown. Free flight arrestments are intentionally conducted during the shore based test program.

d) Off center: All landings don't always occur in the center of the targeted landing area. Off center arrestments, up to 20 ft (6.1 m) left and right of the centerline are conducted to investigate the high side loads imposed on the arresting hook and airframe structure during this type of landing. The wing rock dynamics induced during this type of arrestment are monitored to determine any potential for contact of the wingtip or wing mounted external stores with the runway or arresting gear cables.

e) High sinking speed: To meet the design requirements for shipboard landings, U. S. Navy airplanes are designed for touchdown sinking speeds up to 26 fps (7.9 m/s). High sinking speed tests are the most critical of all the arrested landing structural tests. In the interest of safety, actual flight tests are conducted to 80% of the design limit. During testing, the targeted sinking speed is increased by slowly increasing the angle of the optical glide slope until the targeted sinking speeds achieved. In addition, this sinking speed is required to be tested at three different airplane pitch attitudes; 1) the normal landing attitude, 2) nose down (three point landing or nose gear first), and 3) a taildown attitude 3 degrees higher than the normal landing attitude.

The above five landing conditions are repeated in each of the critical external store combinations that the airplane will experience operationally.

Earlier, figure 4 showed the detrimental effects of weight growth on catapult operations. A more severe impact due to weight growth is realized during recovery operations. As weight increases, the limit arresting gear engaging speed decreases and the approach airspeed increases. The resulting increase in WOD is approximately 3.3 kts/1,000 lbs (453.6 kg).

Approach AOA and Airspeeds

Many factors must be considered relating to the determination of the recommended approach AOA and the associated airspeeds for the range of recovery gross weights. It is desired that the slowest possible approach AOA and airspeed be defined in order to minimize recovery WOD requirements. However, the need to establish the slowest AOA must be weighed against the requirement to ensure adequate flying qualities and performance to safely perform the carrier landing task. To this end, a number of criteria, mainly quantitative, have been developed to enable evaluation of the approach AOA and airspeeds. These criteria are part of the performance guarantees specified in the requirements for new aircraft. Attaining these criteria "***should***" ensure satisfactory carrier approach flying qualities and performance characteristics. For an airplane in the landing configuration on a 4 deg glide slope on a 89.8°F (32.1°C) day and at the maximum carrier landing gross weight, the minimum usable approach airspeed (V_{PA}) will be the ***highest*** of the airspeeds required to meet the criteria detailed in the following paragraphs.

a) Acceleration Response to Large Throttle Inputs: For a large throttle input, such as a waveoff, the slowest airspeed will be that airspeed at which it is possible to achieve a level flight longitudinal acceleration of 5 fps^2 (1.5 m/s^2) within 2.5 seconds after throttle movement. If any flight control effectors or speed brakes are automatically scheduled with throttle movement, then these surfaces may be moved. It is important to note that this requirement does not imply that the airplane must be in level flight with an acceleration of 5 fps^2 (1.5 m/s^2), rather that, during the approach, the engine(s) be operating in a region such that the acceleration characteristics would enable the engine to accelerate from the thrust required on glide slope to that thrust level equaling 5 fps^2 (1.5 m/s^2) acceleration at the same airspeed in level flight.

b) Acceleration Response to Small Throttle Inputs: The second approach airspeed criterion relating to acceleration capability is rapid engine response to small throttle movement. At the approach airspeed, step throttle inputs corresponding to a 3.86 fps^2 (1.18 m/s^2) longitudinal acceleration command will result in achieving 90% of the commanded acceleration within 1.2 seconds. This requirement applies both to acceleration and deceleration. This requirement applies throughout the weight range and anticipated drag levels of the airplane. Figure 7 shows this requirement.

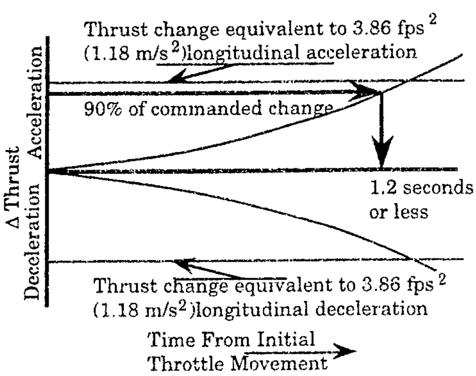

Figure 7
Acceleration Response
to Small Throttle Inputs

c) Over The Nose Field of View: The slowest acceptable approach AOA must provide adequate over the nose field of view. With the airplane at an altitude of 600 ft (182.9 m) above the water in level flight and with the pilot's eye at the design eye position, the waterline at the stern of the ship must be visible when intersecting a 4 degree optical glide slope. The source of the optical glide slope is 500 ft (152.4 m) forward of the ramp of the ship and 65 ft (19.3 m) above the water. This requirement is shown in figure 8.

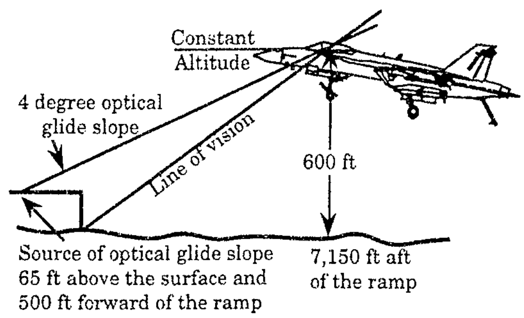

Figure 8
Over the Nose Field of View

This requirement is to ensure that the pilot is provided adequate visual lineup cues during the early phase of the approach. These principle cues are the angled deck centerline and the drop line on the stern of the ship. These two lines provide immediate indication of an off center lineup condition. An example is shown in figure 9. As the approach continues, additional lateral deviation information is provided by the landing area ladder lines as the drop line goes under the nose of the aircraft.

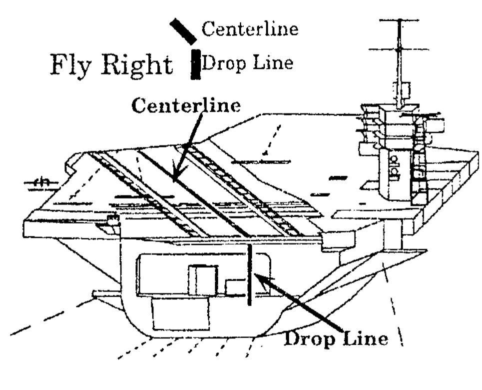

Figure 9
Visual Line-Up Cues

d) Margin Over Stall: The slowest airspeed equating to 1.1 $V_{S_{PA}}$, where $V_{S_{PA}}$ is the power-on stall airspeed using the power required for level flight at 1.15 V_{S_L}, which is the power-off stall airspeed. The determination of this airspeed is to first calculate V_{S_L}, calculate the power required to maintain unaccelerated level flight at 1.15 V_{S_L}, determine the power-on stall airspeed at this power level, then calculate 1.1 $V_{S_{PA}}$.

e) Flying Qualities: The slowest approach airspeed shall provide Level I stability and flying qualities.

f) Glide Slope Transfer Maneuver: This requirement is often referred to as the 50 ft (15.2 m) pop-up maneuver. The airplane is to perform a glide slope maneuver so as to transfer from one glide slope to another glide slope which is 50 ft (15.2 m) above and parallel to the first glide slope. The 50 ft (15.2 m) transfer is referenced to the CG of the airplane. The maneuver must be completed within 5.0 seconds. Longitudinal control can be inputted as necessary with the constraint that the maximum incremental load factor cannot be greater than 50% of that available at the start of the maneuver. The throttle setting cannot be changed during the maneuver. This maneuver is often misunderstood to mean that the altitude of the airplane is increased. In fact, the altitude at the end of the maneuver can be somewhat below that when initiated. For example, if the sink speed of the airplane is 15 fps (4.6 m/s) at the start, the airplane will intercept the new glide slope 25 ft (7.6 m) lower in altitude than when the glide slope transfer was started [15 fps (4.6 m/s)x 5 sec - 50 ft (15.2 m) = 25 ft (7.6 m)]. Once the new glide slope has been intercepted, longitudinal control and throttle inputs can be made to establish a new glide slope parallel to and at least 50 ft (15.2 m) above the initial glide slope. Figure 10 presents this maneuver.

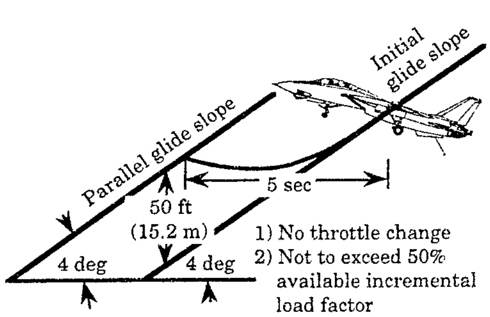

Figure 10
Glide Slope Transfer Maneuver

Additional Considerations:

Single Engine Control Airspeed: For a multi-engine airplane the slowest approach airspeed will not be less than the single engine control airspeed (V_{MC}). This will ensure adequate control in the event of a total engine failure during a waveoff when performed at the approach airspeed.

Touchdown Attitude: Touchdown attitude considerations have on occasion dictated the selection of an approach airspeed/AOA. The pitch attitude must be such that a tail down, free flight, or nose down arrested landing with resultant airframe damage be only remotely possible.

Glide Slope Tracking: The combination of airframe/engine performance is of prime importance in evaluating the handling characteristics of an airplane on the glide slope. The speed/power (or flight path) stability characteristics of an airplane have a great deal of influence on the ability of the pilot to make corrections in airspeed and rate of descent.

The following glide path correction capabilities are considered over the approach airspeed/AOA range:

a. The ability to make glide path corrections by changing the rate of descent at a constant thrust setting.

b. The ability to make glide path corrections by varying the thrust while maintaining a constant airplane AOA.

In making glide path corrections, the pilot instinctively attempts to do so initially with longitudinal control. Effective control of airplane pitch attitude necessitates that the longitudinal control power, damping, and mechanical characteristics be such as to permit small, precise pitch attitude corrections. It is extremely desirable that the airplane have maneuvering capability at a constant thrust setting for small changes in AOA on the order of one or two degrees. For making large corrections to the glide path which are sometimes necessary early in the approach, it is necessary to determine the change in thrust required for changes in AOA. An airplane that possesses this characteristic is easier to fly on the glide slope by correcting to glide slope with longitudinal control, returning to the proper approach angle of attack, and then adjusting thrust to correct for the original erroneous setting. Using this method, rapid glide path corrections are possible and thrust corrections in only one direction are required for each evolution.

If an airplane does not respond satisfactorily to longitudinal control, an alternate technique is evaluated. The airplane is maintained at the desired AOA and thrust corrections are used exclusively to make glide path corrections. With this technique, the airplane response as a function of the excess thrust available for maneuvering ($\Delta T/W$), the increase in thrust for small throttle movements, the engine acceleration characteristics, and the contribution of thrust to lift are all evaluated. Because of the lag in engine and airplane response to throttle movement and because of the tendency to "overcorrect" in order to establish vertical acceleration, it is difficult to determine the proper thrust setting required to hold the glide path. As a result, the pilot gets "behind" the airplane and the airplane follows a mild oscillatory path in the vertical plane of the glide slope. Therefore a procedure in which thrust and longitudinal control are initiated simultaneously is necessary for rapid corrections even though it requires precise coordination and increases pilot workload. The use of speed brakes may lessen the aircraft perturbations since increased power setting may provide better engine acceleration and the addition of any parasite drag device such as speed brakes contribute to speed stability by reducing the airspeed for minimum drag.

A combination of the numerous approach airspeed/AOA criteria dictates that the approach be made on the unstable portion (back side) of the thrust required curve. If an approach is made in this area, the use of the throttles is mandatory for making corrections in airspeed and rate of descent and thereby increases the difficulty of executing a precision approach. Further, if the approach must be made on the unstable portion, it is desirable that the thrust changes required are not large for small excursions from the approach airspeed. In terms of flight path stability, the change in flight path angle with airspeed should not be greater than 0.06 deg/kt. A rapidly increasing slope of the curve means that the airplane may decelerate rapidly and require the pilot to add much more thrust to stop a deceleration when compared to the thrust reduction necessary to stop an acceleration. It is also desirable that the approach be made where the curve has a gradient and not on the flat or neutral flight path stability portion where a range of airspeeds are possible for approximately the same thrust setting.

Lineup Control: Effective control of airplane heading is mandatory for carrier deck lineup control. Lateral control power, damping, and mechanical characteristics (trimmability, stick breakout forces, stick gradients, stick deadbands) should be such that the pilot can effect small, precise line up corrections during the approach. The use of lateral control should not cause distracting pitching or yawing moments.

The previous discussions have highlighted in general terms the numerous items which have a bearing on the selection the approach airspeed/AOA and an optimum pilot technique. Frequently, several flying qualities and performance characteristics become marginal at the same airspeed/AOA and one may mask another. It is important to recognize all of the factors involved since improvements of one may render another more acceptable or unacceptable.

Qualitative Evaluation Tasks: In addition to the quantitative and qualitative evaluation techniques which are used in defining the approach AOA and associated airspeeds, it is possible to evaluate the approach and landing by defining the tasks the pilot must accomplish for each phase of the landing. Table 1 specifies the distinct phases during landing and lists suggested tolerance bands for the required performance. These levels of performance should be attainable with an HQR - 3 or better.

Table 1
Approach and Landing
Qualitative Evaluation Tasks

Phase	Task	Tolerance Band
Downwind	Airspeed Control Heading Control Trimmability	± 2 kt ± 2 deg ---
Base Leg	AOA Control Roll Attitude Control Heading Capture	± 1/2 deg ± 1 deg ± 2 deg
Final	AOA Control Lineup Control Glide Slope Control	± 1/2 deg ± 1 deg ± 1/2 "ball" (see note)
Touchdown	Runway Centerline Longitudinal Dispersion	± 5 ft (± 1.5 m) ± 20 ft (± 6.1 m)
Waveoff or Bolter	Attitude Capture AOA Control	± 1 deg ± 1 deg

Note: A "ball" is equivalent to one cell on the Fresnel Lens Optical Landing System. One cell equals 0.34 degrees of arc.

Waveoff Performance and Characteristics. A waveoff is a frequent occurrence in the shipboard environment and one which may be required due to the landing area going "foul" or not being ready to recover aircraft, unacceptable pilot technique, or conditions outside safe recovery parameters, such as excessive deck motion. A late waveoff is extremely demanding on airplane performance because of airplane sink rate and proximity of the ship. Flight tests are conducted to quantify airplane performance and determine the optimum pilot technique. This information is generated for both the normal recovery configuration(s) and all potential emergency modes, either airframe or engine related, for which shipboard recovery is possible.

Waveoffs are initially conducted at a safe altitude to assess airframe and engine response. The airplane is stabilized on-speed on a -3.0 deg flight path angle. Pitch tendency with power is noted. The landing configuration(s) and emergency conditions should be investigated. Simulated single engine characteristics and airspeeds, both static and dynamic, must be investigated prior to approaches at the field.

Two basic types of approaches terminating in a waveoff are investigated. They are:

a) Stabilized, on glide slope condition: This simulates a stabilized approach condition where a waveoff is required in response to an unsafe condition such as the deck going "foul". The airplane should be in a relatively stabilized condition at the approach AOA with the throttles at the approximate approach setting. To evaluate the variation of waveoff altitude lost and time required to achieve a positive rate of climb with sinking speed, the FLOLS basic glide slope angle is varied. In addition to on-speed conditions, AOA's as slow as 2 degrees higher than the approach AOA should be tested.

b) A high comedown condition: This condition represents a large throttle input by the pilot attempting to correct from a high (above glide slope) condition. The use of this "gross" correction technique will usually result in an immediate waveoff by the LSO. The test procedure should be to stabilize on glide slope, but holding a "one ball high" condition. At the desired time, the pilot retards the throttles to IDLE. From 1.0 to 2.0 seconds later, the waveoff signal should be given. This test technique has limited applicability within 1,000 ft of touchdown (a point approximately 500 ft (152.4 m) past the ramp) as this type of throttle "play" would result in an immediate waveoff being commanded by the LSO; however, this technique will identify unacceptable waveoff performance and excessive altitude loss due to adverse engine response characteristics.

Two pilot techniques for MIL thrust waveoffs are investigated. The first technique involves maintaining the approach AOA throughout the waveoff maneuver. The second technique involves rotation to higher values of AOA. Level I flying qualities must be retained at all times during the waveoff. Airplane pitch response to MIL thrust application and/or automatic configuration changes, such as speed brakes, may result in a slight uncommanded AOA rotation during the waveoff. This can be a favorable response in the noseup direction; however, is unacceptable in the nose down direction. Although rotation may minimize altitude loss, a point is reached near the ramp where rotation is undesirable due to reduction in hook/ramp clearance and the probability of a free flight engagement outside of the airplane design envelope. This undesirable characteristic is most noticeable for aircraft with large linear distance from the pilot's eye to the hook, such as the F-14A, where the vertical hook-to-eye distance increases approximately 1 ft (0.3 m) for each degree increase in pitch attitude.

The use of afterburner, if available, should also be investigated. Frequently, the time required to obtain MAX A/B thrust obviates its use to lessen the altitude loss during the waveoff maneuver. However, Max A/B thrust does provide an increase in acceleration once a positive rate of climb has been established and can avert a ramp strike for an airplane which has developed a high sinking speed prior to reaching the critical distance from the ramp. Average altitude loss determination for the various loadings and approach conditions should be based on at least twelve data samples because of differences in pilot techniques.

Fleet experience has shown that waveoff performance will be satisfactory if the following criteria is met from waveoff initiation during an approach on glide slope with the proper AOA and 0.7 sec pilot reaction time:

a) An hook point altitude loss not greater than 30 ft (9.1 m).

b) A time to zero sink speed not greater than 3 sec with a corresponding level flight longitudinal acceleration of 3 kt/sec on a 90°F (32.2°C) ambient temperature day.

c) A controllable aircraft pitch attitude change not greater than 5 deg airplane nose up or an AOA increase not more than 3 deg.

Bolter Performance and Characteristics. A bolter is an unintentional arresting hook down touch and go landing on the ship. A bolter can occur due to:

a) Improper in-close thrust or pitch attitude inputs or an excessively high glide slope position which result in the arresting hook point passing over the top of all the arresting gear cables. This is the more critical condition in that the minimum flight deck is remaining to execute the bolter maneuver.

b) The arresting gear hook point landing in the desired position, but the hook point failing to engage a cross deck pendant (CDP) due to: 1) hook point dynamics resulting in excessive hook bounce or lateral swing of the arresting hook shank preventing the hook point from engaging a CDP, or 2) improper tension on the CDP from the arresting engine allowing the CDP to be closer to the deck than desired limiting, the ability of the hook point to engage the CDP. In either case this is commonly referred to as a "hook skip bolter".

The distance from the last arresting gear cable to the angled deck round down varies from a minimum of 427 ft (130.1 m) on KITTY HAWK class ships to a maximum of 495 ft (150.9 m) on NIMITZ class ships.

Shore based touch and go landings are conducted to determine bolter performance, characteristics, and desired pilot technique. Landing sinking speeds at touchdown should be at least the mean carrier landing sinking speed to ensure that the airplane's pitch dynamics during the bolter, due to compression/extension dynamics of the main and nose landing gear, are representative of a shipboard landing. Flared landings will not produce realistic test conditions! All normal and emergency configurations should be tested. The forward and aft CG positions can be critical because of the potential effect on nosewheel liftoff airspeeds at forward CG's and adverse longitudinal characteristics at aft CG's.

The preferred method of obtaining bolter performance is to use LASER tracking data. The data is used for ground speed and ground roll only. Desired airborne instrumentation, in addition to the standard suite, includes nose and main landing gear weight on wheels (WOW) discretes which can be used to "time tag" their respective touchdown and liftoff times. The ground roll distances from main landing gear touchdown until nose landing touchdown, nose landing gear liftoff, and main landing gear liftoff are calculated from the LASER data. The calculated bolter distances are corrected for test day surface winds and then recomputed for anticipated recovery WOD in the shipboard environment.

The recommended pilot technique during these tests should be application of MIL power at touchdown and longitudinal control input as necessary to achieve the desired flyaway attitude. However, the use of full aft control can produce undesirable over-rotation tendencies. Other techniques should be considered if the characteristics of the airplane warrant.

It is desired that the airplane achieve both nose and main gear liftoff prior to rolling off the end of the angled deck round-down. However, if this condition is not achievable, it is still acceptable if there is no aircraft sink following rolling off the angled deck. Any CG sink is unacceptable. This requirement for no CG sink is based on a "normal" bolter. Situations will occur that will result in some CG sink. Delayed pilot response to the proper bolter technique of throttle and longitudinal control or initial landing gear touchdown well beyond the last CDP are examples.

The airplane pitch characteristics during the shore based bolter tests should be monitored. Landing gear dynamics can cause pitch oscillations (rocking) during the bolter. In an extreme situation, the airplane could be in a nose down pitch cycle when the nose gear rolls off the angled deck, resulting in unacceptable airplane characteristics and excessive sink following rolling off the angled deck.

Testing at Sea

Final determination as to the suitability of an approach airspeed/AOA, pilot technique, and bolter and waveoff performance and characteristics can only be obtained from actual tests aboard the carrier because of airflow disturbances over the landing area and aft of the carrier. Turbulence in the form of sudden updrafts and downdrafts which occur aft of a carrier cannot be duplicated ashore. The range of WOD's to be used should be from the minimum recovery headwind up to 40 knots, if achievable. Crosswinds components, both port and starboard, up to ship's limit (7 knots) should be investigated to evaluate the ship's island structure.

Initial approaches are terminated in waveoffs at approximately 1/2 nautical mile (1.9 km); the waveoff point is moved closer to the ship as test results merit. The first landings are "hook-up" touch and goes, finally with hook down to achieve the first arrested landing.

Intentional landings beyond the CDP's should be conducted to minimize deck remaining and time available to initiate bolter inputs, and also to evaluate rocking characteristics due to landing gear dynamics.

Wrap-up

The material discussed in this paper covers only a small number of the many disciplines which make an airplane "carrier suitable". Many other areas including 1) flight deck and hangar deck towing, spotting, tie-down, and maintenance, 2) elevator compatibility, 3) electromagnetic compatibility, and 4) adequate clearances between critical airframe components and externally carried stores from ship's structures such as hatches, catapult launching consoles, and deck edge scuppers, are all of equal importance when assessing the suitability of a new airframe.

BASE PRINCIPLES OF CREATING AN EXPERIMENTAL COMPLEX FOR FLIGHT TESTS OF THE AIRCRAFT WITH SCRAMJET ENGINES

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Creating the new generation of aerospace vehicles and hypersonic aircraft needs considerable progress in the sphere of powerplant development. These powerplants are intricate complexes containing turbojet, ramjet, and scramjet engines which are turned on successively.

Among the main problems to be solved in creating scramjet engines are the ensuring of required flow parameters at an air-intake, supersonic combustion of hydrogen fuel, thermal protection of scramjet structure elements. Accurate prediction of force and moment aircraft characteristics with the operating engine is also a complicated matter.

Up-to-date wind tunnels do not support sufficiently full simulation of real physicochemical processes within external and internal ducts of a scramjet engine since it is very difficult to ensure combinations of Mach numbers, total enthalpy, and total pressure, which are real in a flight, during the minimal required period of scramjet testing in ground-based installations for $M > 6$. Geometrical dimensions of a scramjet engine under tests are important to get reliable information. These reasons make flight experiment be of great significance. Its advantages are the possibility of natural combinations of a series of important similarity criteria, experiment duration, etc.

When the programs of creating the transport systems NASP, ZENGER, HOTOL were planned, the stage of some anticipating flight research was foreseen. Flight research of main priority problems may be carried out with experimental aircraft of the following three classes:

— full-scale manned aircraft, such as X-30, Hytex, etc.;
— semi-scale manned or unmanned aircraft;
— small-scale unmanned aircraft.

The analysis of the authors of the paper [1] shows that a demonstration aircraft of the X-30 type make it possible to solve the most important problems.

A semi-scale aircraft ensures the performance of about 60% of requirements for getting full scientific and technical information. Profound study of the ways of problem solution in creating a full-scale manned aircraft showed the necessity of using considerable financial resources.

At the same time, the cost of a semi-scale research aircraft comes to about 50% of full-scale aircraft development. However, when the start in theoretical research and ground-based investigations is insufficient, some problems make the aircraft development irrational in both cases since technical risk is too high.

Reduced-scale aircraft (flying models — as they are called in Russia) are effective tools to investigate some problems of development of hypersonic aircraft with scramjet engines.

A powerplant is the main key problem in creating aerospace vehicles and hypersonic aircraft. A scramjet engine is its least studied part. The investigations to create this engine may be carried out on the base of flying models. This task at the minimum may be formed at the first stage of flight research. Among the most significant scientific and technical problems which might be studied at this stage are the study of the processes of supersonic combustion of hydrogen fuel, powerplant/airframe integration within a wide range of M numbers (including regimes which are inaccessible for up-to-date ground-based installations). Approving ground-based experimental methods of investigations and techniques of modeling is an important purpose of flight research.

The main requirement for the research is carrying out flight regimes within the altitude-airspeed envelope that is described with the dependence in fig. 1. The range of dynamic pressure from 5000 to 10000 kgf/m^2 corresponds to this envelope. To study the combustion process, a way to these regimes may be realized in both climbing and descending. In the first case, a flight trajectory reproduces the original flight. However it is unattractive in power sense and because of structure heating (especially for maximum M numbers) since the acceleration takes place in dense atmosphere. It seems to us that to carry out investigations for the range $M > 7$, the second case is more rational, viz.: an aircraft under study is launched beyond the atmosphere in optimal way and, then, after separation from the rocket booster, enters the atmosphere with the given initial conditions. This variant was used when investigations of the BOR program (for the BOR-5 product) were under way. It makes it possible to employ available rocket boosters without essential modifications.

To keep experiment supplied better with measurement facilities and simplify test organization, it is necessary to reduce the length of a flight-test route and ensure the constancy of a landing area. To reduce the length of a flight way in the atmosphere, the entry into it has to be fulfilled with the greatest negative trajectory angle which is acceptable from the point of view of maximal temperatures on the aircraft structure elements. At the same time, angles of attack have to be great and positive to prevent aircraft dipping into dense atmosphere layers.

The reserve of hydrogen fuel on the aircraft board guarantees the engine operation during 20—30 s. This is enough to fulfil investigations but limited fuel reserve does not make it possible to keep them within all the necessary range of Mach numbers (from $M=14$ to $M=8$) during one launch. That is why investigations with practically constant Mach number are carried out in every launch.

Theoretical study and ground-based experiments show that to ensure sufficiently reliable results, the length of an aircraft nose section, on which the necessary flow parameters at the cross-section of engine nozzle inlet are formed, has to be greater than 4 m. The minimal length of engine interior (including the channels that form flow parameters directly before a combustion chamber and this chamber itself) has to be greater than 2 m.

So, minimal necessary aircraft length is 6 m. The problems of getting thrust and aircraft/exhaust jet interaction cannot be studied in this case.

However, to our opinion, studying these subjects on such an aircraft of 6 m length is of some interest owing to the possibility of using created starts in the forms of existing launching facilities and vehicle structures which are elaborated enough. The example of such a vehicle configuration is presented in fig. 2.

The aircraft without a tail unit has a low-set wing with elevons and great V, two small fins disposed on the lower surface. The vehicle surface is covered with thermal-protection materials.

A high-set scramjet air-intake is the feature of this configuration. It is convenient for thermal protection of the air-intake in non-operational regimes when a flight is carried out with sufficiently great positive angles of attack and the air-intake is in aerodynamic shadow. A flight within the operational trajectory leg takes place with negative angle of attack that is necessary for proper engine operation. A high-set air-intake and fins on the lower surface make it possible to raise the probability of survivability of the scramjet engine when parachute landing since parachute suspension ensures fin landing that makes the shock softer.

The "Cosmos" rocket booster with an additionally developed nose cone may fulfil injections into the flight regimes corresponding to the conditions shown in fig. 1.

To study total effectiveness of the powerplant and engine/airframe interaction with an external expansion nozzle, it is necessary to enlarge the aircraft dimensions at the expense of an exhaust nozzle. According to preliminary estimations, its minimal length is 3 m.

So the total length of the research aircraft is 9 m in this case. The example of such configuration is shown in fig. 3. The launch of this aircraft is assumed to be fulfilled with the "Rokot" rocket.

Studying the complex powerplant that contains turbojet, ramjet, and scramjet engines may become the next stage of powerplant flight research with flying models. $M = 7$ may be considered as the maximal Mach number for this model since the main problem is to study and work out transition regimes from the turbojet engine to the ramjet one, transition from subsonic combustion in the ramjet engine chamber to supersonic one.

The best research variant is to dispose, at least, mock-ups of all the engines contained in the powerplant, on the model. However, if the difficulties resulted from turbojet engine development are taken into account, the investigations may include only reproduction of the channels bringing up a flow to ramjet engines.

A flight trajectory (including the sequence of regimes in time) must correspond to the natural one. Due to the necessity of reproducing operation conditions of, in fact, a series of powerplants, model dimensions have to be increased. Its length ought to reach 12—15 m. Estimations showed that launching the aircraft of such dimensions into the given flight regimes ($M = 3.5$—7; $H = 20$—30 km) by means of rockets starting from a ground-based installation, first of all, needs in development and creation of a complicated rocket system. Therefore more expedient is the usage of a carrier airplane from which the research aircraft with a sufficiently simple system of boosters starts.

Complexity and sufficiently high cost of the aircraft require its non-expandability. Therefore its aerodynamic configuration must meet flight conditions of all regimes: from hypersonic flight regimes to runway landing ones (since parachute landing of the aircraft of such weight and configuration does not guarantee its survivability for new using). So, aerodynamic configuration of this type of aircraft has to be brought closer to the original configuration. Hence, many other problems may be solved during the research process, viz.: more detail study of airframe/engine integration; a series of aerophysics problems (boundary layer transition, shock wave/boundary layer interaction, etc.), making aerodynamic characteristics more accurate, perfecting control algorithms, and so on.

A general view of an aerodynamic configuration variant of this type aircraft is shown in fig. 4.

Tu-95 or VMT airplanes may be used as carrier aircraft. A configuration with the Tu-95 airplane is presented in fig. 5.

Model acceleration after separation from the carrier airplane is carried out with separated solid-propellant and built-in liquid-propellant rocket engines. Aircraft takeoff mass estimation does not exceed

13 tons. One can keep a flight in the quasi-stationary ram regime within almost all the range $M = 3.5—7$ (which is necessary for investigations) by means of selecting the programs of engine operation and aircraft control, both for active and passive trajectory leg.

Thus, to solve more completely the problem of development of powerplants for aerospace and hypersonic aircraft for the future, it is expedient to divide this problem into two flight-test programs which include creation of two different aircraft types, viz.: flying models launched by means of ground-based boosters and flying models launched by means of carrier airplanes.

This flight experiment complex will make it possible to obtain the necessary scientific and technical information within the range $M \leq 14$.

References.

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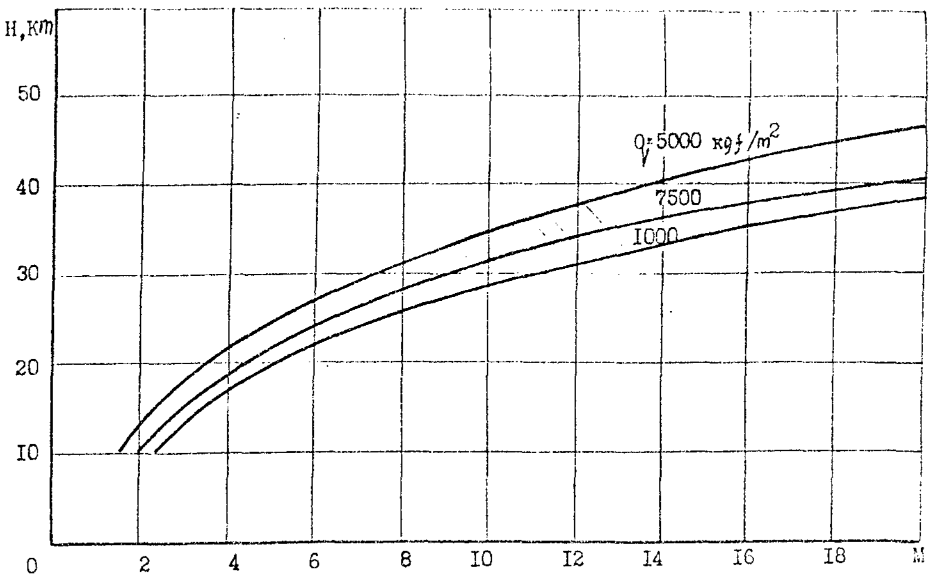

Figure 1

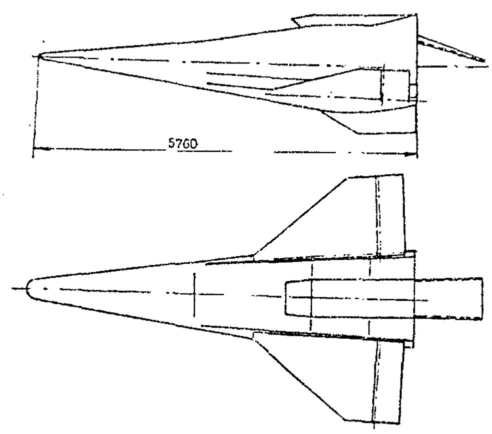

Figure 2

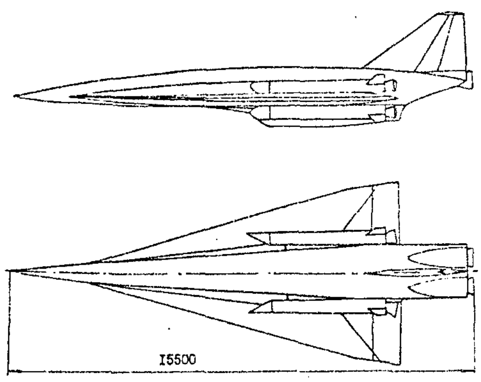

Figure 4

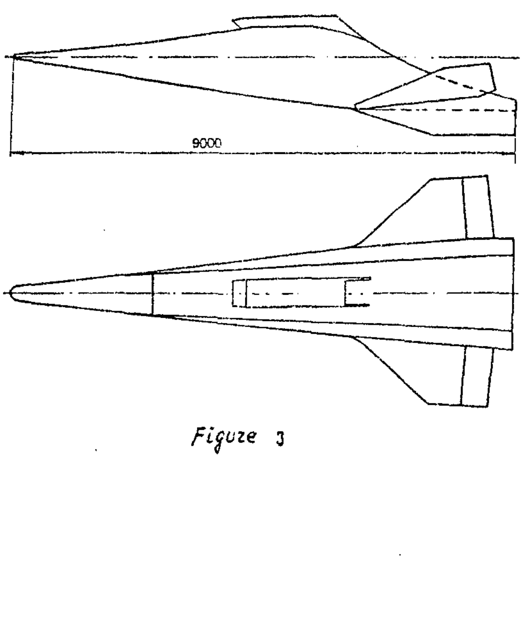

Figure 3

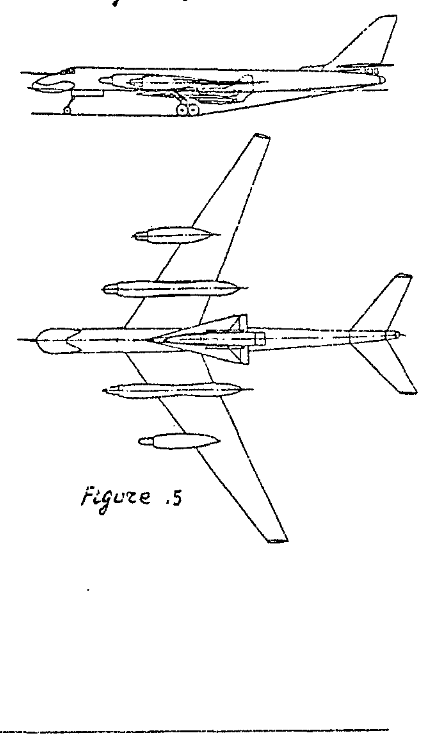

Figure 5

MEASUREMENT FACILITIES FOR AIRCRAFT FLIGHT TESTS

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Modern systems of aircraft measurements (SAM) for flight tests must be highly informative, must have substantial quantity of measured parameters, various ranges of measurements and frequency, compact dimensions, stability of operation and of precision specifications under rigid environmental factors, must be multiplex, self-checking, must allow reconstruction according to flight tasks.

Aircraft on-board sensors operate under especially rough conditions. Therefore, while developing any SAM most reliable and stable sensors must be chosen.

Flight Research Institute (LII), being a leading center of flight tests and research measurements, conducts checking and perfecting of SAM application and some separate measurement facilities (MF) in-flight in order to determine their metrological and operational reliability, functional and electromagnetic compatibility, etc. LII develops MF prototypes, approbates different ideas, SAM formation and MF application principles during flight tests, matching ranges of measurement and frequency, precision, overall dimensions and mass, operational service and flight tasks.

Data on some MF developed in LII are given below.

Piezoelectric vibration sensors of "ВДВ" type designed to measure vibration parameters of aircraft structural parts, of aircraft engines, of on-board equipment under high temperature are used together with vibration measuring equipment of "ПВА"and "АВ6" types.

Vibration sensor "ВДВ-12", with heat-resistance up to 300°C; "ВДВ-17" - up to 250°C to be used under the conditions of high electromagnetic jams; "ВДВ-19"- of heightened sensibility and jam-resistance; "ВДВ-21"-small, highly sensitive, heat-resistant.

"3ДВ6-001"-consisting of three units to measure three inter perpendicular vibration components of aircraft structure under the temperature of +200°C.

These sensors are stable under substantial *g*-load, vibration and shock (100 times upper level of measurement range).

Specification	ВДВ-12	ВДВ-17	ВДВ-19	ВДВ-21	ЗДВ6-001
Measurement range, m/s^2	0...2000	0...2000	0...5000	0...10000	0...5000
Coefficient of transformation, pC/m/s^2	0.6+0.2	1.5+0.5	5.0+0.5	0.5+0.05	0.35+0.05
Basic error, %	6.0	5.0	4.5	4.5	4.5
Frequency range, Hz	2...5000	2...5000	2...9000	2...20000	2...5000
Resonance frequency, kHz	18	17	26	70	50
Temperature range, °C	−50 +300	−50 +250	−60 +200	−60 +200	−60 +200
Complementary error, %	10	10	10	10	10
Lateral sensibility, %	4	4	1...4	5	2
Mass (without cable), g	83	67	21	2	14
Type of coordinating equipment	ПВА-5	ПВА-7	6АВ6-006 8АВ6-001	6АВ6-006 8АВ6-001	6АВ6-006 3АВ6-007

Coordination of vibration measurements equipment "ПВА" and "АВ6" with these sensors is carried out with the help of built-in charge amplifier that allows to conduct vibration measurements in low frequencies' area under high temperature and to use lengthened communication link "sensor-coordinating device" without any marked sensibility change of measurement channel.

Equipment of "ПВА-7" type provides measurement of vibration's acceleration, speed and travel simultaneously on 6 channels and is operationable under conditions of high electric and electromagnetic jams. It is used together with "ВДВ-17" sensors (of two-wire connection diagram). Built-in calibration device (beside operability checking and calibration of measurement channels) determines operability of the through channel, i.e.of the link "sensor-cable-coordinating amplifier". Check-unit displays equipment's state (on 3.5-decade indicator) during its preparation for work.

Equipment of "6АВ6-006" and "8АВ6-001" types provides vibroacceleration measurement on 6 and 8 channels accordingly. Built-in calibration device allows operability checking of measurement amplifiers.

Equipment of "ПВА" and "АВ6" types is easy in operation, stable under environmental effect; no special training of maintenance personnel is needed.

Vibrometer "AB6-009" with 3.5-decade display is intended for verification of vibration measurement equipment, of vibro-calibration device of "BKY-01" type under normal conditions and for exact measurements of amplitude levels of sinusoidal vibro-accelerations. Complete set also includes piezoelectric vibration sensor of "ВДВ-19А " type.

Measurement range can be chosen according to measurement tasks with the help of the switch on the switch panel. Frequency range is determined by the chosen filter type that can be built-in at customer's request.

Standard output ensures equipment coordination with the systems of measurement information registration on magnetic tape or with PC through an additional device.

Specification	ПВА-7	6AB6-006	8AB6-001	AB6-009
Number of channels	6	6	8	1
Measurement range (number of subranges)				
— of vibro-acceleration, m/s^2	0...640(5)	0...320(6)	0...480(6)	0...10
— of vibro-speed m/s	0...160(4)	-	-	-
— of vibro-travel, mm	0...4(3)	-	-	-
Frequency range, Hz				
— of vibro-acceleration	5...2500	2...5000	2...5000	30...1000*
— of vibro-speed	5...1250	-	-	-
— of vibro-travel	5...156	-	-	-
Output signal range, V	3±3	3±3	3±3	
Basic error, %	2.0	2.0	2.0	4.0**
Operating temperature range, degrees °C	−50...+50			0...+35
Complementary error, %	5.0	5.0	5.0	1.0
Power	27 V of DC			
Overall dimensions, dm^3	7.9***	4.6****	4.8	0.8
Total mass, kg	5.0	2.7	3.0	0.65

* irregularity of amplitude-frequency specifications not worse than 1.5 %,

** of vibrometer with sensor

*** including check-unit (2.3 dm^3) and power supply unit(1.4 dm^3)

**** including power supply unit (1.4 dm^3)

Static and dynamic pressure can be measured with the help of small flat piezoresistive sensors "ДД3-020" (no-drain method) or thermostat modules of "16ДД3-021" sensors.

Sensitive element of vibrometers is made of silicon monocrystal in the form of a membrane, in the surface of which there is a strain-gauging bridge of four semi-conducting strain resistors. Vibrometers are shock-proof and *g*-load resistant, have low sensitivity to case deformations and small temperature sensitivity error within the whole operating temperature range. Low output resistance of the sensor makes for its high jam-resistance.

Sensor "ДД3-020" can be glued to lifting surface together with profile fairing; there is no need to perforate object of research surface. Because of sensor's flatness, its application does not lead to noticeable distortion of physical process during research. Case of "ДД3-020" sensor protects sensitive element from damage and allows its multiple use. To attach this sensor to the surface hermetic adhesive of "У-30МЭС-5" or "ВГО-1" type is recommended. Profile fairing is made of cloth saturated with hermetic adhesive of "У-30МЭС-5" type.

"ДД3-020" sensors are not sensitive to surface deformation, have electromagnetic jam-resistance and low inertness. Within temperature range from −55 to +70°C error of static pressures measurement is no more than 2%, of dynamic pressures — no more than 6%.

Requirements to connection cable laying are analogous to those in strain-gauging.

Multi-channel coordinating equipment of "12ДД3-026" type provides sensor's power supply, its output signal coordination and normalization. Possible connection cable length - more than 50 m.

On base of sensitive elements used in "ДД3-020" sensors and of improved elements (with strain-concentrators and of heightened sensitivity) it is possible to make sensors of different design with various measurement ranges, including high acoustic pressure transducers, group thermostat transducers for measuring static pressure fields, transducers with built-in amplifiers, etc.

Specification	ДД3-020-10	ДД3-020-50	ДД3-020-100
Measurement range, kPa	0...10	0...50	0...105
Transformation coefficient (optimum DC 5...10 mA), mV/kPa	2.0	0.8	0.5
Basic error, %	1.0	1.0	1.0
Non-linearity and hysteresis, %	0.6	0.6	0.6
Frequency range, kHz	0...2.5		
Temperature range, °C	−55...+100		
Temperature error, %/°C			
of "zero"	0.15	0.15	0.15
of sensitivity	0.02	0.02	0.02
Output resistance, Ohm	1000...4000		

Overall dimensions, mm	25×18×1.6
Mass, g	4.5

Equipment "16АД3-026" is intended for static pressures' measurements on aircraft lifting surface with the help of small attachable piezoresistive pressure transducers.

Equipment consists of 16 absolute pressure transducers of "ДД3-020" type and multi-channel coordinating device "16СД3-026". This equipment provides for automatic pre-flight balancing and measurement channel checking.

Equipment "64АД3-026" is intended for static pressures measurement on aircraft lifting surface to determine aerodynamic loading with the help of group thermostat piezoresistant pressure transducers "16ДД3-022".

Equipment consists of 1 to 4 group transducers "16ДД3-022" and coordination module "64МС-022".

Structure scheme of equipment "64АД3-022" is shown in fig. 1.

Modules "16ДД3-021" consist of 16 pressure overfall transducers, electronic channels multiplexer and amplifier, power supply of sensors' bridges, diagram and thermostating elements. Measurement is conducted in 16 drained holes connected to group transducer by drain-pipes (diameter 1.6—2.5 mm).

Specification	"64АД3-022"	"16АД3-026"
Number of measurement points	16...64	16
Measurement range, kPa	0...105	−5...+5 −10...+10 −20...+20 −40...+40
Frequency range, Hz	0...5	0...5
Basic error, kPa, maximum	0.15	0.3
Complementary error, kPa, maximum	0.25	0.8
Output voltage range, V	±10.0	3.0±3.0
Power voltage, V (of DC)	27±3	27±3
Volume, dm^3	6.4	2.0
Mass, kg	6.0	1.6
(group transducer included)	0.5	-

Digital output (standard interface RS232) is provided.

Strain-gauging 16-channel modules-amplifiers "16ТМЦ" (output — successive 10-bit code) and "16СУ27" (analogous output, voltage 0..6V) are developed to determine aircraft elements and units loads. In their metrological and operating specifications they are inferior to none of similar foreign samples. Measured deformation ranges are

$(\pm5...\pm40)\cdot10$. Operating frequency range 0...125 Hz. Automatic balance of strain-gauging bridges and remote switching of amplification coefficients are provided. Modules' volume is 0.14 dm^3/channel. Modules can be united in a system. Possible number of modules in the system is from 1 to 12.

To determine load of aircraft engine's rotating propellers is developed "Vortex" compact system with telemetric transmission of information on board of aircraft. Strain-gauging resistors, thermoresistors and antennas are glued to propeller's blades. Modules with amplifiers, multiplexers, analog-digital transducers, radio transmitter, power supply source are attached to propeller hub. Radio receiver, de-multiplexers, power supply unit, information recorders, digital-analog transducers, PC are installed in aircraft cabin. Number of measurement points — 95, number of simultaneously measured parameters — 31. Range of measured deformations is $\pm20\cdot10$. Frequency ranges are (0...250; 0...2000) Hz. Power consumption of propellers' modules is 1 W maximum. They are g-load-proof up to 6000.

To explore strain on constructions made of titanium "BT-4", steels of heightened strength "BΛ-1" and "BHC-5" adhesive wire strain-gauging resistors "ДHMT-8-400" are developed; for structures made of stainless steel "X18H9T" — resistors "ДHMT-16-300"; they can operate in temperature ranges, accordingly, $-30...+400°C$ and $-30...+300°C$. Resistance — 200 Ohm.

For aerodynamic friction measuring special measurement facilities are developed basing on two acting principles: heat method with hot-film sensors and direct weight method with electromechanical sensors.

Equipment "TA-003" is based on heat method and consists of hot-film sensors of "ДТП" type and coordinating device "CTA-001". Coordinating device is a constant temperature anemometer. Equipment is intended for measuring average and fluctuating components of shear stress, for determining areas of laminar turbulent flow transition, boundary layer separation and joining for exploring flow about of aircraft surface under natural conditions or in wind tunnel.

Sensitive element of "ДТП" sensor is made of nickel film applied on quartz base and is protected against mechanical damage and smearing. Two sensor types are developed: "ДТП-001" and "ДТП-002".

Sensor "ДТП-001" is installed in a hole flush mounted with explored surface that does not lead to any flow disturbance.

Sensor "ДТП-002" is glued to the surface of research object. Thus, no interference in the structure is needed and multichannel measurements of skin friction becomes possible.

Specification	ДТП-001	ДТП-002
Shear stress range, Pa	0.1...150*	

Sensitive element dimensions, mm (width ×length)	0.05×1.00	
Electric resistance under 20°C, Ohm	10...15	
Temperature coefficient of resistance, %	0.4	
Maximum temperature of sensitive element, °C	200	
Maximum temperature of the flow, °C	120	
Overall dimensions, mm	∅10×40	3×3×0.5
Mass, g	8.0	1

* Sensors calibrating is conducted with a constant temperature anemometer.

Constant temperature anemometer "CTA-001" has all the necessary controls that allow wide range of changing its functional modes and using other thermo-anemometrical sensors of shear stress, speed and flow rate. It has balance of active and reactive resistance of cable with length up to 40m that connects sensor to thermo-anemometer; measurement of sensitive element resistance within 25 Ohm, sensitive element given overheat setting are provided. For thermo-anemometer's tuning square-wave generator is built-in.

Structure scheme of equipment "TA-003" is given in fig. 2.

Friction equipment "AH2-008" is developed according to direct weight method of measuring local force of aerodynamic friction. It is intended for measuring friction average values. Equipment consists of friction sensor, compensatory sensors of temperature, of linear g-load, of angular velocity and acceleration influence on measurements readings, coordination device.

Structure scheme of equipment "AH2-008" is shown in the fig. 3.

Friction sensor's mobile element is installed strictly flush with the research object's surface. Sensors' mobile systems are balanced along three inter perpendicular axes.

Specification	CTA-001	AH2-008
Measurement range, N	0...0.03(0...3.0)	
Output signal, V	0...6	
Basic error, %, maximum	4	4
Complementary error, %, maximum	10	8
Temperature range, degrees C	-50...+110	-50...+110
Diameter of friction sensor's mobile surface, mm	-	17
Overall dimensions, mm		
— of friction sensor (compensatory)	see the	∅52×100
— of coordinating device	chart above	114×80×75
Mass, kg		

— of friction sensor (compensatory)	see the	0.24
— of coordinating device	chart above	1.0

It is advisable to use the LII experience in instrument-making for aircraft flight tests also in other branches of industry to explore and test different technical objects operating under rough conditions.

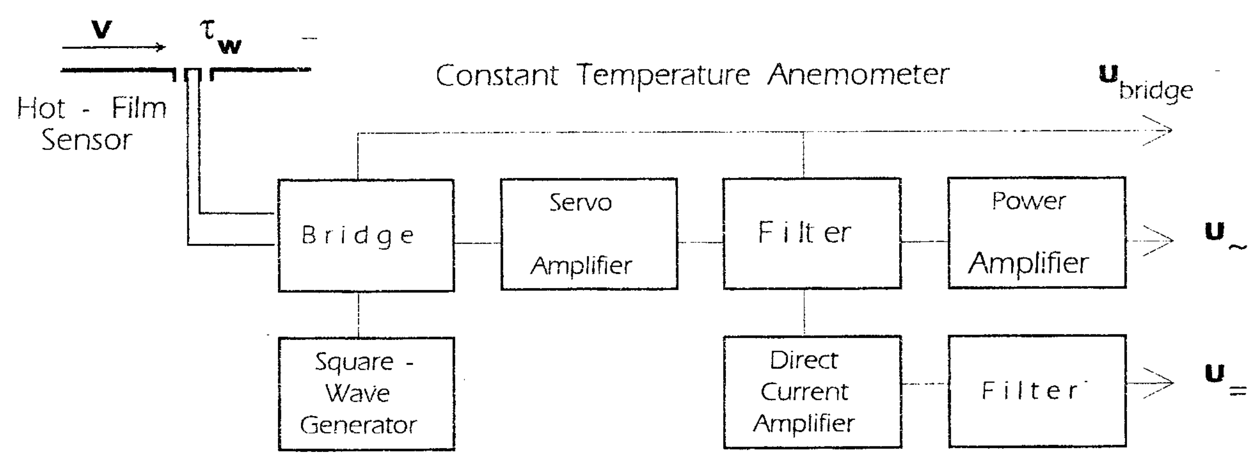

Figure 2. TA-003 Equipment Structural Scheme

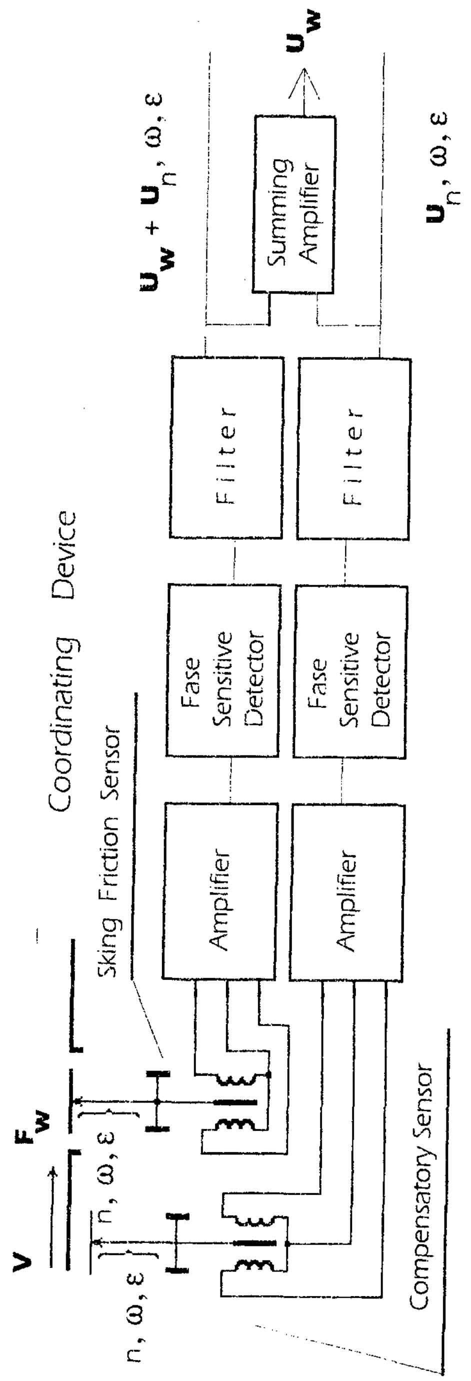

Figure 3. AH2-008 Equipment Structural Scheme

FLYING TESTBEDS AND GROUND EXPERIMENTAL BASE FOR ENGINES AND POWERPLANTS FLIGHT TESTING

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The Flight Research Institute (LII) has available both experimental base to carry out complex flight tests of engines and powerplants, and skilled specialists who develop the experimental base and fulfil every stage of the testing.

The experimental base includes the following:
— specified flying testbeds for engines and powerplants flight tests;
— ground laboratory and bench units, providing flight tests and complex research;
— flight test measurements, data processing and analysis facilities;
— specific equipment to provide the technique of ground and flight tests of engines and powerplants.

The paper considers the general features of the experimental base, taking into account its contemporary state and the prospects of engine design,(in Russia and CIS) and gives some examples of its usage.

At present LII has eleven specified flying testbeds to carry out the flight tests of gas-turbine engines power plants and their units. These flying testbeds were developed from the IL-76, An-12, Tu-22, Tu-16, Tu-134 airplanes and Mi-17 helicopter. These testbeds allow to conduct the following testing:
— turbofan and propfan engines of any dimension with the thrust on the ground up to 25—30 t;
— powerplants and their engines in a maneuver aircraft configuration, including afterburning turbojet and afterburning turbofan engines;
— auxiliary powerplants in aircraft configuration, modeling the extreme temperature conditions;

The flying testbeds and ground experimental base enable to evaluate:
— gas-turbine engine performance, including a case of the high-altitude test bench absence;
— the performance of fuel systems with given operational characteristics of gas and liquid fuel and the possibility of the standard gas fuel utilization;

— air pollution caused by aircraft engines;
— complex interaction of the functional characteristics and failure safety of the engines and powerplants control loops, considering their jam-proof and electromagnetic compatibility;
— engine units and propfan loading in flight conditions;
— airplane and helicopter engines ingestion protection;
— fire fighting systems.

The Flight Research Institute's experimental base allows to conduct flight tests of the engines and powerplants of the aircraft belonging to Russia and the Ukraine in accordance with the tendency of the civil aviation engine design, including dual-flow engines with high thrust (up to 35—40 tons) NK-44 and D-18T1, and propfan engines D-27 and NK-92 without expensive high-altitude test benches, which possibilities are limited.

For the latest five-six years D-18T, D-236, PS-90A and it derivatives (including the one without the inter shaft bearing TV-7117, D-27, VD-100) have been tested.

The outstripping and escorting flight tests of the D-18T engine on IL-76 flying testbed (fig. 1) are a typical example, showing both the structure and the volume of the flight tests. There was no high-altitude bench testing (fig. 2).

The flying testbed was equipped with an engine, having an airplane cowling into the airplane nacelle with a real air intake. The real performance of air bleed-off and mechanical energy were provided. 405 flights took 1252 hours. The obtained results allowed to receive the basic performance of the powerplant and its subsystems in the different climatic conditions, which imitate the ones of the operation. An average flight duration was over three hours, the maximum duration was about seven hours. The gained experience can be successfully used further.

Flight tests on the flying testbed prepared the maiden flight and the first stage of the "Ruslan" prototype airplane flight tests. Besides, the basic data (100%), endurance (100%), strain-gauging (100%), observability (100%), engine control system (80%), starting (70%), oil system (60%) flight tests were completely or basically carried out.

A large volume of the outstripping flight tests of GE-90 engine was fulfilled on the B-747 flying testbed (over 200 hours, 41 flights without high-altitude bench testing.

Flying testbeds complete utilization at powerplant certification can be shown at PS-90A engine flight tests. The engine forms a part of IL-96-300 and Tu-204 airplane powerplants. To certificate the PS-90A engine, 105 flights were carried out, including 61 flights according to chapter 656 of Airplanes Airworthiness Standards (NLGS-3) "before mounting on the airplane", and 44 flights according to chapter 6.6 of NLGS-3. For example the IL-76 flying testbed was used to complete a special program "Flight certification tests of P-S-90A engine with the state bench tests configuration aimed at estimation of its Engine Control System-90, the Onboard Engine Monitoring System-90, and the Complex Information Warning System-1-2M" in accordance with the requirements of chapter 6.6 NLGS.

Taking into account that standard programs of gas-turbine engine powerplants flight tests were developed and worked out for every expecting flight regime of operation, in our opinion the flying testbed could take upon about 40% of the work to be done to certificate the gas-turbine engine.

The electronic control systems of the gas-turbine engines enabled to carry out onboard flight tests, using special equipment and systems, acting on the gas-turbine engine by simulation of the internal and external disturbances through the control system, which is able to change the control programs.

LII worked out and applied the automated processing and analyses of an engine and its units performance in the real time onboard and on the earth using the telemetry. A digital workstation (with the expert system) for the certification principal engineer is nearly completed.

Moreover, a considerable interaction and interdependence of different engine electronic control systems, as well as the necessity to develop and optimize the functional and protection characteristics of the electronic and reserve engine control system require the flying testbeds for the outstripping complex research of the functional characteristics and the failure safety of the mentioned systems, providing at the same time the gas-turbine flight tests safety. As shows the example of the PS-90A engine, the safety of the tests is ensured by the regular reliable engines of the flying testbed, which allow to fulfill the extreme regimes, and analyze the engine performance onboard in real time with automated control of the experiment and the extreme regimes indication for the principal engineer.

The specialized PC and modern techniques of the experiment automated analysis and control used at flight tests decreased the numbers of flights and at the same time considerably increased their effectiveness, that reduced the cost of the total test cycle.

In order to test the auxiliary powerplants in the airplane configuration a multi-purpose flying testbed on the An-12 airplane base is used. This flying testbed allows to test these engines and their systems in the airplane configuration, completely simulating the extreme climatic conditions (extremely high and low temperature and air humidity on the LII aerodrome).

The flying testbeds for ecological research were developed on the base of Mi-17 helicopters and Tu-134 and An-12 airplanes.

The Mi-17 and Tu-134 flying testbeds are equipped with the air bleed-of system and CO, $(N_2O_2)_x$, H_xC_y, SO_2, SO_3 and spray concentration system. They are indented to evaluate air pollution, for the wake natural physical and chemical process research.

The testbed on the base of An-12 airplane is equipped with the facility, allowing to survey the surface of the Earth in a wide range of emission (from the radar one up to the ultraviolet emission).

These flying testbeds allow to fulfil the ecological monitoring of the earth, water and air in any region of Russia. They have been successfully used both in the Russian Federation and abroad.

The IL-76 flying testbed will be used at complex estimation of the failure safety of the TV-7-117 engine control system and IL-114 powerplant onboard systems. The available flying testbeds Tu-16 in operational condition enable to test engines of various type (afterburning turbojet, afterburning turbofan engines) with the thrust up to 16—18 tons, and can be used on different purposes (Tu-16 testbeds have conducted thousands of experimental flights, tested over 20 new engines).

The next step is to develop a flying testbed for NK-92 engine tests on IL-76 airplane. Different to the conventional in the Soviet Union technique when a testbed used to be created for a prototype engine, intended for a specific airplane (or a line of airplanes) that determined the configuration of the flying testbed typical for the airplane, i.e. air intakes, airplane units, power and air bleed-off, etc., now, due to the lack of the data on new airplanes layouts, the configuration of the powerplant with NK-92 engine will be applied to IL-76 airplane. The work to be done has been preliminary distributed among the engine design bureaus, LII and possible customers of NK-92 engine — the Ilushin Aviation and Tupolev Aviation scientific complexes. The problems of developing the required flying testbed have been considered as well. A particular attention here was paid to the following:

— telemetry current cutoff to monitor vibration in the propfan blades;
— flight engine thrust and fan airflow rate determination;
— the techniques of processing, analyzing, visualization and recording the experimental results by means of personal computers.

LII has available an operating dynamometrical thrust bench, which consists of three extensible dynamometrical platforms, set rails, fig. 3.

The bench allows to determine the powerplant thrust of the airplane, weighing up to 70 t. The tract width of the tested airplane is 1—11 m., the wheel base is 2.5—14.5 m, the double wheel or bogie width must not exceed 1.25 m.

The platforms suspended on the steel ribbons reproduce the forces from the wheels to the strain-gauge transducers TVS-2, the set of them measure the horizontal component of the force of the installed airplane from 1 to 20 tons.

The dynamometrical thrust bench allows to determine the following:

— the airplane starting powerplant real thrust;

— the input and output device influence on the engine thrust;
— the dependence of the powerplant units structure change, (warm-up, etc.) on the thrust performance;
— the powerplant starting dynamic characteristics.

The mean square error of the powerplant thrust on the LII's dynamometrical bench is ±1%. The maneuver airplanes of the Sukhoi, Mikoyan and Yakovlev design bureaus were subjected to the tests on the bench, transport and passenger airplanes, such as Yak-42, An-72 and the light passenger airplane "Gzhell" (fig. 4).

LII developed and applied a moment technique to determine airplanes starting trust at these aerodrome basing using a compact measuring platform, bearing the vertical force of the rose wheel (fig. 5).

The experimental base has a wide range of possibilities in research, testing and working out powerplant prototype systems, onboard fire fighting facilities, and separate units of these systems and facilities.

The fuel properties, fuel systems research and tests are carried out on the IL-76, Tu-154 flying testbeds and the fuel benches, which include :

1. A bench for high-altitude and hydraulic tests of the fuel system units.
2. A bench for determination electrostatic discharges at refueling system testing.
3. A bench for the verging anti flame system testing.
4. A complex of unit to prepare fuel for the fuel systems climatic testing:
— with fuel heating up to +60 ,
— with fuel cooling to −60 ,
— with fuel watering up to the limiting ratio,
— with fuel enriching by a neutral gas.
5. The approved devices and the instruments, monitoring the fuel and gas of aircraft tanks over fuel space at ground and flight tests.

The flying testbeds and the bench allow to conduct tests on the earth, at airplane climbing, descent and horizontal flight. Fuel for the experiments may be of low or high temperature have a given steam density and elasticity, watering electroconductivity, content of gas. Specialized fuel trucks and a set of various devices for gas and liquid chemical analysis support the flight and bench tests. Fueling and the fuel transfer electrostatic safety as well as liquefied gas fuel system units functional testing are being carried out on the test benches.

Along with the tests of the fuel systems and their units the available instrumentation and measuring equipment allow to completely test and study powerplants oil system units and the fire-fighting means both on the earth and in flight. A bench to study the fuselage fire fighting is being now equipped as well.

Besides the prototype aircraft development the experimental base and high skills of the specialists in powerplants and working fluids allow to create and apply new advanced methods and techniques, to study the normative and systematical documentation.

A special equipment of the experimental base allow to study the gas-turbine engine runway ingestion protection. The following facilities are used to investigate the vortex cord, to select the vortex cord protection parameter of gas-turbine engines:

— force sensing device, measuring the vortex cord force on the aerodrome surface;
— measuring plates, evaluating the rarefaction caused by the vortex cord;
— simulators of the runway surface level for the study of the vortex flow under the aircraft prototype engine air intake;
— wind making facility;
— a bench simulating air intakes on the operating engine, for studying influence of the shape and slope angle of the air intake inlet section upon the vortex cord intensity.

In order to investigate the process of solid and liquid bits scattering from under the wheels, to select the engine protection parameter, and to estimate the effectiveness of the developed protection measures the following is used:

— ensnaring nets, set on the air intake inlets;
— special segments on the runway with poured sifted fractions of granite or with stacked baths, filled with water of different levels or with diluted soil of slush or melted snow;
— a special bench for determination of the bits scattering region from under the wheel, depending on the wheel boot shape and on the speed at which the wheel passes the segment;
— video-tape recording equipment to record the jets, scattering from under the wheels when an airplane passes the experimental baths;
— airplane speed indicators in the range of 60—200 km/h, and indicators of more precise small speeds by the wheel rate.

The investigation of the reverse jets, raising up at an airplane roll out with the thrust reverse is one more reason, causing the bits ingestion into the air intakes. Scheduling the engine protection of the reverse jets and the admissible speed range at thrust reversed roll out LII uses:

— reverse jets visualization by means of either oil steam setting the oil injectors into the hot gas jet on the thrust reverser outlet, or by means of harmless loose powder (such as talcum powder or dry snow poured over the runway);
— fast-response temperature sensors or the total pressure fluctuation an the air intake inlet sensors for recording the initiation of the reverse jets ingestion;
— video tape recording equipment to record the spreading and ingestion of the reverse jets at airplane roll out;

— ensnaring nets, set on the air intake inlets;
— special segments of the subway with poured fractions of granite.

For experimental determination of the distances and intervals at a group take-off, that would be safe in terms of ingestion into the air intakes of the wingman airplane of the aerodrome obstruction bits, raised by the leader jets, the ensnaring nets, imitating the wingman air intakes are set behind the leader. After the leader nozzle the runway segment with granite fractions is placed.

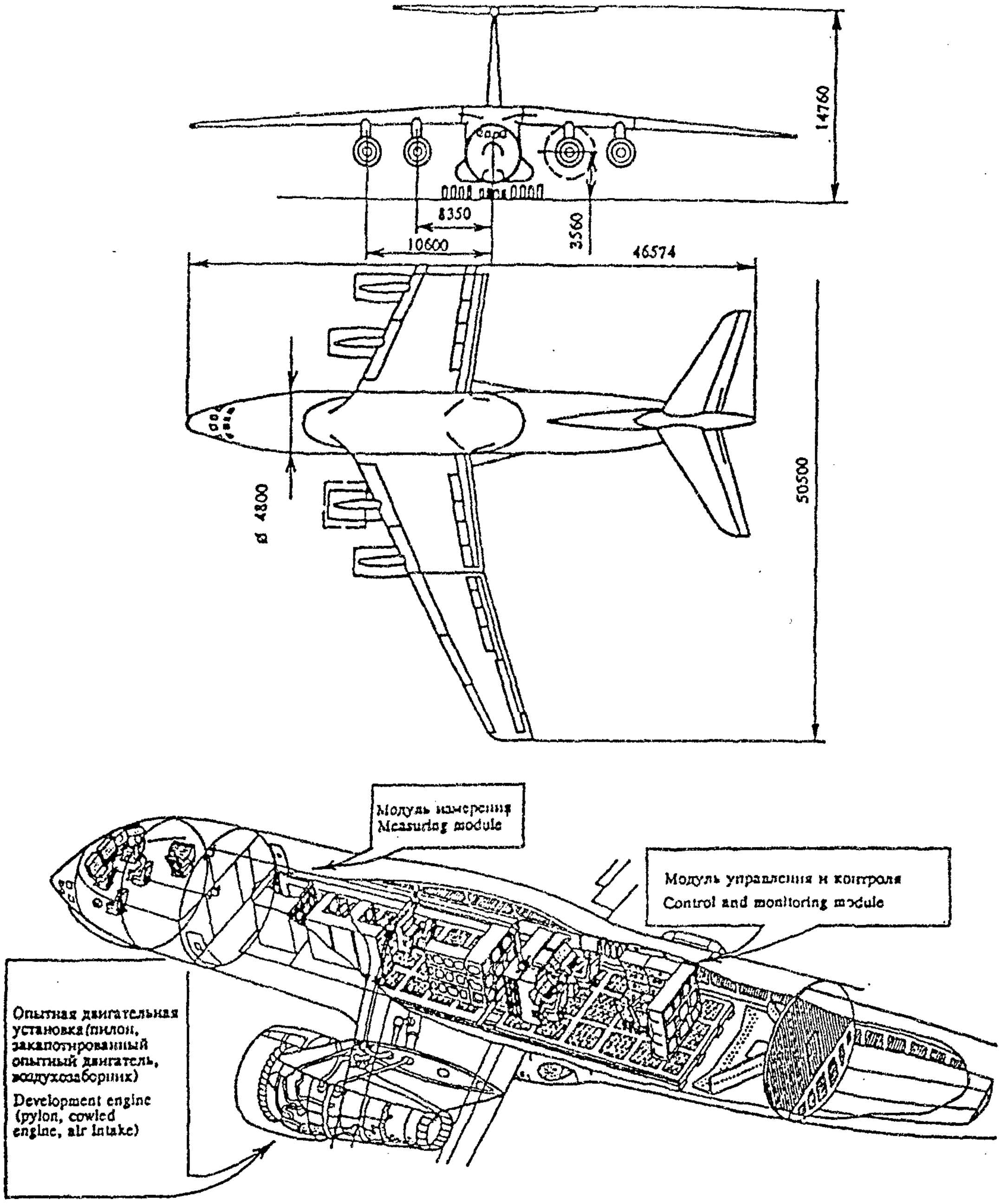
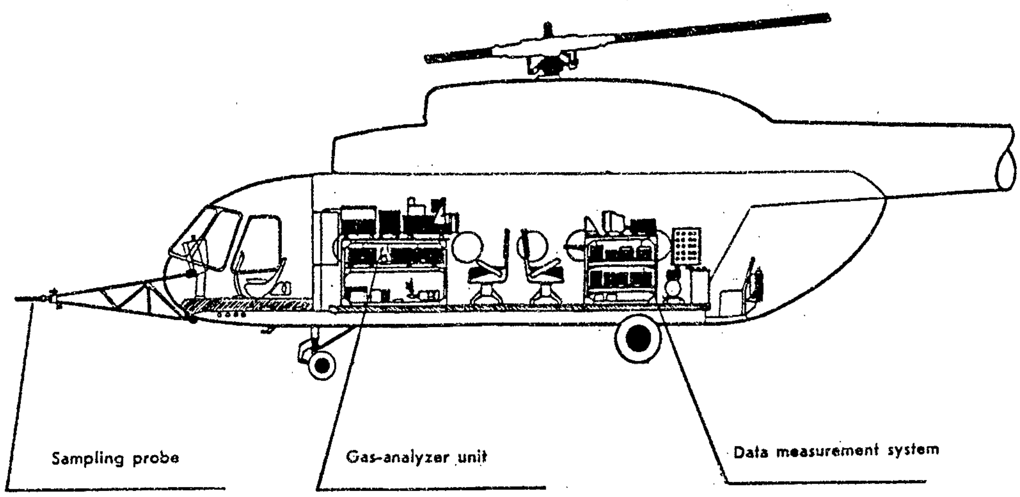


Figure 1. Scheme and basic specifications of flying test-bed IL-76 for tests of gas turbine engines.

ОСНОВНЫЕ ЛЕТНО-ТЕХНИЧЕСКИЕ ДАННЫЕ ЛЕТАЮЩЕЙ ЛАБОРАТОРИИ

Максимальная высота полета	12000 м
Максимальная скорость полета	600 км/ч
Максимальное число М полета	0,77
Минимальная скорость полета	280 км/ч
Потребная длина ВПП	3000 м
Максимальный взлетный вес	170000 кг
Максимальный посадочный вес	140000 кг
Максимально допустимая вертикальная перегрузка	-0,3÷2,0, ед.
Максимальный крен	30 град.
Максимальный угол атаки	15 град.
Максимальная угловая скорость	5 град/сек

MI-17 LIZA FLYING TEST-BED FOR ATMOSPHERIC CONTAMINATION ECOLOGICAL INVESTIGATIONS



TU-134A FLYING TEST-BED FOR PHYSICAL AND CHEMICAL PROCESSES OF ATMOSPHERIC POLLUTION RESEARCH

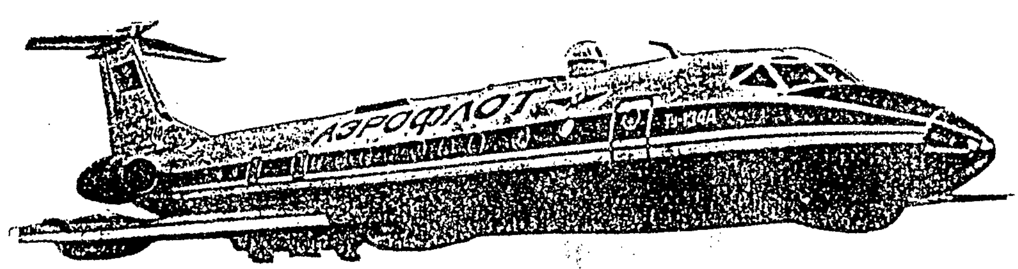


Figure 2. Flying test-beds for ecological investigations.

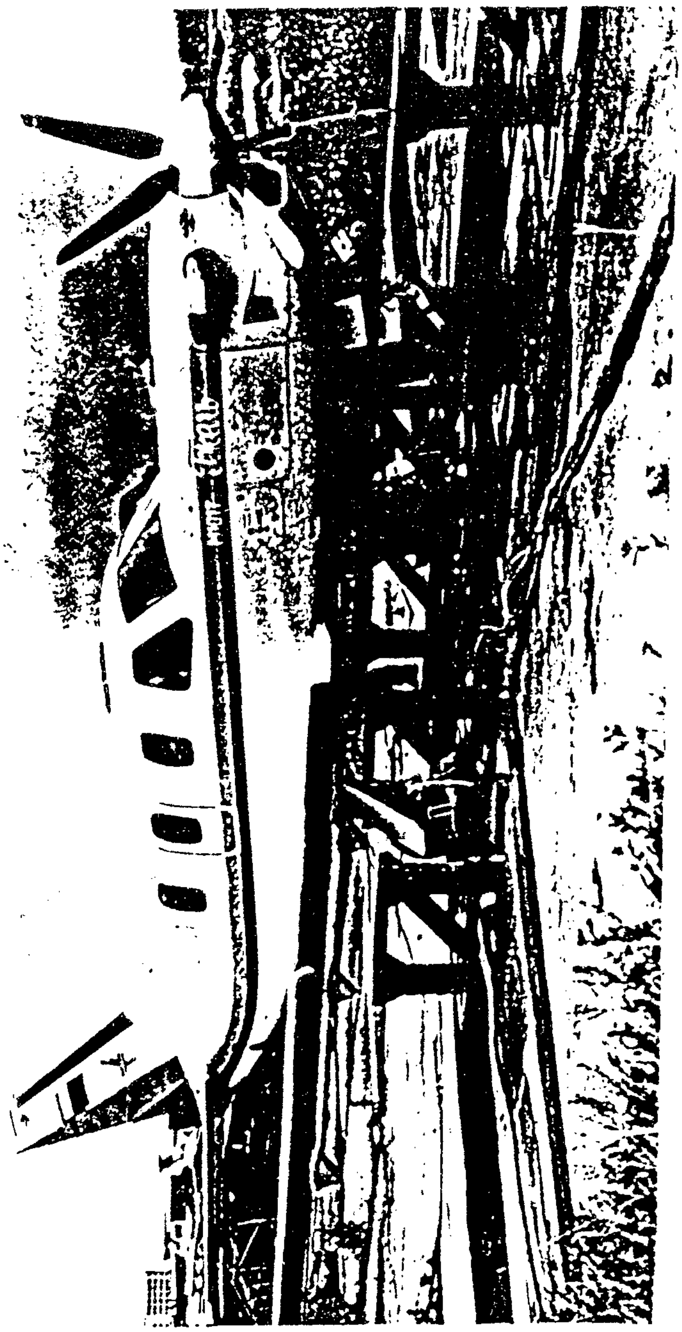
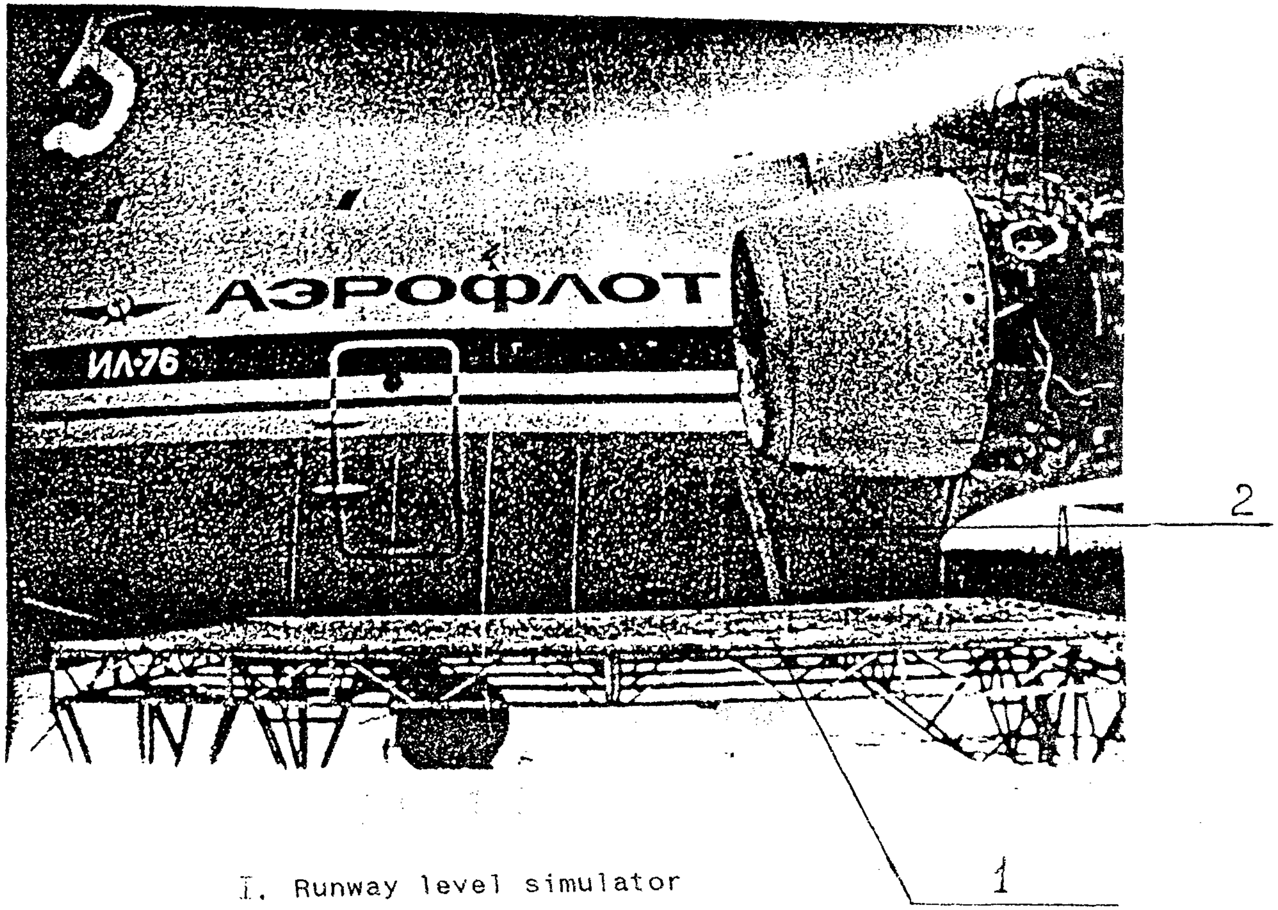


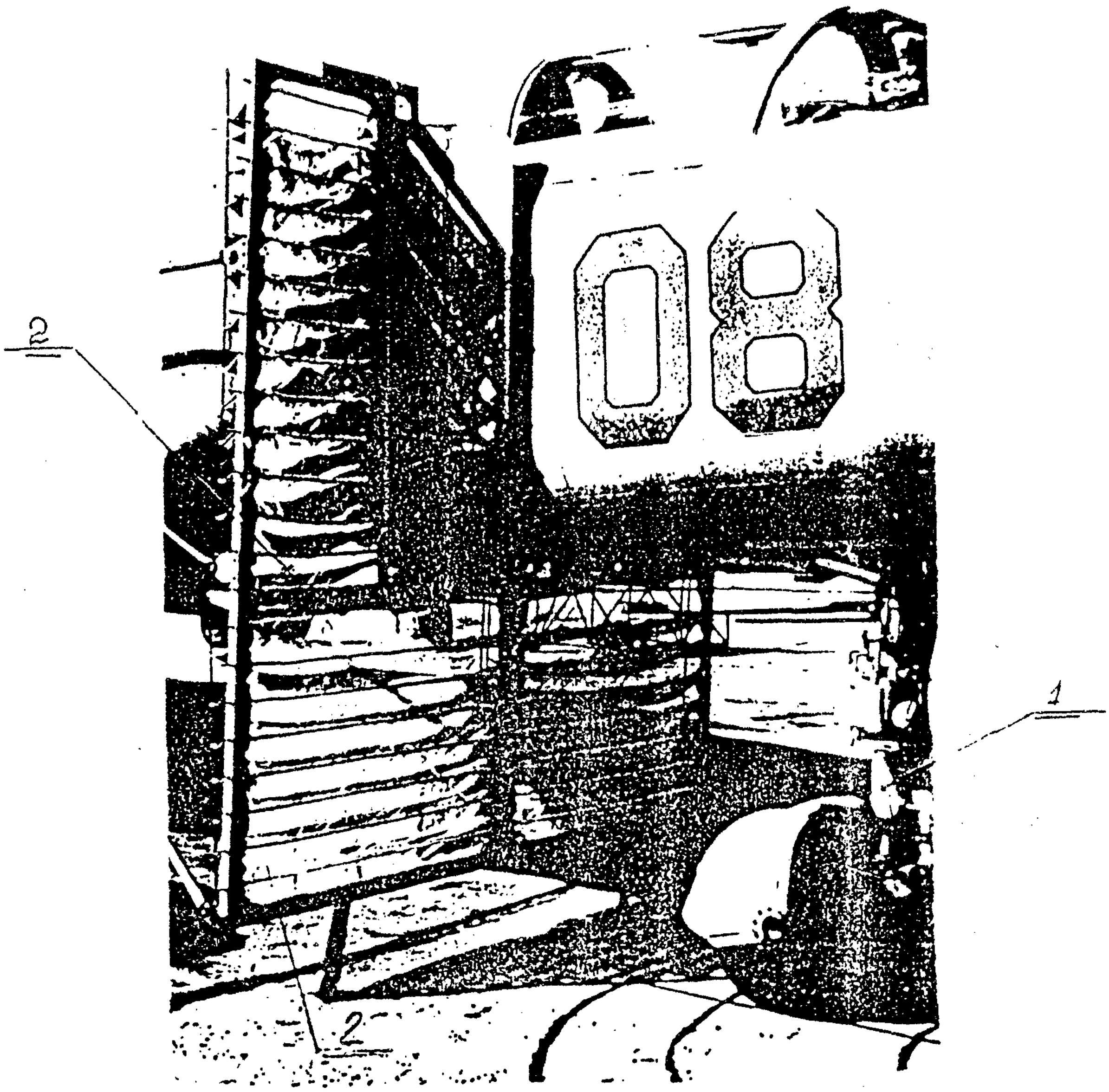
Figure 3. Light passenger aircraft ''Gzhel'' mounted on dynamometric test-bench for thrust measurement.



1. Runway level simulator

2. Vortex cord

Figure 4. Runway level simulator for research of vortex cords under engine inlet.



МиГ-23

I. Nose gear

2. Nets

Figure 5. Nets mounted in front of engine inlet.

AIRBORNE MEASURING SYSTEM (AMS) FOR AVIATION POWERPLANT TESTS USING THE GROMOV FLIGHT RESEARCH INSTITUTE FLYING TESTBEDS (FTB)

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The scientific and technical level and the completeness of flight test results for aviation powerplants and their gas-turbine engines are in many respects determined by the AMS composition (elements), functional capabilities and quality. Modern AMSs are complex integration interconnected components, among which the major ones are primary (measuring) transducers, secondary transducers (converters), recording equipment, means of data processing and display.

Major volume of the AMS measurement information is recorded by magnetic tape recording devices (MTRD). A part of the information converted into the digital code, as well as digital data from the standard airborne control systems are processed, displayed and stored in the processed form on diskettes directly in flight conditions on board the flying testbed in real time. Another part of the information is down-linked using a telemetry system, where it is processed by more complicated algorithms in real time, displayed and monitored by specialists who can give the FTB crew necessary recommendations and introduce corrections in the course of the mission (performance).

The measurement information that does not need processing is re corded with light beam oscillographs (LBO). The oscillogram-recorded information is used to logically interpret the character of time history of some values for a limited number of parameters.

A variety of parameters being measured, a wide range of their value changes, a high accuracy required, increased requirements for resistance to external non-informative value effects — all results in application in AMSs of multiple sensors, secondary transducers, recorders different in purpose, principle of function and other characteristics. These devices , along with the commercial products, include widely used apparatuses built in LII, which feature high metrological qualities and are capable of operating under severe operational (service) conditions with significant levels of electromagnetic interference and other destabilizing factors. These are MTRD 16 CM3 and 14 CKM, apparatus for measuring vibration characteristics — ПВА-7 that uses piezoelectrical pickups ВДВ-17, equipment to measure air flow pressure fluctuations — ПАД-5 with pickups ДД6-007, high-frequency and static-dynamic strain measuring equipment — 12СУ-18, 12СУ-17, 16СУ-27, a statistical characteristics computer — ВСХ-1, a dispersion meter — СВД impulse generators - 6УРС-1, 1ГІРТ-1, and other equipment produced as prototypes. Table 1 shows some AMS characteristics, intended for aviation powerplant flight testing on FTB

Table 1.

AMS characteristics for FTB aviation engine flight testing on FTB.

Characteristic	Value
Number of parameters measured, pcs.	800—1200
Measurement range of measured values being, % of nominal.	5—120
Range-transformed measurement limiting error, %	±0.2—±2.0 for low-frequency measurements. ±5.0—±15.0 for high-frequency measurements.
Sensor (pickup) thermal resistance, °C	– 50— + (250—350)
Sensor (pickup) vibration resistance, vibration acceleration units.	5—15
Number of sensor (pickup) types, pcs.	up to 40
Number of recorded parameters by AMS, pcs.	800—1200
Number of parameters recorded by MTRD, pcs.	up to 200
Number of parameters processed on board FTB in real time, pcs.	250—500
Number of parameters down-linked by the telemetry system, pcs.	250—300

Table 2 gives the limiting error values (±3δ) attained with modern AMSs in powerplant testing on FTB.

Table 2.

Parameter measurement error values attained in powerplant flight testing on FTB.

Parameter	Limited reduced error, %
FTB flight altitude.	±0.2—±0.7
FTB flight speed.	±0.5—1.0
Air stagnation temperature.	±1.0
Linear *g*-load.	±0.5
Engine rotor speed.	±0.2
Fuel consumption.	±1.0
Powerplant air passage (section) pressure.	±1.0
Air, oil, hydraulic fluid pressures.	±2.0
Turbine exhaust gas temperature (EGT).	±1.5

Compressor air discharge temperature.	±1.5
Fuel, oil temperatures.	±2.0
Displacements.	±1.0
Vibration characteristics.	±5.0—±15.0
Mechanical stress.	±2.0—±10.0
Air pressure fluctuations.	±5.0—±7.0

Measurement information obtained in powerplant flight testing can be taken divided in terms of the quick-response principle into low-frequency (up to 2—3 Hz) and high-frequency (up to 25000 Hz) ones. According to this, AMSs use low-frequency MTRD with Gamma 1101 code-impulse modulation and light beam oscillographs K 20-22, as well as high-frequency MTRD, 16CM3 equipment designed and built in LII (combined low-frequency and high-frequency magnetic tape recording) or 14CKM (a code-modulation system) to record information. In-flight recording is made on the by mode basis; the total one flight-recording time is 30—60 minutes.

Gamma 1101 records the low-frequency information discretely in a binary-decimal parallel code which is changed for each parameter at 8, 16 and 32 bits/second; this is caused by the record routine program chosen from 23 possible versions. The 16CM3 device records high-frequency parameters by the frequency modulation method, and low frequency ones-using a several code. The 14CKM equipment provides recording of the high-frequency parameters in a digital serial code on several parallel magnetic-tape record tracks. Part of the primary high-frequency measurement information is statistically processed on board the flying testbed using specialized devices forming the AMS. In this case the electrical signal is transformed into slowly varying ones which are recorded by the low-frequency MTRD. Fig. 1 shows a block-diagram of information acquisition performed by the airborne measuring system in powerplant and turbine engine flight testing on FTB.

Time synchronization, if the measurements recorded by several MTRD and LBO examples, is performed by all recorders of the impulse signals generated by one GAMMA 1101. The diagram provides for a delay of the first input impulse, that ensures simultaneous start of the impulse count by all the recording devices with different time of each tape deck spin-up. For convenient record processing, the time count in each mode begins at 0 that is provided by he MTRD automatic setting to null of the time impulses upon completion of each recording mode.

AMS on FTB is remotely controlled from a special panel. The operator using keys selects recorders which must operate in the subsequent mode and then switches on all earlier selected recorders using one key. This AMS control, having a by-mode recording capability, permits effective usage of the magnetic-tape and photo paper reserves available in the recorder cassettes. The recording time in one recorder switch mode is 15—60 sec, and the total recording time in one flight is 15—45 min. 15 minutes are a limited magnetic tape reserve in 16CM3, 14CKM MTRDs; 45 min-reserve is available in GAMMA 1101 MTRD. Application of the special control system allows automatic switching the second, standby, memory store after the tape forward and back wind by one GAMMA 1101 MTRD memory store, thus having doubled the time of in-flight measurement information up to 40 min. , that ensures obtaining the necessary information under unforeseen conditions.

To improve reliability of the measurement data acquisition, redundancy of the tape decks, i. e. simultaneous information recording by two memory stores, is provided. Of the same purpose are built-in devices for monitoring the measurement channels and AMS electrical circuits. The operator having received signals of the AMS operation in flight can take actions, if necessary, to eliminate faults or can correct the flying mission. The AMS monitoring is performed in the AMS laboratory and pre-flight preparations.

A feature of the FTB data acquisition system to test powerplants is its flexibility to changes in the information elements and distribution addresses in the recorders and their individual channels. The devices being used — cross-fields — permit of prompt variation in the electrical measuring channel directions during the preliminary preparation to flight, which, in its turn, provides rapid and high-quality preparation of AMS to solving the tasks which appear in testing.

During the laboratory preparation the measuring sensor (pickup) and transducer calibrations are made using a personal computer-based semi-automatic system. The calibration results are output in the form of polynomials of the 1—3 power with evaluation of the root-mean-square calibration data deviations from the recommended polynomials.

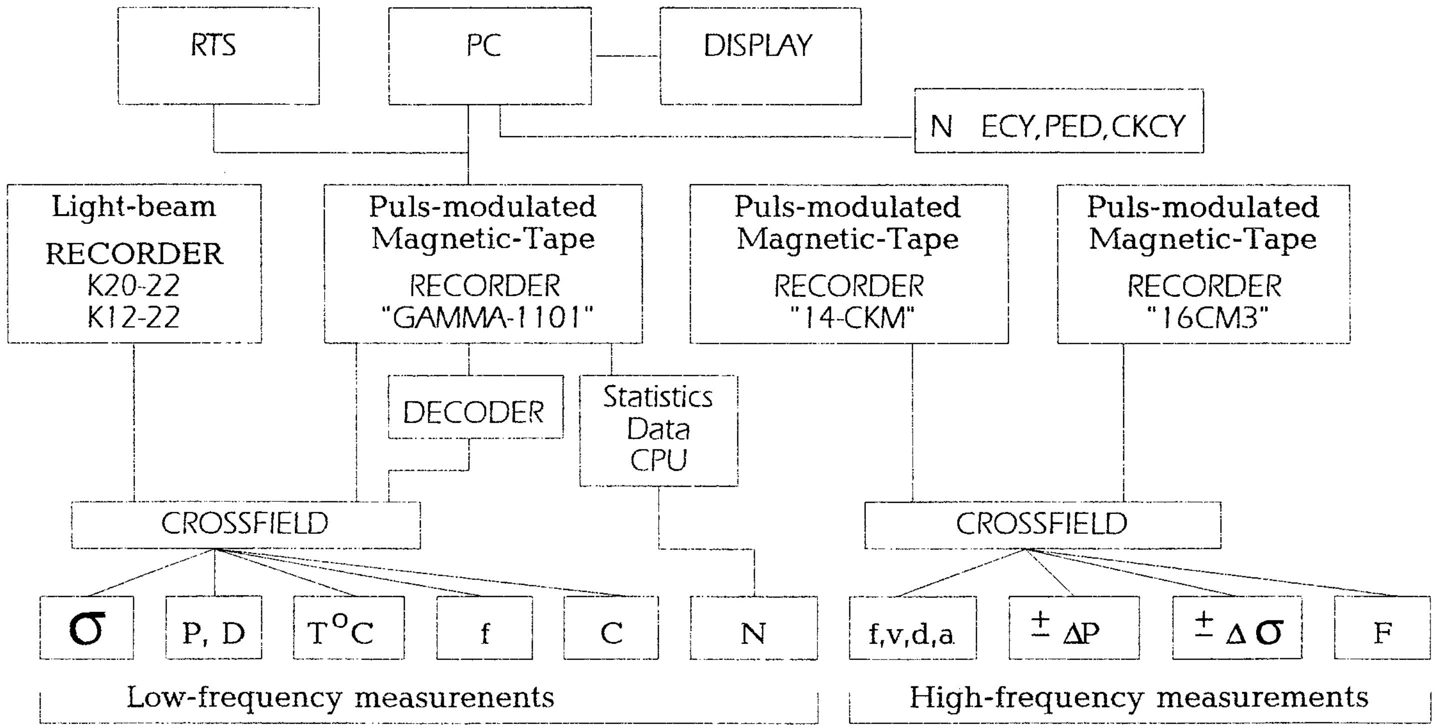

Figure 1. AMS Block-Diagram for Powerplant Flight Test on FTB

RTS - telemetric system
PC - personal computer
σ - mechanical stressness
P, D - pressure, displacement
$T^{o}C$ - temperature
f - fuel consumption, rotor speed, oth. friquently par.

C - discrete commands
N - digital code
f,v,d,a - vibration parameters
$\pm \Delta P$ - pressure fluctuation
$\pm \Delta\sigma$ - high-frequency mechanical stressness
F - pyrometer signals

FLYING TEST-BED STUDIES ON COMPUTERIZED SYSTEM FOR CONTROLLING FLIGHT EXPERIMENT IN GAS-TURBINE ENGINE TESTING

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At present computerized flight experiment control systems (CFECS) used in gas-turbine engine test-bed work are important means of improving the effectiveness and quality of flight tests. Main components of such systems as a man-machine complex functioning on the flying test-bed are information-measuring, computing and controlling systems. The paper deals with a computerized flight experiment control system as a variant of controlling systems, because this part of the computerized system is less investigated in gas-turbine engine flight tests, as opposed to the information-measuring system (IMS), where a considerable advance has been made, which resulted in creating onboard information-measuring systems based on mini-computers with onboard real-time data processing.

The flying test-bed is intended for carrying out flight experiments (FE) in the process of preliminary researches and refinement of gas-turbine engines and their systems. Flight experiment is a variant of scientific experiment and constitutes the most important link in the experimental understanding of an object. There are two kinds of ex- — active and passive. In the active FE the levels of factors ..ances) are specified by the leading test engineer in each test, while in the passive FE they are only recorded. On the basis of technical means applied to flight experiments the FS can be divided into four stages of its development (fig. 1). The second stage of the active FE development is a preparatory phase aimed at creating the CFECS. The second stage (25 years) provided mainly the development of methods for obtaining external and internal engine disturbances in flight in the form of test (exploratory) actions on the gas-turbine engine and its systems. Until the introduction of onboard computers the active experiment was most effectively applied in gas-turbine engine testing using a flying test-bed for solving such problems as altitude engine starting, evaluation of performance change at air bleeding and mechanical power take-off, achieving gas-turbine engine stall conditions, engine refinement in afterburner power conditions [1].

Long-standing foreign and domestic studies and development related to the introduction of computer technology into the sphere of testing made it possible to develop computerized systems for controlling gas-turbine engine bench tests and address the problem of creating such systems for gas-turbine engine testing on flying test-beds. As a result of these studies, the stages of the gas-turbine procedure most suitable for computerization were defined and the relative efficiency of the computerization of the most typical test procedure stages was estimated. These stages are:

— optimal adjustment of the engine performance in the process of its operation — efficiency 40%;
— measurement, recording and data processing — efficiency 33%;
— control of test conditions — efficiency 13%.

The availability of standard electronic systems for automated control (ACS) of gas-turbine engines typical for the third stage of the active flight experiment evolution as well as introduction of dedicated onboard computers into the practice of flight experiment made it possible to automate a wide spectrum of actions upon gas-turbine engines and their systems. Active actions upon the gas-turbine engine using the automated control system were accomplished as follows:

1. by simulating the signals of initial transducers or by controlled input signal distortion with retaining feedback;
2. by managing the operation of gas-turbine engine ACS on the digital channel of its interaction with the ground control panel.

The typical CFECS structure includes, as its primary element, a controlling computer (dedicated onboard or/and standard gas-turbine SAC with a digital electronic computer). In addition to the gas-turbine engine and input and output signal transducers, the CFECS incorporates also an information-signal unit; a unit of gas-turbine engine protection against impermissible regimes; assigning, controlling and executive elements (can be integrated with the standard automated control systems or installed additionally as separate transducers and units). The system can be used for gas-turbine engine testing on flying test-beds both with electronic and hydromechanical automated control system, on which the configuration of standard and additional units depend.

In active regime the CFECS makes it possible:

— to change the control loops tuning;
— to carry out sinusoidal time variation in tuning high-pressure rotor speed and minimal fuel consumption loops;
— provide stimulating signal for simulating the surge, changeover to reserve program for controlling high-pressure compressor variable inlet guide vanes, test reduction of tunings the loops limiting the ultimate engine operation parameters.

The first method of action upon the gas- turbine engine was implemented in flight testing the CFECS on the flying test-bed

Tu-16 N401, where the signals of the standard transducer A-97 measuring for the electronic ACS the lowering air temperature at the engine inlet were simulated using a resistance box.

By introducing a controlled distortion into the electronic controller, the turbocompressor throttling against the rotor speed at different turbojet afterburner power, including afterburner full on was obtained.

This technique made it possible to determine the boundaries of the afterburner start and stable afterburner combustion.

The second way of generating the actions upon the turbine engine ACS was investigated in refining the CFECS on the test-bed IL- 76 N3908 using PC of IBM PC/AT type.

The possibility for remote control of the electronic ACS through the channel of digital data exchange has been provided by the electronic controller developers (fig. 2). Specific results were obtained during flight testing of the PS-96A engine with application of the CFECS in the active regime at estimating the influence of the high-pressure compressor variable inlet guide vanes on the main engine operation parameters and available gasdynamic margins of the compressor as well as in determining aerodynamic characteristics of this engine.

The generation of actions at testing the effect of the high pressure compressor variable inlet guide vanes (HPCVJGV) deviation from nominal control programs (±8 deg.) was accomplished in flight using the PC keyboard. In this case the codes of the regulating element R3 of the controller RED-90 were varied in the ranges of-400,-200, 0, +200, +400 units. The investigations were carried out at altitudes of 7000 and 9000 m at $V_{ind} = 400$ km/h in stable conditions of the engine operation, corresponding to a reduced rotor speed of 81.9% and 90.4%. As ... ...e, the influence of the HPCVIGV deflection is shown in ...y controlling the variation of the regulating element R3 codes ...nin —100, 0, +100 units (±3 deg.) after keeping the nominal regime for 3 min. in the above flight conditions, the effect of the HPCVICV deviation from nominal values on the compressor gasdynamic stability at throttling back from nominal to idle was assessed. As a result of the investigations it was revealed that there existed reserves for increasing the gasdynamic stability of the compressor due to the HPCVIGV angle optimization, namely, by correcting the program of the HPCVICV control through turning the regulating element R3 of the controller RED-90 by —100 units of the code, which provides "closing down" the IGV by −3 deg.

The dynamic characteristics were defined by assigning with the CFECS the sinusoidal time tuning variations of control loops for the high-pressure rotor speed in engine conditions from idle to nominal power, These characteristics were evaluated in terms of coefficients representing the influence of such factors as fuel consumption excess, HPCVIGV deflections and reduced rotor speed versus throttling characteristics on the high-pressure rotor acceleration. The flight conditions and parameters obtained in stable conditions and oscillatory regimes are shown in Table 1.

№	P_m/t_m	Parameters of stable conditions						Parameters of oscillatory regime					
1.	0.777/5.5	3.8	70.8	762	33.7	−41.9	0.2	5.2	0.63	0.74	0.45	0.5	0.8
2.	0.777/5.5						0.5	4.3	0.32	0.18	0.25	0.3	0.6
3.	0.777/5.5	31.7	84.8	2490	65.7	−19.4	0.2	8.0	0.91	2.13	1.48	1.45	1.7
4.	0.777/5.5						0.5	8.0	0.77	0.84	1.05	1.5	3.0
5.	0.777/5.5	59.7	93.4	5670	88.8	−4.4	0.2	9.1	0.96	2.14	2.0	1.2	2.0
6.	0.777/5.5						0.5	4.4	0.48	0.62	1.15	1.3	2.2
7.	0.777/5.5	59.7	97.2	7120	95.7	-0.04	0.15	4.35	0.72	1.15	0	0	0

An analysis of the influence coefficients for high-pressure rotor at different values of reduced high-pressure rotor speed from 70.8% to 97.3% has shown the following:

— the influence coefficient for fuel consumption excess increases monotonously from 0.11 to 0.25;

— the influence coefficient of HPCVIJV deflections varies within —0.4%...—0.7%, with maximum value equal to —0.7%/s/deg being noted at 85%;

— the influence coefficient of the reduced low-pressure rotor deviations changed from low positive values of —0.1%/s/deg to negative ones of —1.2%/s/deg.

Further development of the CFECS in test-bed gas-turbine engine testing is planned to be accomplished in the following directions:

— introduction of on board small-size (portable) PC of a Notebook Type;

— generation of actions using information;

— application of a new method of actions in the form of output signal simulation (electrical, hydraulic, pneumatic and others);

— development of special electronic devices enabling similar test bench and flight procedures of controlling the active experiment.

Use of dedicated panels produced by the firms-developers of electronic controller had a profound impact in computerization of assigning exploratory actions in flight by variations of control channels tuning, shifting totally or separate parts of control programs, assigning stimulating surge signals. They made it possible to provide the refinement and introduction of first domestic electronic gas-turbine engine control systems including systems of overheat and surge protection.

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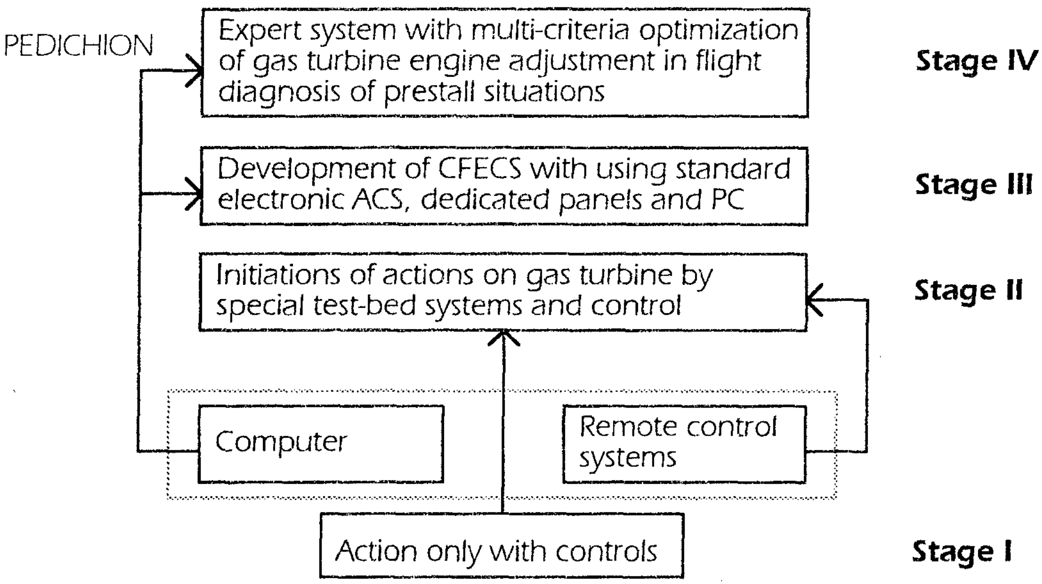

Figure 1.

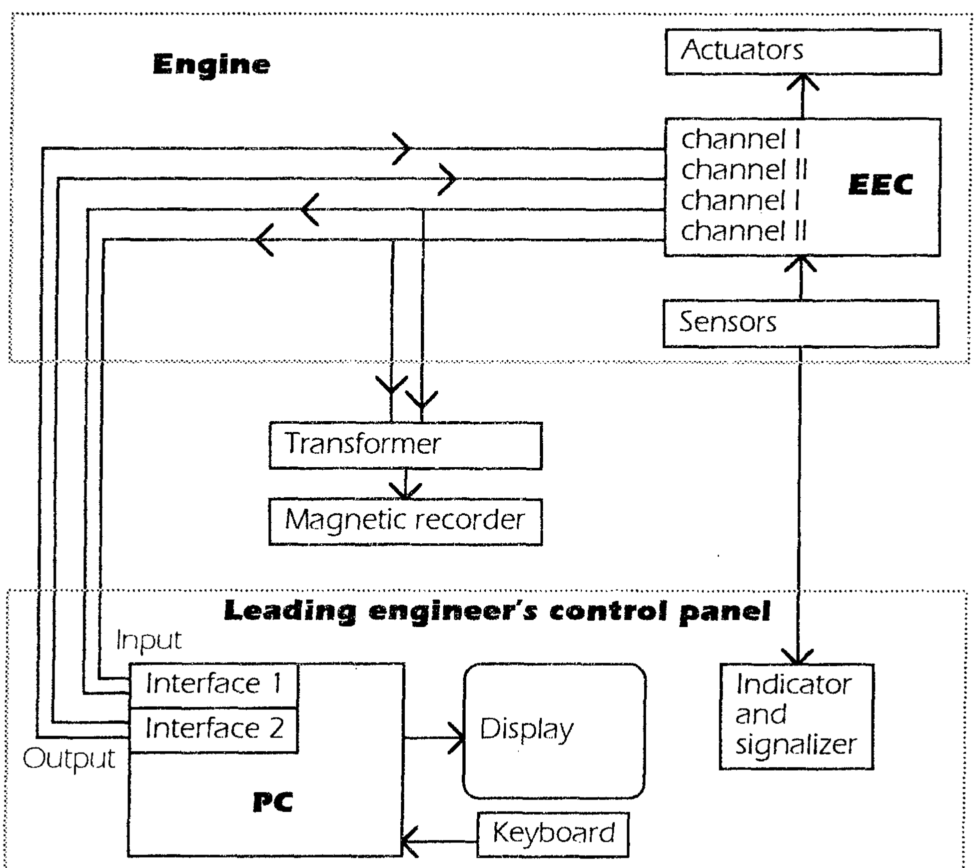

Figure 2.

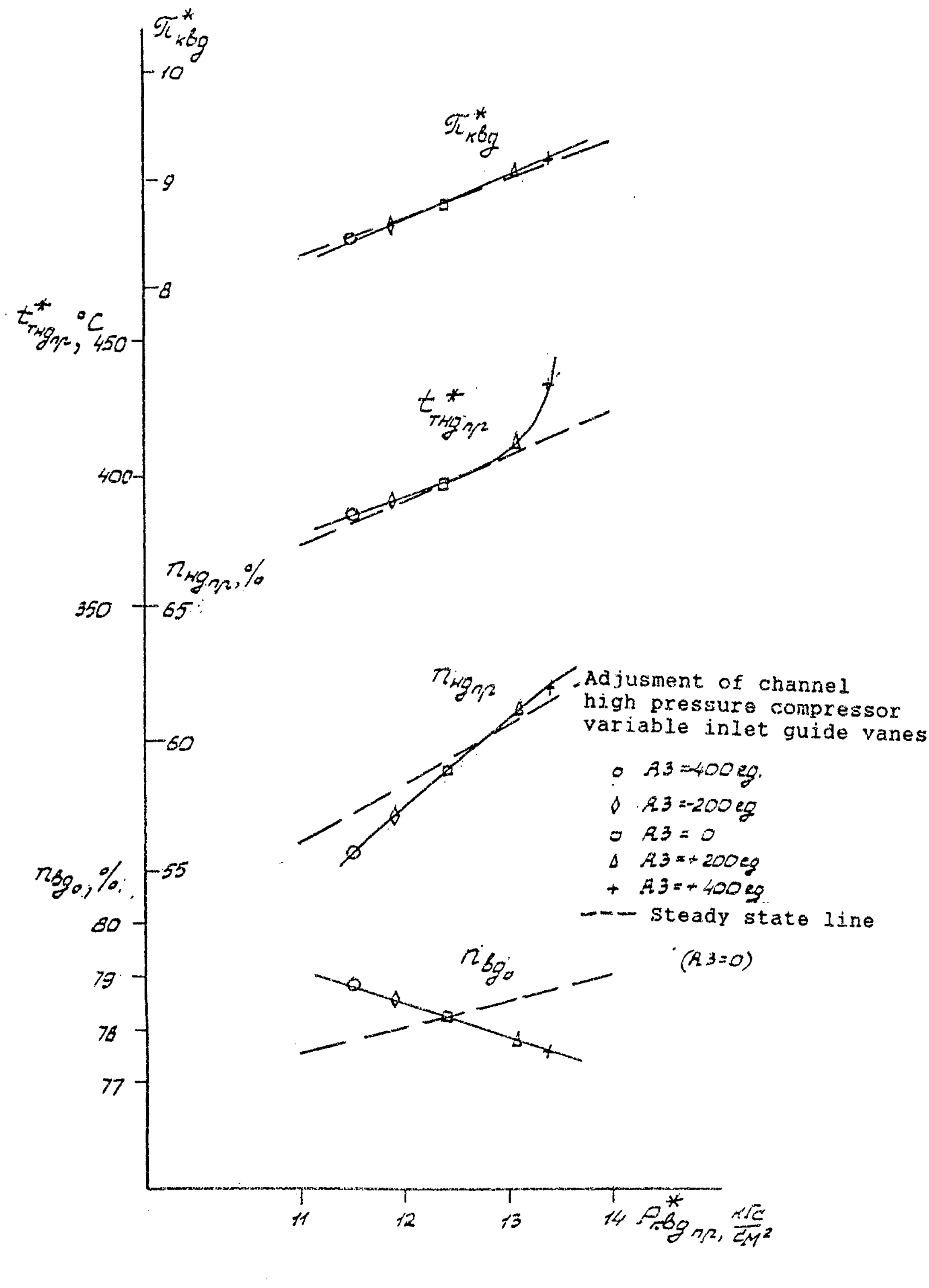
Adjusment of channel
high pressure compressor
variable inlet guide vanes
o A3 =-400 eg.
◊ A3 =-200 eg
□ A3 = 0
Δ A3 = +200 eg
+ A3 = +400 eg
--- Steady state line
(A3=0)

Figure 3

METHODS AND RESULTS OF ESTIMATING THE MAIN NEW-GENERATION TURBOPROPFAN ENGINE PERFORMANCES USING A FLYING TESTBED

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A universal flying testbed equipped with improved data acquisition processing and analysis system is an efficient and cost-effective means of estimating the main data and altitude-speed performances of an engine having a large diameter propeller or a fan since a full simulation of flight conditions on an altitude bench is either impossible or too expensive.

To solve the problem of a comprehensive evaluation of flight conditions effects upon the serviceability parameters and performances of the experimental turbopropfan engine built by the ZMKB "Progress", the IL-76-based flying testbed has been used.

The methodology of the onboard measurement analysis is based on the experiment design theory and consists of recommendations on estimating the thrust-versus-rpm curves under fixed environmental condition parameters, forms of the performances presentation, as well as mathematical support to identify and metrologically prove the engine empirical-mathematical models.

Each of the performance (power, consumption, thrust, gas passage parameters) is described by the following equation:

$$Y(x) = S(\overline{X}, Q) + e,$$

where:

$\overline{X} = (x_1, x_2, x_3, ..., x_K)^T$ is parameters-factor vector;

$Q = (Q_0, Q_1, ..., Q_p)$ is unknown parameters vector to be estimated;

S is the specified continuously differentiated function of $\overline{X}$ and Q;

e is the random error of Y response measurement (estimation);

K is the number of factors considered;

p is the number of regressions (basic functions) incorporated in the equation.

The performances are presented in the form of relationships between the inlet-condition-reduced parameters and the overall engine air compression ratio $P_{IS}=P^*_{\text{high pressure compressor}}/P_{amb}$.

Relationship $Y_i = f_{1i}(P_{IS})$ permit elimination in the first approximation of the Mach number effect upon the operating process parameters.

The relationship between the inlet-condition-reduced parameters and P_{IS}, flight Mach number — Mach, ambient temperature and pressure — T_{amb}, P_{amb}: $Y_i = f_{2i}$ (P_{IS}, Mach, T_{amb}, P_{amb}) are determined when variance of the parameters about the average values are greater than the similar variance estimated during ground runups. The values of the reduced parameter variance and their confidence intervals for the identified performance are also estimated from these data.

The operational hypotheses for the causes which resulted in the increase of the performance parameter variance in flight could be velocity variation in ambient air flow blowing the powerplant (engine passage geometry variation, nozzle section flow interference); changes in ambient temperature and pressure (engine passage Re number variation, changes in the fuel combustion completeness).

The expert assessment is performed using the curve plats to qualitatively verify consistency with the data obtained in bench testing and to reveal significant point deviations with the secondary factors changes.

The bench test data for the experimental turbopropfan engine have satisfactory similarity to the IL-76 FTB test data for ground conditions in the fuel consumption, low pressure turbine exhaust gas temperature, high pressure compressor air temperature, thrust, and propeller power.

The analysis of the flight test data has shown that the engine performances are polyfactorial and the application of the overall air compression ratio, both

$P_I=P^*_{\text{high pressure compressor}}/P^*_{\text{inlet}}$ and

$P_{IS}=P^*_{\text{high pressure compressor}}/P_{amb}$ as a single ordinate is insufficient because the parameters are greatly affected by the flight Mach number, ambient temperature, T_{amb}, and pressure, P_{amb}.

The experimental turbopropfan flight tests performed on the IL-76 FTB within the ranges of altitudes $H=0$—11000 m, stagnated ambient air flow temperatures

$T_{amb}= +20...-40$,m Mach numbers $M=0$—0.72

have permitted development of empirical-mathematical models of the thrust-versus-rpm and altitude-speed curves in the following form:

— fuel consumption

$$GtTO = F\,(P_{IS}, M^2)\ ,$$

— low pressure compressor rotor speed

$NHDO = F\,(P_{IS}, \mathrm{M}, P_{IS}\cdot\mathrm{M}, \mathrm{M}^2)$,

— high pressure compressor rotor speed

$NHDO = F\,(P_{IS}, \mathrm{M}, P_{IS}\cdot\mathrm{M}, \mathrm{M}\cdot T_{amb}, P_{IS}\cdot\mathrm{M})$,

— high pressure compressor delivery exhaust temperature

$T20 = F\,(P_{IS}, \mathrm{M}, P_{amb}\cdot\mathrm{M}, \mathrm{M}\cdot T_{amb}, P_{IS}\cdot\mathrm{M})$,

— low pressure turbine discharge temperature

$TTDO = F\,(P_{IS}, \mathrm{M})$,

— low pressure turbine discharge pressure

$PTHO = F\,(P_{IS}, \mathrm{M}, P_{amb}\cdot\mathrm{M}, \mathrm{M}\cdot T_{amb}, P_{IS}\cdot\mathrm{M}, \mathrm{M}^2)$,

— free turbine discharge temperature

$TCTO = F\,(P_{IS}, \mathrm{M}, (P_{IS})^2, \mathrm{M}\cdot T_{amb}, \mathrm{M}^2)$,

— free turbine discharge pressure

$PCTO = F\,(P_{IS}, \mathrm{M}, (P_{IS})^2, \mathrm{M}\cdot T_{amb}, \mathrm{M}^2)$,

— propeller thrust

$RBBO = F\,(P_{IS}, \mathrm{M}, \mathrm{M}\cdot T_{amb}, P_{amb}\cdot\mathrm{M}, (P_{IS}\cdot\mathrm{M})^2, P_{IS}\cdot\mathrm{M}, \mathrm{M}^2)$,

— free turbine power $P_{IS}\cdot\mathrm{M}$

$NNBO = F\,(P_{IS}, \mathrm{M}, P_{IS}\cdot\mathrm{M}, P_{amb}\cdot\mathrm{M}, \mathrm{M}\cdot T_{amb})$.

The errors of the models obtained are estimated through the calculations of the confidence intervals at an arbitrary point of the experiment plan from the formula:

$$\hat{Y} = \pm S_s \cdot d \cdot t(d_f, \alpha),$$

where $\hat{Y}$ is a tube of the response variance for the polyfactorial model within which, at a given probability, there will be a "curve" (the response mathematical expectations);

$$S_s = \sqrt{S_S / d_f},$$

$d_f = n - p - 1$, the number of the degrees of freedom;

S_S is the residual square sum;

n is the number of the basic functions;

$d = \sqrt{\overline{X}\cdot G\cdot \overline{X}}$ — the error corridor (let us take an average value for all n points of the experiment plan);

$\overline{X}$ is the vector of the basic function values;

$G = (X^T \cdot W \cdot X)^{-1}$ is the variance matrix;

W is the observation weight;

$\overline{X}$ is the experiment plan matrix;

$t\,(d_f, \alpha)$ is the Student criterion;

$\alpha = 0.95$ is the confidence probability.

The estimation results have shown that the error corridors, d, with the model identifications over the entire flight envelope and operation models are in the limits of 0.1 to 0.25.

Operating with the actual numbers shows that the confidence intervals at the top and at the bottom of the average value are 19 kg/h for the fuel consumption; 0.4% for the low compressor speed; 0.4% for the high compressor speed; 9 degrees for the high pressure compressor exhaust discharge temperature; 8 degrees for the low pressure turbine discharge pressure; 8 degrees for the free turbine discharge temperature; 1% for the free turbine discharge pressure; 2.5% for the propeller thrust.

The empirical-mathematical models developed for the thrust-versus-rpm and altitude-speed curves of the experimental turboprop-fan engine are used in calculating the thermogasdynamic characteristics of the turbine-compressor elements, as well as in analyzing the data of the basic aircraft flight tests and design of a prototype for an automatic powerplant conditions system to monitor operation.

INTERFACING AND FAIL-SAFETY EVALUATION TECHNIQUE FOR MODERN CONTROL AND MONITORING SYSTEMS OF GAS-TURBINE POWER PLANTS DURING FLIGHT ON FLYING TEST-BED

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The characteristic feature of modern gas-turbine powerplants for civil and military aircraft is an extension of functions performed by digital automatic control systems (ACS) and on-board diagnostics and monitoring systems (ODMS). These systems interface closely between each others and, with stand by hydromechanical gas-turbine powerplants as well as with aircraft flight control systems, cockpit recording systems and on-board recording systems. In this case a complication of modern gas-turbine engine structures and function limitation of hydromechanical ACS systems lead to a significant increase in responsibility of powerplant electronic systems and in their influence on the fail-safety of powerplants and on aircraft in a whole. At the same time a wide use of electronic indication means as well as emergency and operation recorders on board an aircraft appears to arise some problems in their interfacing with powerplant digital systems. This is an insufficient validity of the powerplant operation and the fault data reported to a crew and recorded during a flight.

The tasks of digital ACS and ODMS systems of gas-turbine engines studies are being solved in complex on the main stages of development and implementation. At the same time a large volume of laboratory, bench and flight tests is being conducted. They cover:

— mathematical modeling of digital ACS and ODMS systems;
— preliminary studies using ACS and ODMS systems prototypes;
— laboratory tests of digital ACS and ODMS systems;
— tests on half-slack benches;
— tests on benches equipped with gas-turbine engines;
— flight tests of gas-turbine engines with digital ACS and ODMS systems using flying testbeds;
— flight tests on the stage of aircraft experimental operation.

The possibility to acquire a valid data on the affects of functional failures of the ACS and ODMS systems is very significant at every stage. The main difference between the possibility to use the technical means and hardware and to provide the appropriate investigation such as effect range of different factors and fault types which determine the operation of the ACS and ODMS systems. It is possible to extend these ranges at particular stages (especially at the flying testbed stage) by flight condition simulation, by simulation of aircraft system powerplants which interface with the ACS and ODMS systems, by the simulation of the ACS and ODMS systems faults.

As a rule the stages of flight tests on flying testbeds and aircraft prototypes are the final stages in the refinement of the digital ACS and ODMS systems. But a number of special tasks such as selection of the appropriate control and monitoring algorithms, the dynamic characteristic studies of the ACS and ODMS systems in all range of gas-turbine engine operating conditions, the check-out of the ACS and ODMS system BIT facilities to determine the completeness and thoroughness of their monitoring capabilities at the switches on the standby control circuits, the collection of logical operations in time cycloramas of the ACS and ODMS systems are used arise even at these stages during the check-out of the ACS and ODMS system functional characteristics. In this connection, despite a high level of the refinement of the digital control and monitoring systems at the bench test stages the following tasks are solved during the process of flight tests of the digital ACS and ODMS systems:

— to determine the compliance of the main functional characteristics of the ACS and ODMS systems with the design specification requirements;
— to evaluate the operating factors effects (vibrations, environmental temperature and pressure, acoustic pressure, electromagnetic fields) at the reliable operation of the ACS and ODMS systems and their functional characteristics;
— to evaluate the completeness and thoroughness of the monitoring capabilities of the ACS and ODMS system BIT equipment during simulation of single faults as well as faults in various combinations;
— to improve the valid reception, recording, analysis and documenting of the data from the digital ACS and ODMS systems.

At conducting flight tests on flying testbeds a special emphasis is made to the work concerning the determination of fault-safety of the powerplant digital systems, criticality level for the various types of faults in the powerplants systems and their interfacing with aircraft flight control systems, display, data and warning indication systems which provide flight safety of flying testbeds. Flight test technique for the determination of flight safety consists of artificial interference in the powerplant digital system operation and the systems interfacing them with the help of simulation of every possible fault and failure of interrelation between the interfacing aircraft systems. As to the powerplant digital electronic systems the following types of effects are considered:

— faults of sensors, actuators, enunciators (break, short circuit, power interrupt);

— input of artificial (uncontrolled) signals (another signal level, another frequency, alternate contact) to the sensor, actuator circuits;
— supply of high frequency interface of high-intensive fields to the on-board circuits of the powerplant digital systems;
— alternation of power voltage parameters of the powerplant electronic systems;
— false data input intothe interfacing systems;
— false data reception by the control, monitoring, display and anuation systems.

In the process of flight tests on flying testbeds all types of faults should be identified by flight safety attributes and criteria, in other words, all faults and fault combinations should be identified unambiguously. In cases when they are impossible to recognize a fault algorithms should be worked out to recognize them by indirect or complex parameters and attributes as to the criterial faults required an immediate crew reaction for their crimination they should be arrested in the digital ACS and ODMS systems, stated in the flight operational manual instructions and displayed on the control and display means. Other types of faults should be recognized by BIT equipment of control and monitoring systems and displayed by indication systems as an additional data. The purpose of the fail-safety test technique is to check-out any fault combination so that to prevent even hardly probable faults during operation.

As conducting the flight-safety tests of power-plant digital control and monitoring systems the technique improvement was supported by the implementation of IBM PC/AT and Notebook computers for recording purposes as well as for remote active control of the powerplant digital control and monitoring systems. On the base of IBM PC/AT computers LII named after M.M. Gromov has designed an integrated information control system (IICS) which was flight evaluated on flying testbed with engine prototype and its digital control and monitoring systems.

IICS system presented a combination of the hardware consisting of IBM PC/AT computer and installed interface modules and software working under the control of MS DOS 5.0 (6.0) operation system. The IICS system allowed a complete real-time input, recording, display, documenting and analyzing the data from the powerplant digital systems with simultaneous output of control effects to the powerplant digital systems. In the active operating mode the IICS system sent the simulating signals for the simulation of surge, switching to standby control programs and reduction of limitation programs on the marginal engine operating parameters.

Flight evaluation of the IICS serviceability on the flying testbed has illustrated its high effectiveness. The validity, operational flexibility and labour content of the flight experiment have been increased significantly. Work is now being done to expand the range of the IICS system possible applications. This work includes thorough developments for simulation of faults and operation of BIT equipment of the powerplant digital systems and some software branches of the powerplant electronic equipment.

On the base of PC/AT computers there were designed and implemented in flight tests on flying testbeds Chief Research Engineer Workstation for Studies of Powerplant Control and Monitoring Systems (PCMS CREW). The PCMS CREW software has provided:

— data input from the powerplant digital systems;
— input data monitoring;
— data compacting without loss of its content richness;
— data analysis;
— documenting graphs, tables and cycloramas;
— selective copying;
— editing data display files;.

The designed PCMS CREW workstations have showed high effectiveness in the ACS and ODMS data processing during the flight tests of gas-turbine engines with the flight test of gas-turbine engines with digital systems on the flying testbeds.

The results of flight tests and refinement of the digital ACS and ODMS systems have demonstrated that IBM PC/AT computers have wide possibilities and high efficiency of application on-board a flying testbed as a part of the IICS system as well as for computer-aided data processing on PCMS CREW workstation during conducting flight fault-safety tests to provide the required aircraft flight safety level.

CAUSES OF INCREASED VIBRATION ACTIVITIES OF MODERN GTES THEIR OPERATION ON AIRPLANES AND HELICOPTERS

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The `experience of GTE operation has shown that despite significant improvements in analytical methods of calculating engine dynamic properties, high quality of balance and advanced manufacturing technology, use of elastic and hydraulic dampers in rotor bearings, reduction in the vibration levels of the engine rotor casing and components to the limit permissible in operation is a great problem.

Fig. 1 is an illustration of the primary vibration effects typical of modern GTEs in flight. This is a thermal effect characterized by considerable intensification of the rotor vibration level at the initial stage of the structure warm-up (fig. 1j). The time interval at which the maximum rotor vibration level determined from the moment of the latest variation in the engine operational mode, would vary in the range from tens of seconds to tens of minutes (depending on the engine design features, its initial thermal state and environmental conditions).

Vibration characteristics of certain engine types are also responsive to the aerodynamic factor (fig. 1e). For example, ON-OFF of the turbine cooling in some types of engines results in discontinuous and sometimes impermissible increase of the rotor vibration level.

Fig. 1a shows the most popular feature of the modern GTE vibration characteristics in the airplane configuration — their response to inertia load effects at aircraft maneuvering.

The interdependence between the acoustical processes in the gas flow passage and the GTE vibration characteristics is detected at the combustion chamber vibratiory regimes, and some engines encounter problems of the case vibration at blade frequencies.

It should be noted that the acoustic mechanism is not the only one to excite vibrations at blade frequencies. For example, intensive vibrations at these frequencies are caused by the rotor brushing against the case.

With an aircraft moving along the runway or in flying in a turbulent atmosphere, a kinematic mechanism of the engine vibration (at frequencies of the airframe configuration) show up.

Peculiar GTE loading occurs in a helicopter powerplant configuration. In this case the loading level is associated with a complex combined action of static and dynamic forces of different nature and presence of regimes with the disengaged overriding clutch (fig.2) in flight due to improper engine and rotor matching.

A variety of vibration effects presented unambiguously indicates a modern GTE to be substantially a non-linear system and due to this fact its characteristics are equally determined by changes in the exciting forces and in the structure properties.

The impossibility to simulate all the factors defining the engine vibration activity with the test benches dictates the urgency of conducting GTE flight strength tests.

All the causes of GTE vibration effects considered for full-scale conditions act in a complex combination simultaneously. The results are determined by the probability laws and, hence, their specific display in the vibration condition of different engine specimens takes place.

Analysis shows that primary forces that "incorporate" non-linearity are of aerodynamic, inertial and thermal natures.

The aerodynamic non-uniformity of a corona, accident in assembling, results in the rotor aerodynamic disbalance.

The upper estimate of the latter on a single corona (stage) can be obtained from the following formula:

$$R_s = \frac{b \cdot R_h}{Sin(\pi/z)}$$

where

b—variation range of the stage aerodynamic force value;

R_h—nominal value of the aerodynamic force acting upon one blade;

z—number of the corona blades.

Calculations indicate that the level of these forces for some rotor stages in particular engine specimens can be of the order superior to that of the forces, generated by the permissible engine rotor disbalance.

The flow aerodynamic non-uniformity which parameters also adhere to the statistical distribution laws leads, under the condition of

$$\frac{z}{n \pm 1} = \text{integer},$$

where n — is the number of the circumferential non-uniformity harmonic, to the stage aerodynamic force to occur even with a uniform blade corona and it is stationary about the non-uniformity.

The maximum value of the force radial component can be found from the formula:

$$R_s = \frac{z \cdot b \cdot R_h}{1.41}$$

The value estimate of this force shows that its bearing components can be several times larger than the rotor weight forces for individual engine specimens.

The specific mechanism of the aerodynamic force influence upon the GTE vibration activity is related to the axial force action. Because of a very large value of the two force components applied to the compressor and turbine rotors, should be a several-millimeter non-axiality of the point of their application (random low). Forces, proportional to the rotor right arise in the rotor bearings

$$MaxR_{brg} = \frac{\max(R_{ax.k} \cdot ek + R_{ax.T} \cdot et)}{b_{brg}}$$

where

ek, et— eccentricity of the point of the axial force application for the compressor and turbine, respectively;

$R_{ax.k}$, $R_{ax.T}$— values of the axial forces for the compressor and turbine, respectively;

b_{brg}— distance between the rotor bearings,

Analysis shows that essential contribution to the aerodynamic force level is made by GTE turbine module.

Fig. 3 gives a diagram showing the complex action of the main rotor forces in its bearings for GTE operating in an aircraft configuration. It is seen that to obtain the maximum change in the non-linear bearing stiffness, both the maximum level of each force separately, and their in-phase application must be attained.

This suggests that the required reduction in the level of modern GTE vibration activity can be reached with only a combination of the dynamic balance and a reduction in the level of the aerodynamic-nature forces fixed about the engine case (housing) and applied to the rotor.

The thermal disbalance phenomenon is of special nature. It is found from the experimental and theoretical research that this type of disbalance shows up in the vibration characteristics as follows:

— by changes in the rotor deflection which lead to significant increase in the rotor vibration level with subsequent discontinuous or smooth vibration reduction in the process of increasing the rotor speed or with time, after reaching the steady-state mode; with a considerable deflection, there may be a vibration activity increase that cannot be eliminated during the entire engine operation to its shutdown i. e. thermal disbalance of the first type (the primary indication is a high vibration level during startup, with the vibration level depending on the rotor bearing);

— by means of mutual displacement of the rotor bearings, which realizes through a force action of the rotor on the stator via the non-linear support a transition to another portion of the non-linear support stiffness characteristic-thermal disbalance of the second type (the primary indication is an increase with further vibration level reduction, with the engine operating at a steady-state mode).

As for the mechanism of the inertia force effects upon vibration activity. The extent to which GTE vibration activity varies under inertia forces depends on the engine onboard arrangement (see fig.2).

The problem of the modern GTE increased vibration activities is related to the task of providing in-service vibration monitoring efficiency.

To provide the required completeness level of developing flaws detection by vibration monitoring systems it is necessary to choose from all the changes in vibration arising under effects of internal and external factors, the changes which areindividual and random for various engine specimens within the type (fig. 4) the indications unambiguously related to the flaw (failure) development.

It should be noted that current vibration monitoring systems use two of three vibration parameters containing diagnostic information. Theoretical analysis shows that significant improvement of GTE onboard vibration monitoring efficiency is provided by the use of vibration phase angles.

The possibility to employ the parameter appears only within an individual approach to the vibration monitoring.

Conclusions

1. To reduce vibration activity of modern aviation engines having non-linear rotor support under operational conditions it is necessary to provide, besides a high-quality dynamic balance, the compensation for the static force actions, applied to the rotor and stationary about the case.

2. To provide reliable in-service identification of aviation engine failures by a vibration signal monitoring, individual methods of vibration monitoring are necessary, which use all the three vibration parameters of each vibration spectrum diagnostic component for vibration diagnostics-level, frequency and phase angle.

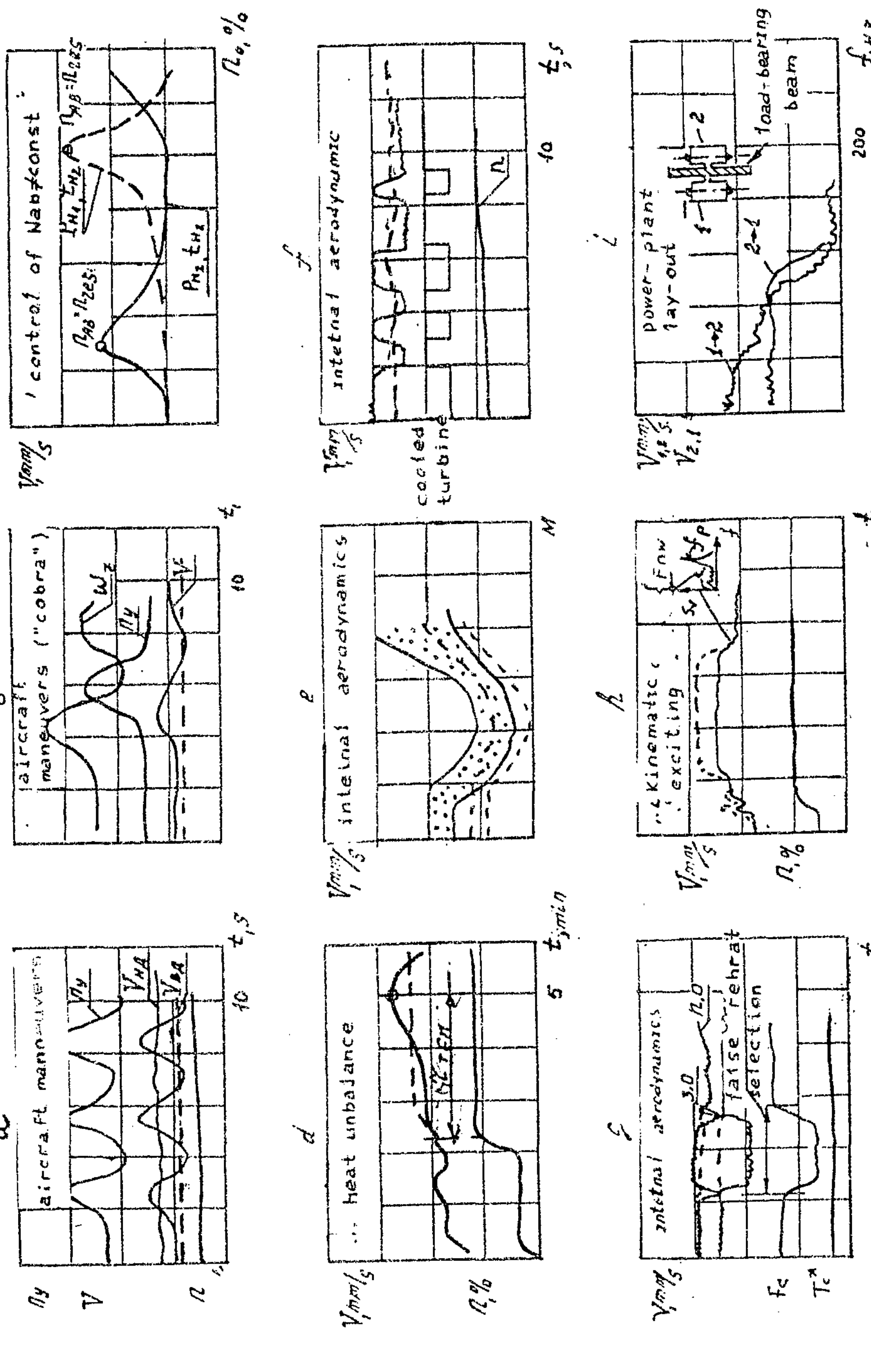

Fig.1 Vibration activity features of gas turbine engine in airframe configuration.

------ } different engine examples
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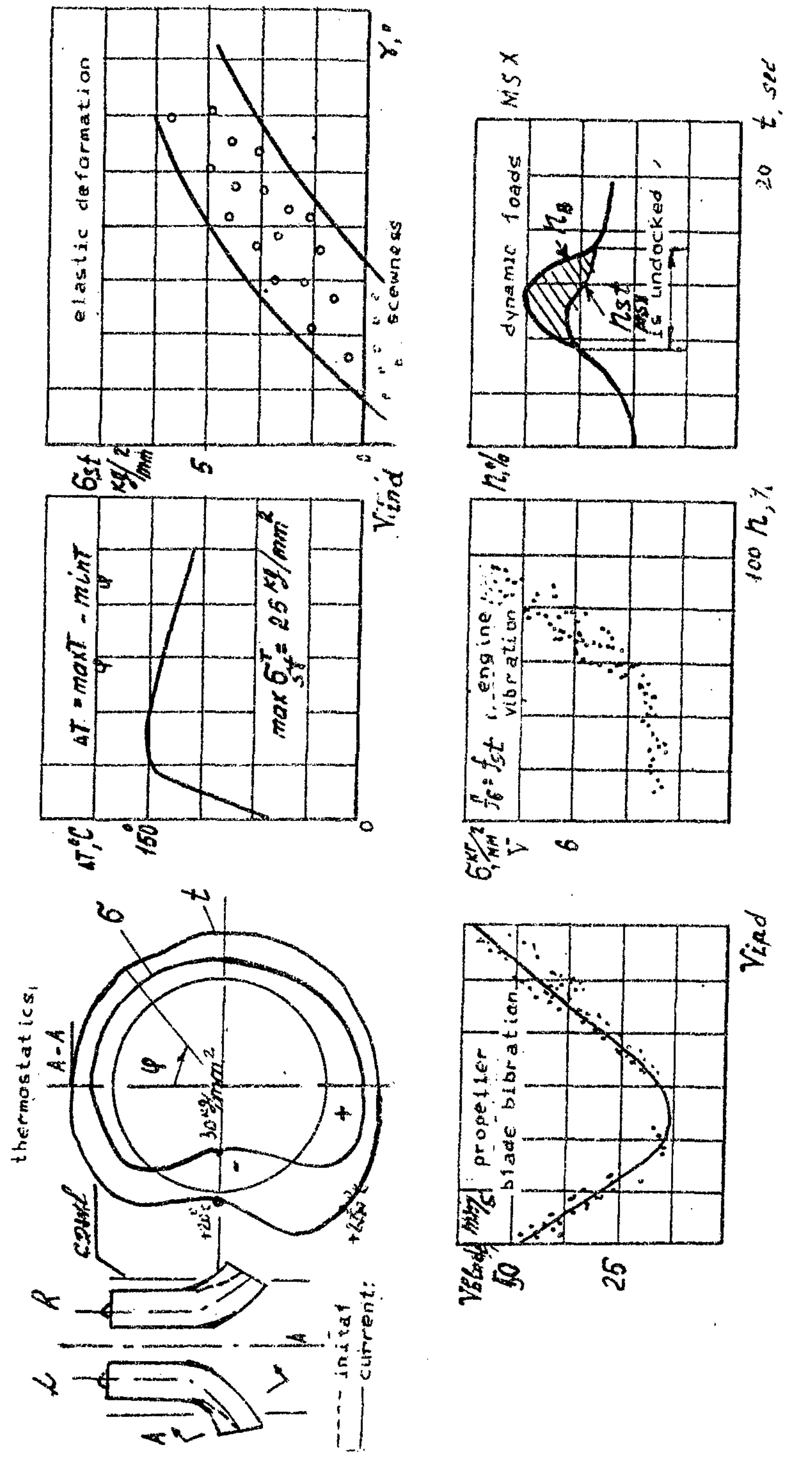

Fig.2 Summary of combined static dynamic loading peculiarity of helicopter power-plant stator.

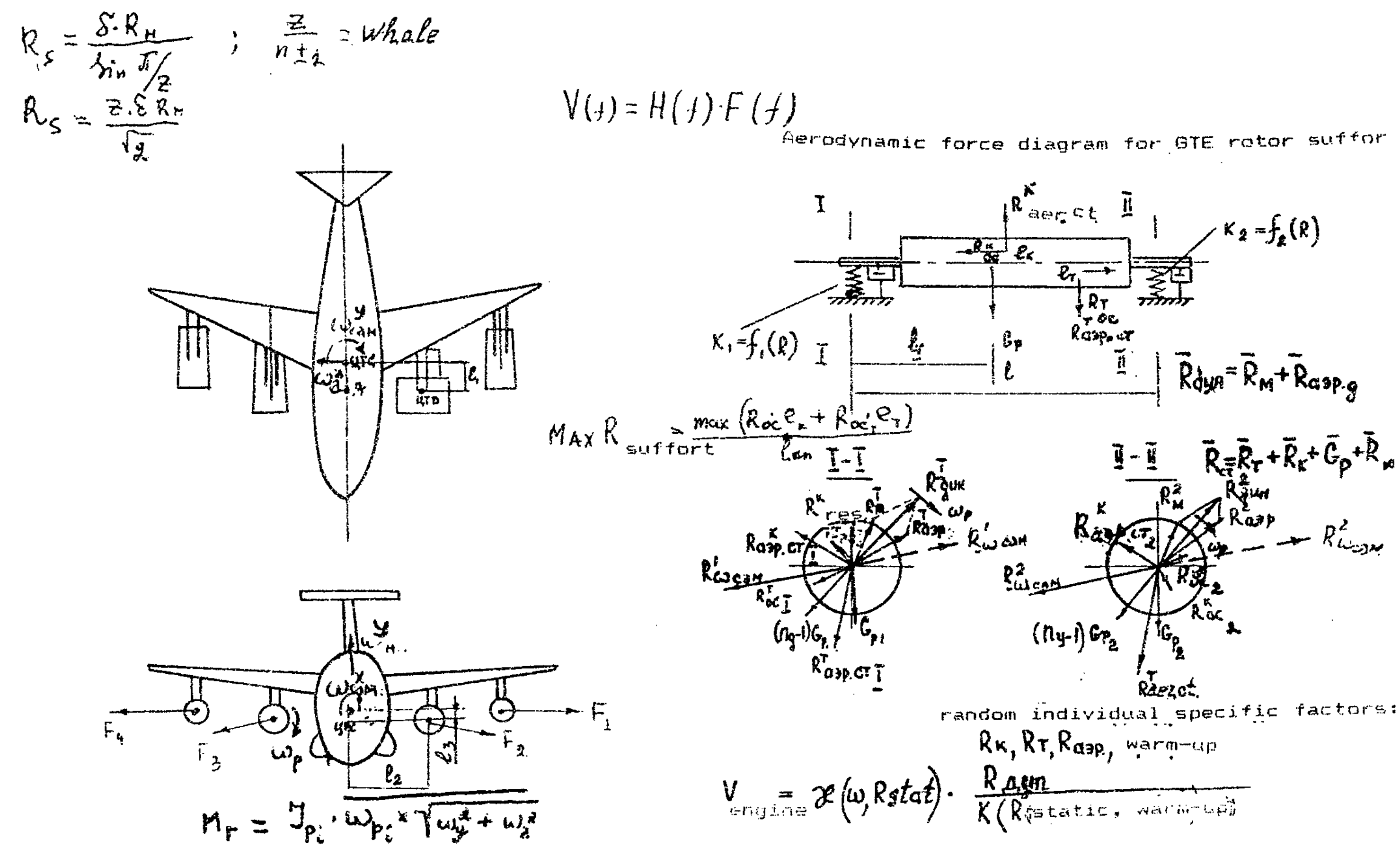

Fig.3 On understanding the individual (specific) GTE vibration characteristics having, non-linear rotor supports in an aircraft configuration.

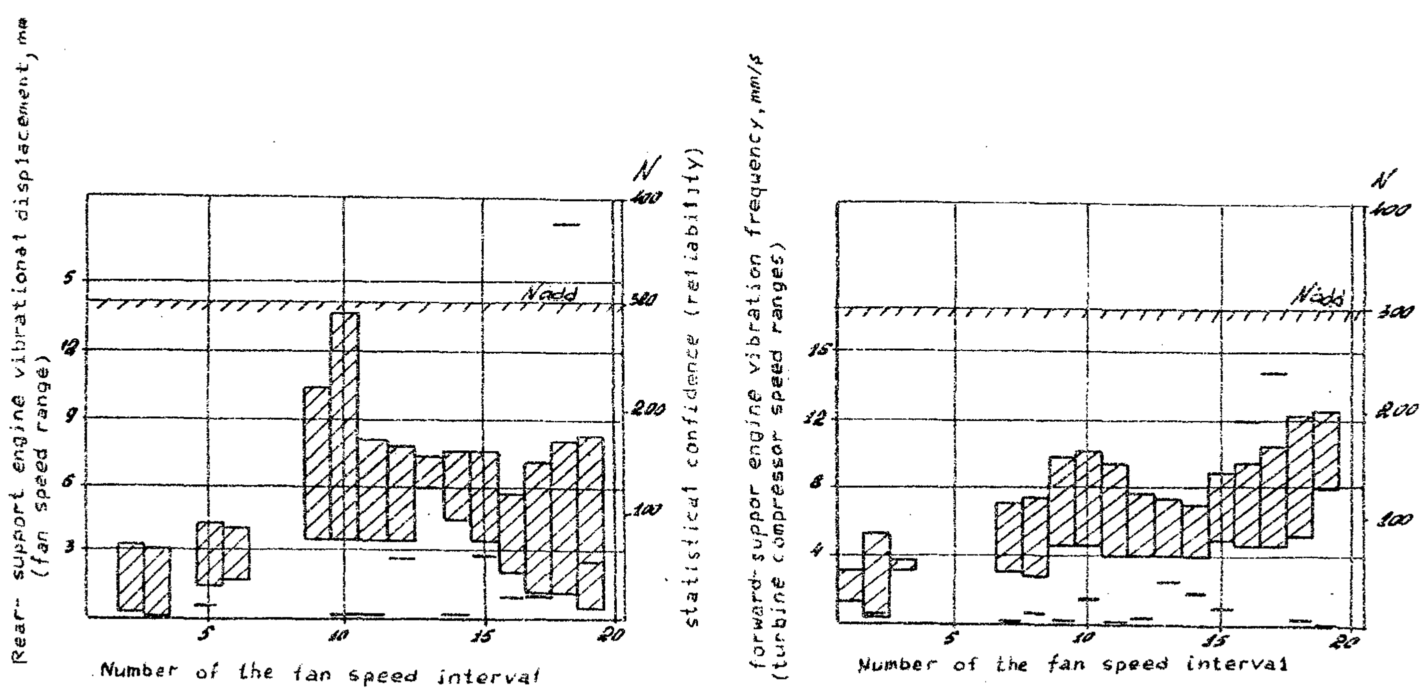

Fig 4 Randes of possible turbinefan engine vibration level variations according to the standard monitoring equipment at a stage of the engine individual vibration model generation during flight servl life tests.
(harizontal dashes define the level of statistical support for vibration variation ranges).

TECHNOLOGY AND PROCEDURE OF PISTON-ENGINE POWERPLANT TESTING TO LAUNCH FLIGHT TESTS AND CERTIFICATION OF GENERAL-PURPOSE AIRPLANES

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In the last few years general-purpose piston-engined airplanes weighing 2—3 t have been developed. Easy in control and economically efficient in operation, requiring no sophisticated ground facility and paved runways, these airplanes are widely used in our country in many branches of national economy and by separate individuals. The competitive demonstration flights carried out in Saint-Petersburg (1993 and 1994) and in Moscow (June 1995) have shown that a lot of amateurs and established design bureaus, previously engaged in developing civil and combat aircraft powered by gas-turbine engines, are involved in creating general-purpose vehicles. Specifications for developing reliable vehicles meeting the international standards are available. These are the Airworthiness Regulations АП-23 for airplanes with weight over 750 kg, IAR-VLA and IAR-22 for airplanes weighing no more than 750 kg (at present in Russia Regulations are being developed on the basis of IAR-VLA) and Regulations АП-33, in accordance with which aircraft engines, including piston, are developed.

When designing the airplane powerplant, the aircraft designers run into difficulties in selecting the engine, propeller, instruments and equipment of home production, since the aviation industry of the past years have been almost fully oriented to creating large turbine-engined airplanes. Aircraft piston engines were produced in a very limited quantity, their power was more than 300 hp. As an example, we may refer to the M-14 engine and its derivatives with power of 360...400 hp. It is a reliable and efficient engine successfully used for sport, transport, agricultural and ambulance airplanes. For many years it is being produced in series and now it has received the type certificate. However, at present the home industry does not satisfy the demand for piston engines with thrust of 200-250 hp at most for two/three-seat light airplanes. The aviation engines of this class with the type certificate are not produced in our country. Therefore the aircraft developers use piston engines of foreign firms such as Rotax (Austria), Teledyne, Zycoming (USA) and rotor-piston engines of the MWAE (Great Britain) having the type certificate, sufficiently reliable and efficient, although domestic design bureaus have developed and tested light and efficient piston, rotor — piston, diesel engines. However because of financial problems these engines are being produced in small quantities or not produced at all and have not been subject to type certification testing.

Engine instruments are not being developed and produced too. As a rule, aircraft developers order foreign certificated engine instruments and control devices to install them on foreign engines.

Usually wooden or plastic two/four blade fixed-pitch propellers are used for light general-purpose airplanes. The development and production of such engines have been mastered by our manufacturers. But these engines were not subject to certification tests and require special flight testing for 50 hours of running time to be installed on an airplane developed in accordance with the IAR-22

If the airplane is developed according to the АП-23 Regulations (with weigh above 750 kg), which require that the engine and propeller should have the type certificate, the airplane developers furnish foreign engines with propellers of foreign companies (Hurtzell, Muhbauer and other) having the type certificate.

Before starting flight tests a set of ground functional checks is performed to assess the operation of the powerplant, engine and propeller.

These procedures are fully in conformity with the IAR-VLA, IAR-22 and АП-23 Regulations and involve:

1. Assessment of the access to the engine, its units, powerplant units for performing inspection, maintenance and adjustment (Regulations points 901, 971, 973, 977, 999, 1125).
2. Assessment of the installation and mounting of joints, units, pipelines to provide normal operation of the engine, propeller and powerplant systems (points 925, 967, 975, 993, 1013, 1023, 1103, 1123, 1125, 1145).
3. Assessment of the powerplant fire safety, protection of the cockpit and passenger cabin against exhaust gases and fuel vapours (points 853, 975, 995, 1121, 1125, 1141).
4. Check of the engine, propeller, supercharger operation in steady-state and transient conditions from idle to take off thrust. Measurement of the propeller thrust (points 33, 903, 905, 909, 1041, 1301, 22-1801, 22-1807, 22-1919, 22-1933, 22-1945).
5. Check of fuel, oil systems and systems of air supply and exhaust gas bleed. Determination of the trapped unusable fuel (points 951, 955, 961, 1011, 1091, 1125, 22-1835, 22-1839)
6. Assessment of the possibility to control and set the required mode of engine and powerplant units operation using levers, switches and airborne instrumentation (points 995, 1013, 1114, 1143, 1147, 1305, 1337).

7. Check of vent systems operation (points 967, 975, , 1103, 1193).
8. Check of fuel filters operation (point 977)
9. Check of the system for cooling the engine and its units.(point 1041).
10. Check of the operation of draining devices and breathing systems (points 969, 071, 999, 1017).
11. Check of the powerplant units operation: generator, starter, magneto, relay, booster coil and others (point 1301).

In flight tests the following procedures meeting the airworthiness requirements IAR-VLA, IAR-22 and AП-23 are performed:

1. Functional check of the engine, propeller and discharger in the following conditions:
   — engines ground run-up before the flight from idle to takeoff thrust in steady-state and transient conditions
   — airplane takeoffs with the engine running in takeoff and climb conditions
   — in level flights over the whole range of altitudes and speeds with the engine running from idle to takeoff conditions
   — at engine acceleration and throttling at smooth and fast thrust lever movements from idle to takeoff conditions at negative and positive vertical and lateral g-loads; (points 33, 901, 905, 909, 943, 1047, 1091, 1521, 22-1807, 22-1808).
2. Test of the propeller at the airplane acceleration in level flight and descent with marginal flight speeds and the engine running from idle to takeoff thrust (points 33, 22-1905). The shaft speed should not exceed the maximum allowable value.
3. Check of the engine in flight start over the operational range of altitudes and speeds of the airplane flight. To define the dependence of the wind-milling speed, the pressure of fuel and oil at the speed of the engine-off flight, the airplane decelerations down to a minimal allowable speed are carried out.
   To verify the engine reliability in restarting, starts of the cold (after extended glide) and hot (in 5—15 s after stopping) engine are made. The start with one ignition loop shut-down is tested. The starts are carried out in the terminal area at a distance ensuring the airplane landing, if the engine fails to start (points 903, 22-1835).
4. Functional Check of the full and minimum loading over the entire operational range of altitudes and speeds is performed in level flight with the engine running in steady-state and transient conditions, in flying the horizontal and vertical maneuvers with pitch angles ±30 and bank angles ±60 and negative and positive vertical and lateral g-loads.

The functional engine check is carried out on the ground and in flight, when determining the trapped unusable fuel before there occurs engine malfunction, that is, before appearing the first signs of its unstable operation, partial or complete power loss, fuel pressure decrease below a minimum allowable value, fuel pressure pulsation.
The altitude tolerance of the fuel system is checked in climb flights to the maximum altitude at the temperature in fuel tanks of +40—45°C. Simultaneously checked are the systems of vent and breathing, absence of siphon leakage, operation of fuel filters (points 943, 951, 955, 959, 961, 967, 977, 1011, 1017, 22-1835, 22-1839).

5. Check of the system for cooling the engine and its units over the entire range of flight altitudes and speeds, in climbs to the ceiling and in descents (points 1041, 1047, 22-1821).
6. Check of the possibility to set a required mode of the engine and powerplant units operation with the aid of the cockpit controls and to monitor the operation of the engine, fuel and oil systems using airborne instrumentation (points 995, 1141, 1143, 1337).
7. Check of functioning and strength of joints, units, powerplant fillings in the range of altitudes and speeds of the airplane operation (fuel and oil tanks, oil cooler, pipelines, discharger air-supply passage, exhaust system and equipment (points 909, 963, 965, 967, 973, 993, 995, 1013, 1023, 1103, 1123, 1125, 1141, 1193, 22-1833, 22-1919, 22-1933, 22-1939, 22-1945).
8. Assessment of the protection of the pilots and passengers against exhaust gases and fuel vapours (points 1121 1125).
9. Check of the powerplant units operation (generator, starter, magneto, relay, regulator, booster coil and others) (points 13011, 1309).

In general, the flights related to the powerplant testing are carried out in the scope of test flights for determining main aircraft characteristics.

DEVELOPMENT TENDS OF THE LII AS THE CENTER OF FUTURE AEROSPACE INSTRUMENTATION TEST

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ABSTRACT

The search for a means of the synthesis of aviation and space engineering for obtaining some new features for them is being conducted now. One of the mutually beneficial options of obtaining these features is forming the common information and radio communication areas which will meet the operational requirements of both aviation and space engineering.

The present paper is devoted to solving this very important and urgent problem in view of building an unique radio communication test range for conducting some evaluation and certification tests of aircraft of any type including some aerospace vehicles.

INTRODUCTION

The Flight Research Institute (LII) continuously improve the operational characteristics of its airfield to conduct the tests of various aircraft, special complexes and airplane systems safely and effectively.

One of the main objectives of this research is the application of science to the experimental efforts to develop a new aerospace engineering involved in flight testing.

To make such efforts the designers and specialists of LII created testbed aircraft, unique airborne and ground-based instrumentation systems, original approaches and mathematic programs. The main and technical role of LII was determined by the fact that all these first systems had been flight tested under the LII methodological and scientific guidance. It was in LII that jet airplane, hypersonic delta airplane (Su-9, M-50) VTOL aircraft Yak-36, hypersonic passenger aircraft Tu-144 as well as the Shuttle Buran made their first take-off.

The technical aids of the LII allowed to set and register over 86 world records. Research and modification of the full automatic air refueling operations , the escaping procedure as well as the *g*-zero training of the first astronauts are some main important results obtained in LII.

Great efforts have been made to create a technical complex which would allow to solve successfully the problems of air target intercept and automatization of the landing procedure up to stopping the aircraft motion.

The first automatic landing was performed by the aircraft target Yak-25 in 1961. Over 1000 automatic landings in the ILS configuration were made by Tu-154. All the steps of the automatic landing and the landing runs of the prototype of the shuttle "BURAN" and version of "Buran" after testing in LII were improved. Some difficulties were encountered in making some automatic landings of the carrier aircraft but they have also been eliminated successfully.

To solve the two last problems the landing systems were used as the information systems in the centimetric-wave and millimeter-wave emission bands respectively.

The wide spectrum of testing caused the necessity of creating the test ranges of the various kinds.

Akhtubinsk base was used to test in flight some flying models and special systems. The long-range operations increased the number of the instrumentation stations up to the island Sakhalin.

The flight testing of the antisubmarine aircraft caused the necessity of creating the sea-based test ranges in the Crimea.

Some special test complexes to modify the carrier aircraft were also build in the Crimea.

Because of the disintegration of the Soviet Union into some independent states and because of sharp reduction of aviation funding the operational conditions in LII have essentially been changed.

1. In spite of the sharp reduction of the development of the new airplanes and helicopters their nomenclature as well as the testing procedure have remained without a change.
2. The changing conditions in the Crimea, Kasakhstan test ranges and bases have caused the necessity on the one hand to create some new ones (Gelendgik, Dubna), and on the other hand to revise and to modify the loading of the LII airfield.
3. It is necessary to use some new development according to the new technology in the test procedure (SKIPs-equipped aircraft instead of the ground stations the experiment control center and some complicated test techniques, the synthesis of the tests of the various kinds).

Establishment of the ground connections point (NUS) as well as the development of the ground equipment for evaluating the aircraft observability in the visible and IR range was of the especial importance for building the test range.

The above described efforts testifies the complicated and multipurpose approaches to form the test range department in the LII.

However the ideology of its forming failed to take into account the modern requirements of the complicated evaluation of the airborne and non-flight equipment and neglacted fast development of the aerospace engineering, the optimized tends such as in-flight equipment for general aviation and for sea aircraft.

Another factor of great importance is the necessity of taking into account the reduction of testing cost to the minimum under the new economical conditions.

The relative share of all expenses in the full flight testing capacity to design and develop the aviation engineering is 40—60%.

On the other hand ICAO formulated the concept of aircraft transport systems which offers not only their new features but some additional functional requirements necessary for them.

Fig. 1 shows the common airplanes traffic diagram block. As it is seen, this diagram block includes, besides aircraft, the airfield with its equipment , the air operational and data acquisition measuring system defining the weather limitations on the flight conditions, and the data base including the satellite communications and navigational channels as well as the air traffic control with digital air-to-ground communication channels. If an aircraft isn't equipped with the systems interacting with the airplane traffic system, or it does not meet the operational requirements of observing the flight rules, its flight will not be cleared.

Fig. 2 shows the list of conceptional requirements for advanced plane traffic system. These requirements are very capacious according to the functional purpose and quite complicated in terms of the technical level.

These new tends in developing the satellite technologies and in strict meeting some requirements to air traffic surely to result in the necessity to change the technology of conducting flight research and tests. It is clear that one of the objectives of this technology should be an increase in the flight test effectiveness. Fig. 3 verifies it.

The short characteristics of the LII complexes and facilities providing flight testing:

- communication and data link aids in the ground communication point (NUS) providing testing , modification and certification of the flight complexes of the various aircraft besides the satellite radio link. To solve the problems in the NUS to the full degree it is necessary to set up the connection with the airfield equipment operating in the "Horizont" range (band). The set of the semiscale and mathematical models offer the feasibility of the static true evaluation of the objects under test following flight testing. These evaluations are thought to be improved in the future fig. 4
- The satellite navigation, landing, ATC and tracking systems are estimated by means of the radio tracking and optical equipment as well as by the same satellite navigation system which operates in the differential range, that became possible

due to the method of mutual usage of air-to-ground data link for measuring the trajectories(VTU). It is expected to apply the method VTU in the flight test of aircraft as a main technique, fig. 5

— The basis of the radio measuring complex of the evaluation of the effective scattering surface of the aircraft is the radar stations of the various wave band with the control and test data processing system and further it is necessary to widen some operational frequency bands, to put into service the data link between RIP and PULE, and to create the data base.
The mathematical models provide the approximate design EPR of the aircraft, that reduces the capacity of flight experiments (LE). Further it is necessary to make the works for improving MM, to increase the design accuracy, to develop the models of the evaluation of the reliability of the result of the flight experiment, fig. 6.

— The electromagnetic compatibility (EMC) is estimated by the standard techniques following the specially developed procedures. The theoretical prediction of EMC reduces the flight experiment capacity, fig. 6
We need the devices EMP of the Rode Schwarz type for protection from the powerful electromagnetic emission, the program of the automatic evaluation of EMC of the aircraft in the flight. There are a lot of techniques of generation of EMC, and the system of the induced currents in the network of the aircraft (Bagerovo). It is necessary to create the electromagnetic test site in the LII fig. 7.

— The flight experiment control facilities PULE and VTI allow to conduct the flight tests of the various kind effectively. It is believed to use SNS for the trajectory measurements operating in the differential range, to provide PULE with the aerodrome facilities of the satellite data link for widening the area of conducting the controlled flight experiments, fig. 8.

Due to the tests conducted in 1993—94 the LII created the static control-correcting station of the system SR (both static and mobile V versions) which provides the feasibility of using SNS for landing VS and for measuring some trajectories. The conducted flight tests testified the high accuracy characteristics. The SNS in the autonomous regime with the radio channels MV can be used for air traffic control (ATC) in the aerodrome area but if it operates with the satellite radio communication system it can be used as the global observation (tracking) system (VP), fig. 9. Thus, the LII has the technical and methodological basis for building PIP (the unique tests range for the aviation.

CONCLUSIONS

1. FAI and the organizations have developed and used some new concepts of the development and application of the aviation engineering and set the standards which allowed to change to the new technological solutions to provide communications, navigation, landing tracking and organization of the air traffic.

2. The LII studies verified the high effectiveness of the proposed technologies (particularly the change to the satellite systems, the realization of the digital data transfer in the ground-to-air link in solving the problems of very high accuracy navigation and take off/landing operations.

3. The evaluation and certification of the respectively high precise in-flight and ground facilities will require to improve the existing LII base and to build the radio test range (RIP) meeting the requirements, of metrological estimates of accuracy, of integrity, continuity and argonomy effectiveness.

4. At present LII has the following facilities of RIP:
— the mobile control correcting station for operating in the differential regime of the high precise measurement of the coordinates of the aircraft providing VTI with the error more than 3—5 m;
— two channel ground-to-air digital data link;
— the set of the programs for building the given approach trajectory and for deviating from them while using the signals of the satellite system:
— the urgent equipment for evaluation and certification of the lightning protection, the electromagnetic safety, characteristics EPR;
— the procedure of the synthesis of the tests of the various kinds which allow to minimize the necessary amount of the in flight realizations due to the results of the mathematical simulation of the ground experiments with the real systems and due to the flight test results (some times).

5. The expected result can be shown by the following data:
5.1 The decrease of the data processing time to 2—3 times. It is possible to make the estimate in the real time.
5.2 The reduction in the flight regimes of the prototype (VS) 5—8 times
5.3 The total amount — the time reduction of the flight test to 10—15 times.

The Block Diagram of Air Traffic Systems. (ATS)

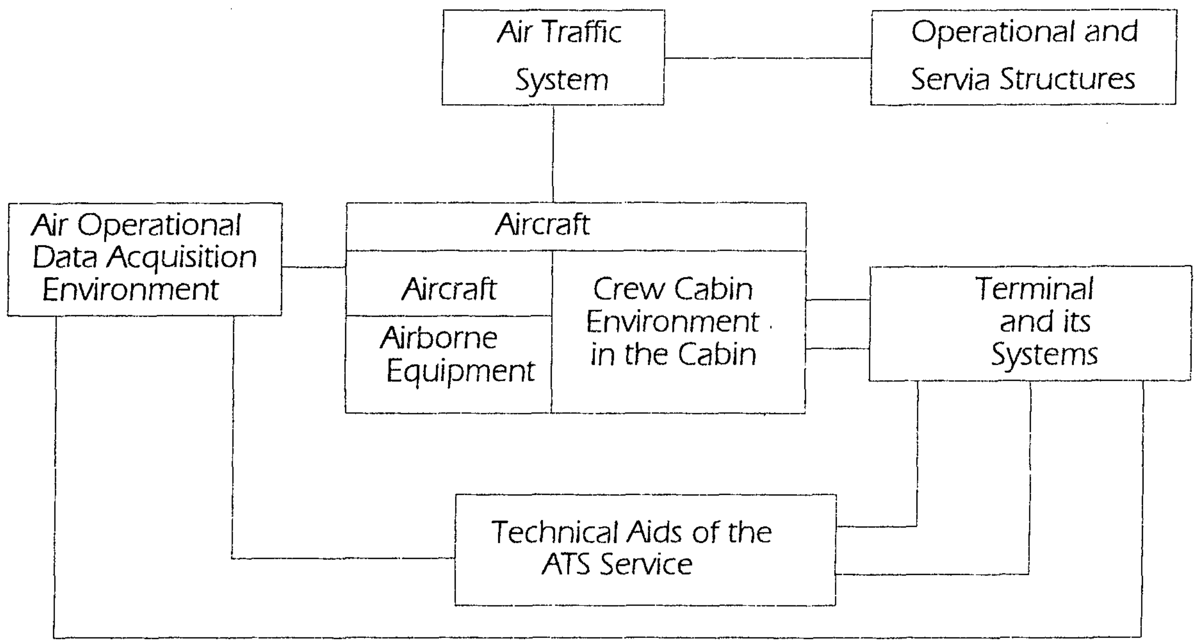

Figure 1.

The List of Conseptional Reqirements for Advanced Plane Traffic System

The List of Conseptional Requirements for Advanced Plane Traffic System

1. Functional - Qualitative Requirements
1.1. Global communications, navigation and observability (tracking) of the aircraft in any air space point.
1.2. All - Weather Landing.
1.3. Automatization of air traffic control procedure.
1.4. Automatization of the VS control procedure.

2. Technical Requirements
2.1. Accuracy $[Z] < 7$ m $[H] < 1.5$ m.
2.2. Integrity $(1- 3.3) * 10^{-9}$, Warning Time - 1 sec.
2.3. Continuity $(1- 4.0) * 10^{-6}$, (Duration - 30 sec).
2.4. Readiness 0.9990

Figure 2.

The List of The Factors Causing the Necessity of The Improvement of The Efficiency of The Flight Testing

1. The increase operational conditions for VS and aircraft, of the nomenclature of design regimes (all-weather transiosion to 300 m level up to 11000 m, high dendity of air space saturation.
2. Increasing requirements to the provability of flight test results (transition from the refence level $5 * 10^{-2}$ to $1 * 10^{-7}$
3. High flight test cost.
4. The increasing requirements to the errors of the other trajectory measuring some meters and even decimeters.
5. The complexity of the reproduction of all the design operations.

The necessary condition of the test efficiancy and of minimizing test expendure is the chage to the technology of the synthesis of the test results of all kinds (simulation, bench airplane- analog, prototypes, aircraft).

Figure 3.

RADIO COMMUNICATION AND DATA LINKS FACILITIES

AVAILABLE		REQUIRED
- All Wawe Band Aviation Radio Stations (HF, VHF, UHF)	HARDWARE	- Aerodrome Stations of Satellite Communication, Operating in "INMARSAT" "HORIZONT" modes
- P.N.M. Simulator Complex - MM LPD Packaged Programs HF, VHF, UHF ranges and Satellite Link	METHODICAL AND INSTRUMENTAL LINKS EVALUATION	- Equipment Complex to Evaluate LPD Operation are Intact and Continuity - MM Improvement to Provide Maximum Compliance with Tests Results

Figure 4.

LANDING SYSTEMS

AVAILABLE		REQUIRED
1. Systems of International Status 1.1. I.L.S. Category II 1.2. MLS Category II / III 2. Trimble Navigation (T.N.) Satellite System 3. PRMG - 5U VHF Landing System 4. Redundant Electrical Generating System.	EQUIPMENT	1. I.L.S. Category III A 2. GLONASS Satellite System with test-correcting station for II / III category 3. Updated TN for II / III category 4. Experimentators and Landing Instructors WorkStations 5. Multifunctional Radar of "ALMAZ" R.& P. Corp. with Integrated I.L.S., MLS and TN receiver.
1. I.L.S. and M.L.S. 2. PRMG - 5U and Satellite Landing system (SLS) for Uncategoried Approches 3. Appropriate SoftWare	TECHNIQUES AND SOFTWARE	1. Techniques for Advanced Modes 2. WorkStation Utilization Instruction 3. Techniques for WNR Certification 4. SoftWare for Advanced Systems and Models

Figure 5.

RADAR INSTRUMENTATION COMPLEX OF THE EVALUATION OF THE EFFICIENCY OF THE SCATTERING SURFACES OF AIRCRAFT (RIC ESS)

EQUIPMENT		REQUIREMENTS
1. Radar Instrumentation Stations with the 18- th and 17- th centimetric radiation band. 2. Secondary Radar Station of the type "COREN-A". 3. Flight Test Control (SU) System (PULE). 4. Control System of the Radar Instrumentation Complex based on 2 IBM PC AT 486 5. Flight Test Data Procession System in the Experiment Time "SEB". 6. The System of Common Time with accuracy up to 10^{-9} - 10^{-12} sec.	HARDWARE	1. Radar Instrumentation Stations for the 23- th and a 35- th centimetric radiation band 2. Two RadoiModem for Information exchange PIC — PULE. 3. The Complicated Control and Monitoring System. 4. The Data Base.
1. The Models of Approximate Disign ESS of Aircraft, the Active Response Range. 2. PMO for the Evaluation of the Flight Experiment Results. 3. The Models of Results and Design of Flight Experiment.	PROCEDURE AND PMO	1. The Models of Precise Designe ESS. 2. The Models of the evaluation of the Relibility of the Flight Experiment Results.

Figure 6.

ELECTROMAGNETIC COMPATIBILITY AND THE PROTECTION FROM OUTER ELECTROMAGNETIC FIELDS

	EQUIPMENT	REQUIREMENTS
HARDWARE	1. The complex of the aids of electromagnetics fields generation, the measuring instrumentation of fields and induced currents (p. Bagerovo). 2. Separate facilities in the Ramenskoe Airfield ARM "CM-14"	1. To improve some electromagnetic test sites and to build the test complex in the agreement with the rules. 2. To acquire the instrumentation complex "RODE SHVARZ", IBM PC AT 486.
SOFTWARE	1. The prediction of object and interobject electromagnetic compatibility.	1. Some programs of automatic airborn evaluation. 2. Some programs of the disign of operational fields by means of the full-scale test results.

Figure 7.

THE FLIGHT TEST CONTROL FACILITIES AND THE TRAJECTORY MEASUREMENT TECHNIQUES

EQUIPMENT		REQUIREMENTS
1. PULE providing the flight experiment in the direct visibility range. 2. The VTI Radar System (low accuracy). 3. The VTI Optic System (high accuracy, long data processing time, direct visibility).	HARDWARE	1. The Satellite Data Link (LPD) providing the flight experiment control in its operational range. 2. The VTI system using GPS GLONASS in the trajectory measuring in real time.
They are used fore the traditional informatic system in some flight elements.	PROCEDURE AND PMO	MPO is used for considering and applying the satellites to form data bank.

Figure 8.

Therminal Area Chart (The Ground Airtraffic Control Stations).
(GPS - Global Posision System)

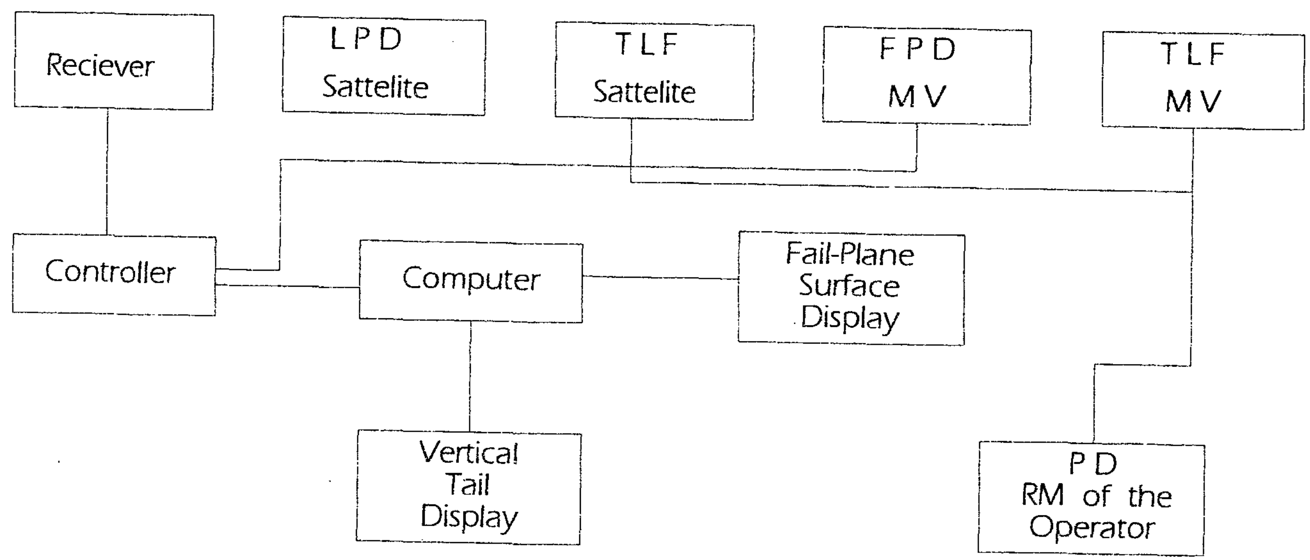

Figure 9.

TECHNOLOGY AND SCIENTIFIC-ENGINEERING MEANS OF DEVELOPING CIVIL AIRCRAFT/HELICOPTERS PROTECTION SYSTEMS OF VARIOUS APPLICATIONS AGAINST AIR AND GROUND TERRORISTS ACTIONS

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One of the most acute problems is designing and updating modern airplanes and helicopters, now in use in area of local conflicts or possible terrorist actions, is a provision of protection against low-altitude destruction means — "surface-to-air" and "air-to-air" guided missiles. On the whole, the protection problem appears to cover a number of factors:

— development and manufacture of universal means and devices of optoelectronic protection;
— development and manufacture of mating and control devices for optoelectronic protection systems;
— selection of a unit of protection devices for a given type of aircraft (helicopter);
— arrangement of protection devices on board the air-vehicle;
— evaluation of onboard protection means effectiveness and recommendations to be worked out for their application.

Recently the Gromov LII, GosNIIAS and other enterprises have developed, introduced and got an official approval of the technology to solve the task indicated, the technology having been finally evaluated in real-life conditions on 10—15 aircraft.

The technology is based on an experimental calculation method of problem solution resulting in obtaining protection effectiveness evaluation for a given type of aircraft.

To implement technology development for selecting the airplane (helicopter) protection system it is required to collect and analyze initial data. Initial data includes:

— carrier aircraft flight performance characteristics;
— flight routes and profiles;
— IR-radiation of the aircraft engine and air-frame at various flight modes;

— climatic conditions in flight test regions.

As practice shows, the most popular and widely used by almost every army in the world IR protection means are flares to be dispensed.

In this connection there is analysis performed of initial characteristics which determine the FIT's effectiveness.

They are:
- — spectral and integral responses of IR-radiation;
- — time characteristics of burning;
- — direction and speed of jettisoning;
- — a priori given spectral characteristics of missiles IR-homing heads and principles of antijamming.

While determining the make-up of an airplane IR-protection system and evaluating its effectiveness one takes into account the influence of different factor upon the effectiveness, with the parameter of a greater effect being corrected or changed. In general IR-protection stands for:
- — a set of measures providing the required IR-radiation parameters and visual acquisition range;
- — flares dispenser with a control system permitting the selected jettisoning rate and number in a salvo to be implemented, depending on the application conditions;
- — flares whose spectra-power characteristics are functions of time, altitude and speed of airplane flight, and whose trajectory characteristics are affected by the dispenser conditions.

Provided there is required some of information and mathematical models available, the make-up that is the composition of an IR-protection system for a given air vehicle can be defined, and recommendations elaborated for its location and application.

After the systems' make-up is defined, there comes the designer and technological documentation stage dealing with the dispenser installation onboard, then the installation stage, and that of the necessary ground evaluation and development.

Flight testing is to be conducted to confirm the calculated data and effectiveness evaluations of the technical solutions.

The LII and GosNIIAS have worked out, for the latest years, and tested MIRK Mobile Measuring-recording unit to provide optoelectronic protection of aircraft against "Surface-to-air" and "air-to-air" heat-seeking missiles in flight conditions.

The Unit allows the following tasks to be solved:
- — evaluation of effectiveness of airborne optoelectronic counter-measure means, both expendable and non expendable, used separately and as a unit;

— elaboration of recommendations for IR-means application: algorithms for their use and determination of the required onboard reserve of them;
— development and improvement of effectiveness evaluation techniques of airborne optoelectronic counter-measure means in real-life conditions;
— provision of training of airplane (helicopter) crew members and evaluation of their actions in using optoelectronic protection means.

MIRK unit consists of several functional units:

— receiving-measuring equipment including full-size samples of practically all types of IR-homing heads of portable antiaircraft guided missile systems, both home-made and foreign, as well as IR-homing heads of "air-to-air" missiles. This measuring unit, having various types of IR-homing heads of guided missiles, permit evaluation of a jamming signal effect when it gets into the electronic section of the homing heads with different antijamming arrangements in flight conditions. To do away with technical problems caused by the fact that in real-life conditions the antijamming systems of IR-homing heads are engaged and operate effectively in the trajectory after the missile launch, there was conducted further development of the IR-homing heads electronic sections to meet real-life launch requirements;
— matching and control-recording instrumentation, providing the necessary bearing and correction signal conversion for several types of IR-homing heads, from the coordinate system to the Cartesian one for the signals to be entered into a personal computer, and performing analog-digital conversion of appropriate analog signals and one-time commands from IR-homing heads outputs. The "unit" carries out digital processing of a television picture of the full-scale experiment, with air-vehicle's and jamming signals contours determined to find their mutual coordinates to be subsequently input in the models. On the whole, the "unit" provides acquisition and analog-digital conversion of output parameters as well as initial date file formation as regards the experiment information to use in the "Dispatcher" program and carry out automatic processing of the information simultaneously with the experiment;
— a visualization and display unit serves to show on the screen the flight experiment under way and prompt evaluation of the noise effect upon the IR-homing head, as an operator at the monitor can see the target, noise signal, boundaries of the homing-head's momentary vision field and auxiliary

information and symbols delivered by the alpha-numeric data generator;

— a control and communications unit makes it possible for a leading engineer to get in contact with the aircraft systems and possibly with the flight control officer.

MIRK Unit can be installed in any place where there is the mains power supply of ~380V,50Hz and gas nitrogen in a bottle or from a gas-station.

Prior to testing there is conducted checking of the most significant and informative parameters of the homing heads as well as the homing head adjustment with a remote object and test parameters registration.

The flight experiment is to be conducted in the day time or at night, in accordance with the techniques developed, and with use of MIRK Unit, the aircraft passing near the Unit being performed. The flight mission includes information on the carrier-aircraft's course which ensures the target aspect angle prescribed; altitude, flight speed, passing maneuver to be performed, power setting, flares dispenser mode, number of flares in a salvo, interval between dispenser in a train, the beginning of dispenser, aircraft of dispenser, distance to the dispenser point and land-marks.

Operators at every optical station of MIRK Unit are supposed to check IR-homing heads readiness for operation; perform IR-homing head target guidance with a zero bearing or with a fixed lead bearing; provide transition of the homing head's "caging" mode to "tracking" one and back; make a visual evaluation of the passing being performed and reject noise signals on the oscillograph or monitor and establish voice communications with the experiment manager.

Synchronization of all the equipment in operation is done with the aid of common timing signal (CTS) through the radio channels sent from the aircraft at the time of interference rejection.

The Unit's hardware and software permit automatic current data processing to be done at the rate of the experiment, and PC data input to simulate guided missiles launches with the interference rejected and obtaining a final evaluation of IR-counter-measure system effectiveness.

There have been developed algorithms and programs for TV-data processing and these help determine the contours of aircraft and noise images. The software packages for interference detection are selected after the carrier's preliminary target designation, with characteristic features of interferences being indicated: carrier proximity, exponential attenuation boundaries and others.

The visual-data algorithm allows 1—5 moving objects to be detected, their coordinates to be established in the system tied to the Unit's turret and their use in models.

An algorithm for IR-homing head's output signals to be entered in PC has been prepared to ensure automatic current data processing at the rate of the experiment. In this case, F = 1—1.5 KHz carrier frequency signal are detected smoothed by filtration and converted to a digital code; bearing and corrective signals are delivered to the phase detectors which separate orthogonal components; one-time commands are converted, which are of constant voltage registered with a time accuracy of 0.05—0.1 sec.

It is convenient and convincing to present flight test results in tabular form with indications of results in tabular form with indications of flight conditions, power settings, airplane aspect angle, flares dispenser regimes and the fact of an interference effect upon the IR-homing head.

Evaluations of IR-homing head operation can be presented in various ways:

— reaiming at the interference is accomplished;
— reaiming at the interference is not accomplished;
— repeated target lock-on after the interference burn is accomplished;
— reaiming is not accomplished as the kinematic selection system is operative;
— reaiming is accomplished with a time delay due to the operation of the kinematic selection system;
— interference capture is implemented without a preliminary target lock-on.

In general the MIRK Unit's hardware and software for automatic data processing provide evaluations of error probability of aircraft tracking mode at the rate of an experiment and formation of an initial data file for modeling to obtain evaluations of IR-protection means effectiveness and statistical analysis of the experiment data.

The results of modeling are represented as time-dependence graphs of miss, angular velocity, sight line, current distance to the target, elevation and azimuth components of the gyroscope angle, corrective signal, bearing, pitch and yaw angles and three coordinates. Based on the results of modeling there can be determined protection areas for the prescribed flight and interference rejection conditions, these places are zones of effective and non-effective protection, they are of a different area and effectiveness (probability) on the guided missile miss criterion.

Express-analysis of the experiment visual image, automatic processing of data on every airplane passage at the rate of the experiment and, if required, statistical analysis permit working out recommendations on interference usage for an aircraft under study at various flight modes-climb, en-route cruise, approach and maneuvering as well as determining the onboard ammunition establishment.

In conclusion it should be noted that MIRK Unit, unlike any individual measuring-registering device used before, has considerably improved the reliability and effectiveness of flight test results and evaluations of optoelectronic counter-measure means effectiveness.

A similar technology to be corrected for a specific protection system is recommended by the LII and GosNIIAS for use in developing protection means against terrorists in civil airplane passenger cabins.

EXPERIMENTAL STUDY RESULTS ON APPLICATION OF GPS DIFFERENTIAL OPERATING MODE FOR LANDING OPERATIONS PERFORMANCE

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This report presents the results of flight tests carried out in the LII named after M.M.Gromov on application of the satellite navigation systems in providing landing approach. These results regard to autonomous GPS application as well as to its integration with other airborne systems (ILS and MLS).

Currently in many countries are conducted studies on estimation of possible application of the global satellite navigation systems (GNSS) for the performance of landing operations. The results of these studies show that such systems when operating in the differential mode could provide the aircraft position accuracy enough for the performance of landing operations.

ICAO considers DGNSS to be one of the primary systems (along with MLS and ILS systems) to provide a landing approach, landing and take-off. But at the same time "Global strategy of transition to new technical systems for landing approach, landing and take-off support" developed by ICAO has the following provisions:

— ILS flights will continue there, where this is justified from the operation and technical points of view;
— MLS will be introduced there, where this is justified from the operation and technical points of view;
— it is being evaluated the DJNSS application for Category I flights, and studies of the possible application of DJNSS with a required function addition for Category II/III flights are on their final stage.

So far, on certain stages of transition to the new systems there will take place a simultaneous utilization of several primary systems that could lead to a new quality in provision of landing operations. In particular, it is seem quite possible to create a hybrid system combining DGNSS with the existing elements of ILS glideslope equipment or the elements of MLS elevation equipment. Such hybrid system could present a number of advantages that will allow to:

— eliminate interference related to ILS localized;
— provide landing approaches including curved approaches from a point selected for some operational reasons.

In its turn ILS and MLS equipment present a good addition to DGNSS providing a required accuracy in the vertical plane. The Comparison of ILS/MLS guidance data with an artificial glideslope trajectory calculated from DGNSS data could improve the integrity of the system due to the independence of two data sources.

In the places where ILS/MLS systems have been already introduced into service their utilization in accompany with DGNSS could improve the in-service integrity of the system.

The cost-efficiency analysis of the MLS introduction has shown that the cost of on-board and ground-based DME/P equipment is about a half of the total equipment cost. The DME/P means could be excluded from the MLS equipment in case of joint application of MLS and DGNSS.

In 1992—1994 LII named after M.M.Gromov conducted flight studies on the evaluation on automatic DGPS approach. The results of the studies completely confirmed all the above-mentioned in regard of joint application of DJNSS and carrier implemented primary ILS and MLS approach systems.

Fig. 1 & 2 present the curves of the aircraft deviation from the approach path set by the equisignal zones of ILS localizer and glideslope beacons on one hand and preprogrammed on airborne GPS hardware so that it should have to concur with the ILS equisignal zone on the other hand. The approach was performed in automatic DGPS mode up to the height of 60 meters. The almost complete concurrence of curves is an evidence of high pilotage accuracy and possibility to perform ICAO Category I approaches.

Fig. 3 shows the curves of the aircraft deviation from the extended runway center line calculated on MLS data using data at distance to azimuth beacon computed with the help of DGPS and received directly on the output of GPS receiver operated in the differential mode. The concurrence of the results confirms that at provision of a required signal integrity level the DGNSS system can be a source of distance data for MLS systems instead of DME/P equipment when performing the categorized approaches.

Alongside with the development of DGNSS approach mode in the flight tests there was made an evaluation of coordinate determination accuracy of an aircraft-laboratory. It was made in comparison of "Opal-Amber" cinema-theodolite stations data with DGPS data.

The example of such comparison is presented in fig. 4. The statistical processing of a sample consisting of 245 points has showed that at 95 % probability the absolute accuracy of aircraft positioning doesn't exceed 5.2 m value practically for all coordinates (latitude, longitude, altitude). Such accuracy allowed to apply the dependent observation mode for the arrangement of automatic controller workstation (ACW) for circle controllers. Air situation was displayed on the display of a personal computer. According to the conclusion of flight directors the informative level of the displaying data was significantly higher than in traditional means.

The conducted flight tests have showed that the DGNSS-based landing complex consisting of an on-board GNSS receiver and communication controller and also of a ground station for differential corrections formations and data received on-board has a high level of mobility and can provide aircraft homing to an airfield, prelanding maneuvering and landing approaches on insufficiently equipped aerodromes and contemporary landing places. The same equipment can be utilized for the high-accuracy trajectory measurements in all-weather conditions without additional equipping flight routes.

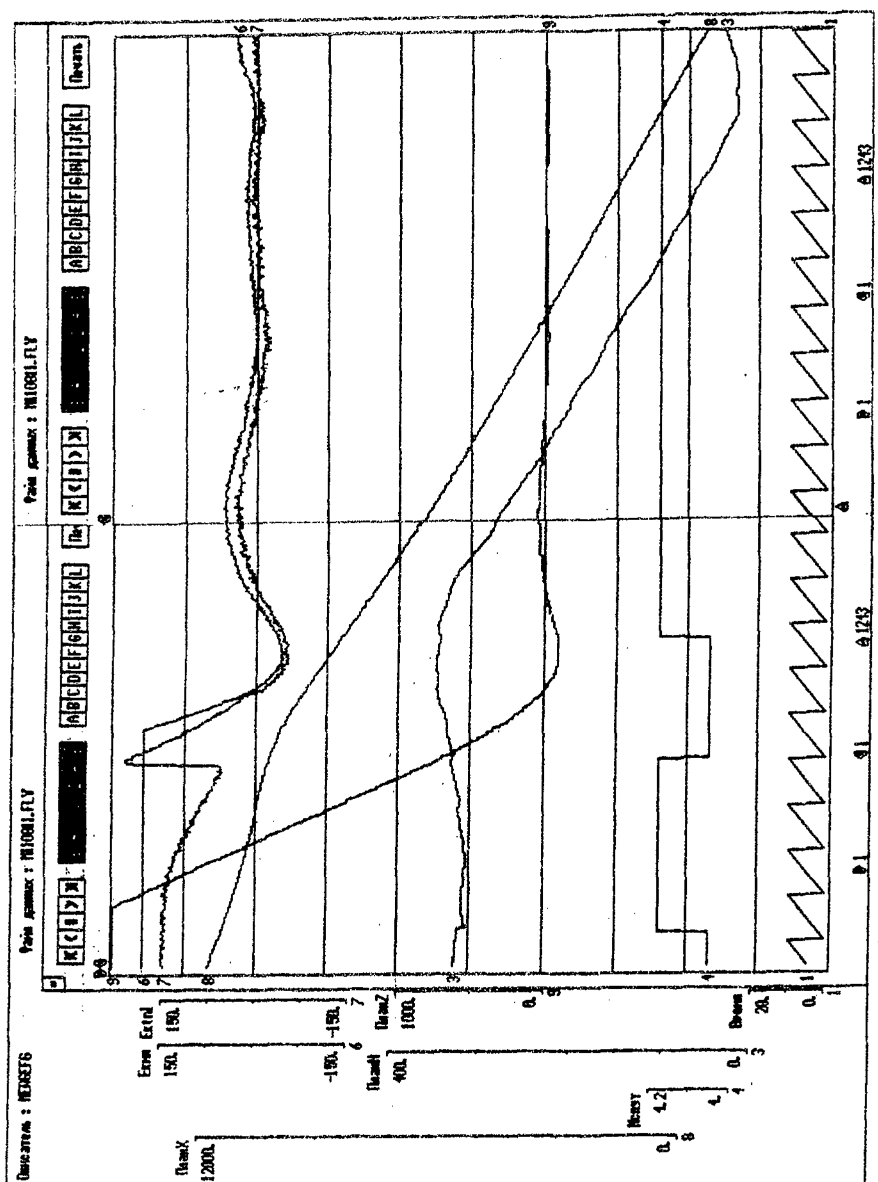

Figure 1.

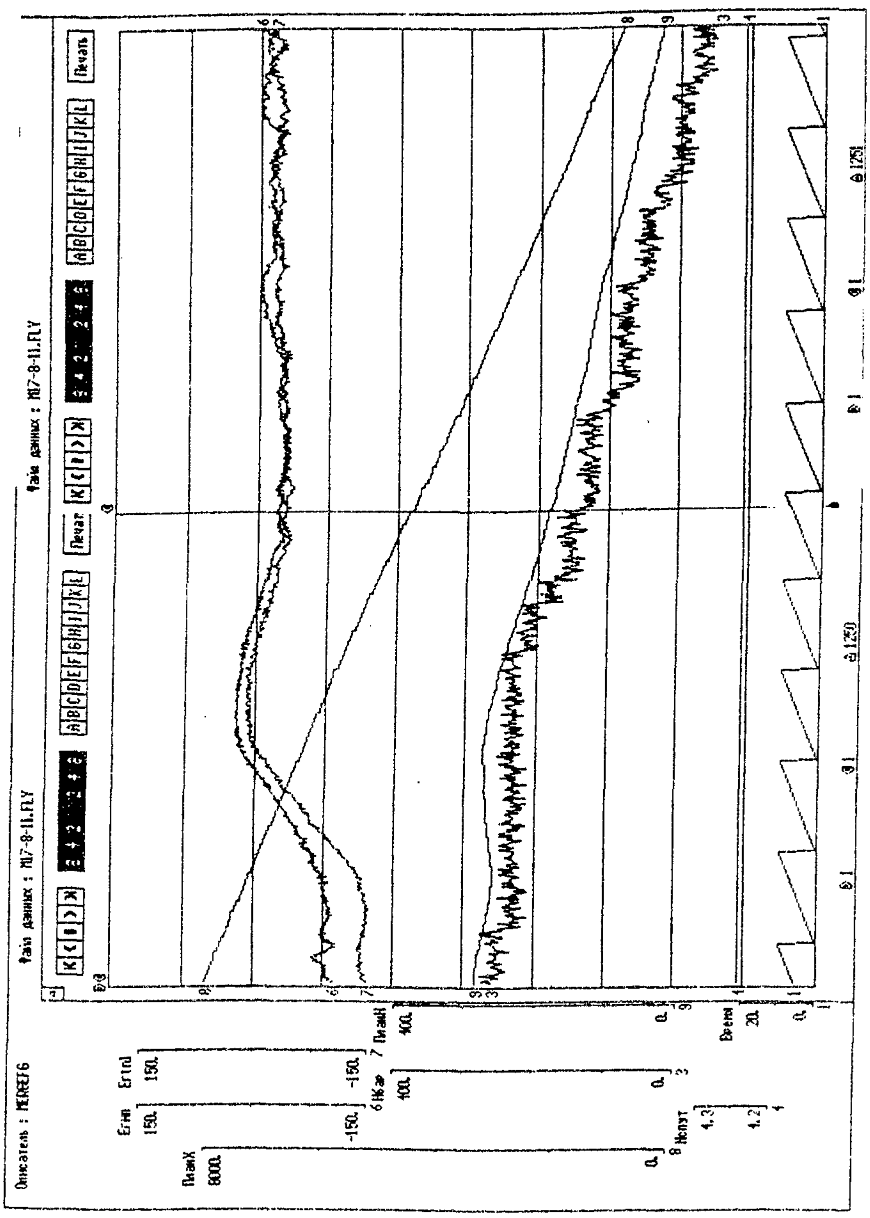

Figure 2.

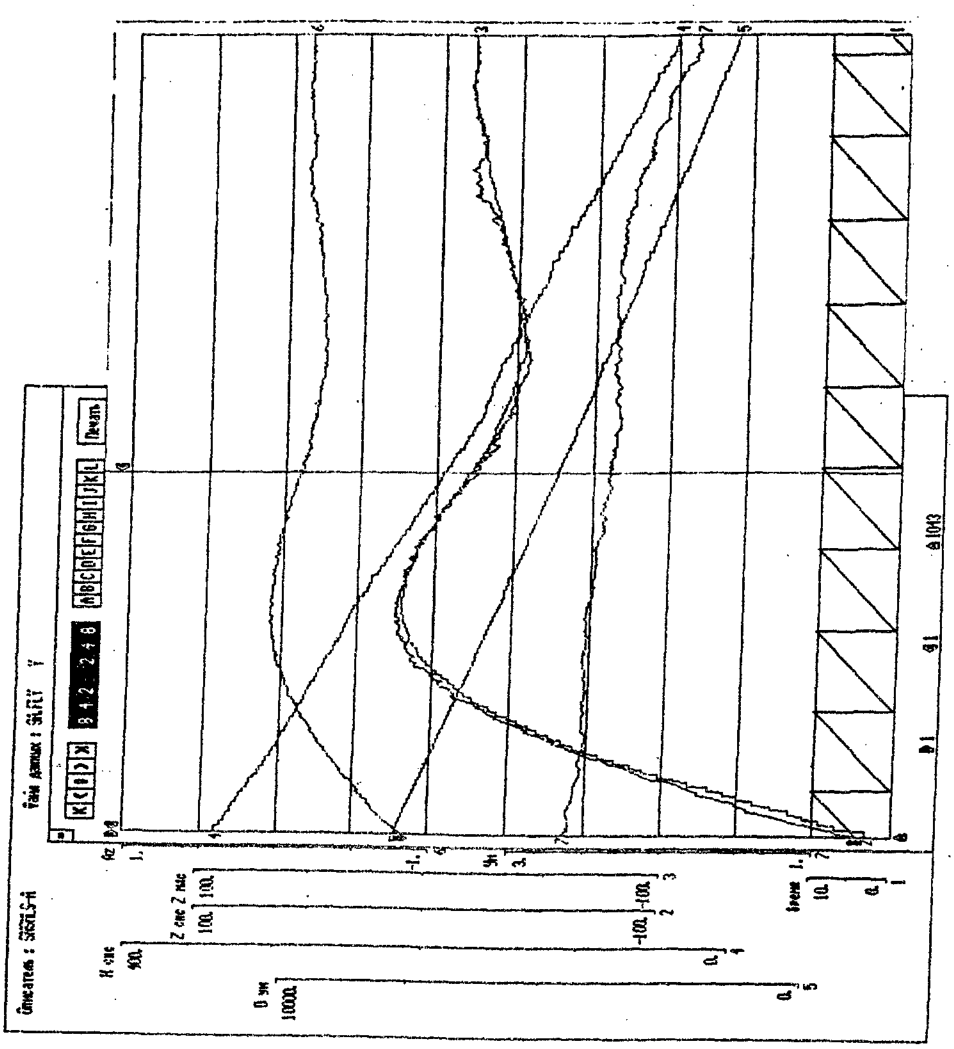

Figure 3.

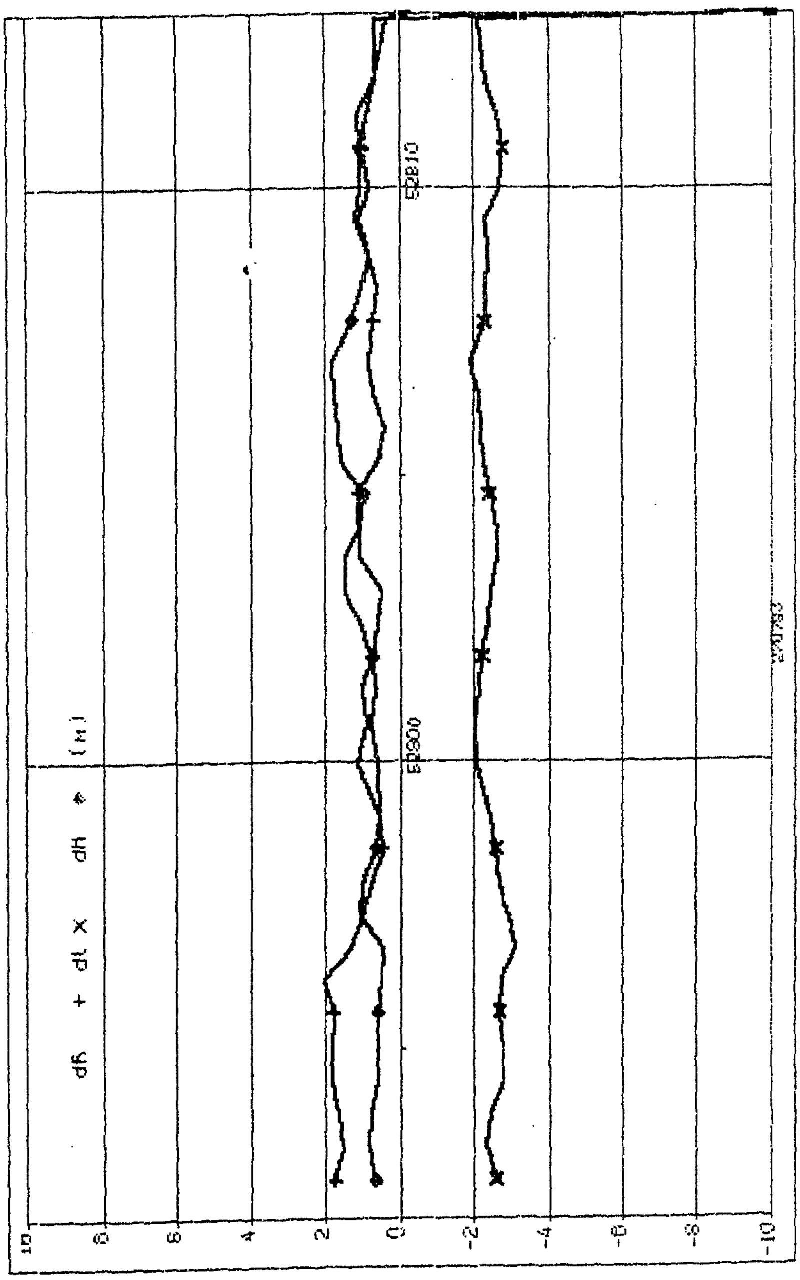

Figure 4.

TECHNIQUE OF CONDUCTING INTEGRATED STUDIES, TESTS AND CERTIFICATION OF ADVANCED AERONAUTICAL RADIOCOMMUNICATION UNITS AND MEANS

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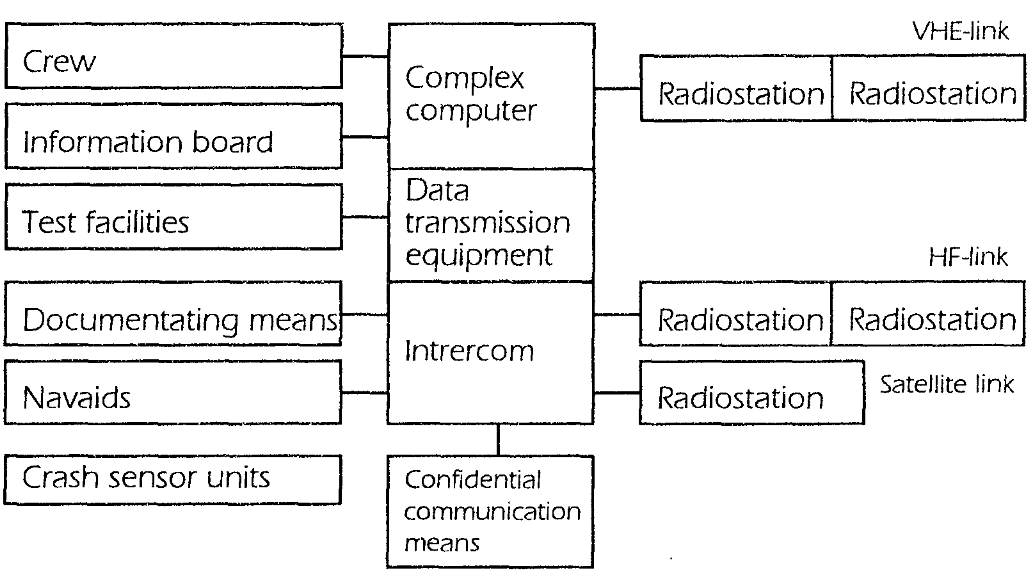
Main Functional Tasks Being Solved by Aeronautical Radiocommunication Means:

- Voice communication via HF, VHF and satellite communication links with ATC services and airplanes
- Computer-aided and automatic data exchange between crew/airplane systems and ground-based ATC aids
- Crew warning about emergency situations using automatic transmission of emergency system signals and flight data via all aircraft radio lines
- Digital command information display (documenting), telephone information documenting
- Intercom telephone communication
- Automatic control/monitoring of radiocommunication means



Block-scheme of airborne radiocommunication complex

EVALUATION OF AERONAUTICAL RADIOCOMMUNICATION MEANS OPERATION BY SOME BASIC IN SPECIFICATION REQUIREMENTS

Basic criteria for evaluation of aeronautical radiocommunication means during tests:

— Communication reliability of the selected range for the voice communication (VC) and telephone data exchange (TDE) modes of operation:

$$R_{VC} = \frac{K_{com}}{K_{\Sigma}};$$

$$R_{TDE} = \frac{Q_{com}}{Q_{\Sigma}};$$

K,Q — number of two-way communication contacts in VC and TDE modes of operation;

— Validity of Digital data transmissions (Receive)

$$P_{val} = \frac{m}{n};$$

m — number of properly received messages,

n — number of transmitted messages;

— Link quality

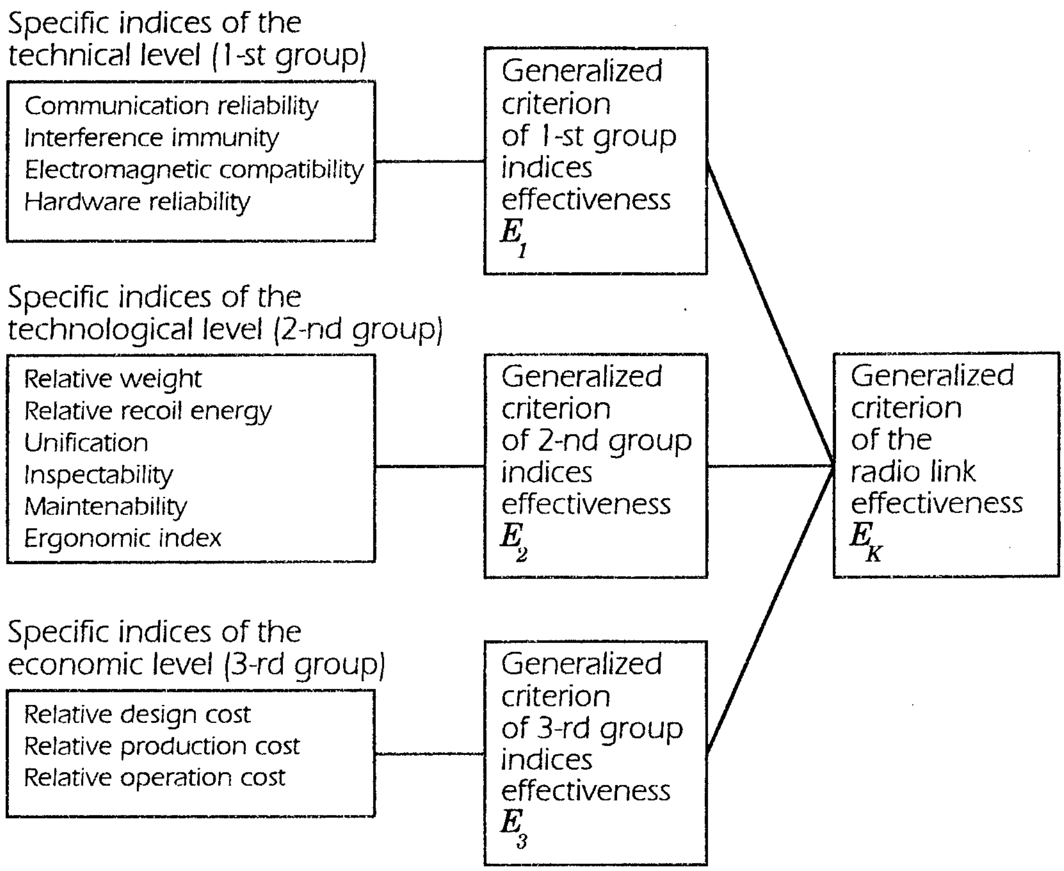
$$P_o = \frac{P}{Q};$$

P — number of the erroneously received primitive messages,

Q — number of the received primitive messages;

— Speech intelligibility in the telephone mode of operation is determined by IKCC 5-number scale according to GOST 1660-79 document.

Integrated evaluation of aeronautical radiocommunication means by a set of basic indices by "Effectiveness—cost" criterion



Scheme of indices and criteria that determine the aeronautical radio communication means effectiveness in performing their basic functional tasks.

Generalized criterion of the aeronautical radiocommunication means effectiveness

$$E_K = \sum_{i=1}^{I} \beta_i, \quad E_i = \sum_{i=1}^{I} \beta_i \sum_{j=1}^{J} \alpha_{ji} Q_{ji};$$

Q_{ji} — specific indices of the aeronautical radiocommunication means effectiveness,

j — number of specific indices,

i — number of index groups,

α_{ji}, β_i — weight coefficients of the specific indices and index groups.

Test Evaluation Validity of Aeronautical Radiocommunication Means

$$\gamma = Ver\left(\left[P - P^*\right] < \Delta\right);$$

P — desired parameter,

P^* — parameter received in the result of experimental data processing

Δ — selected accuracy (confidence interval).

Values of the minimum experimental data volumes required for the statistic evaluation of the aeronautical radiocommunication means operation

For the evaluation of communication reliability:

Type of radio link	Confidence level of evaluation γ	Number of communication contacts		
		E<0.3	E<0.3	E<0.3
HF	0.95	>45	>100	>380
VHF	0.98	>60	>130	>530
Satellite	0.98	>60	>130	>530

Confidence level of evaluation	Number of communication contacts		
	E<0.3	E<0.3	E<0.3
0.9	>110	>250	>1000
0.95	>220	>500	>2000
0.98	>550	>1250	>5000

$E = \dfrac{\Delta}{P^*}$ — relative evaluation accuracy.

Required predicted time for the evaluation of the aeronautical radiocommunication complex performance:

civil aircraft — 40...100 flight hours;
military aircraft — 500...2000 flight hours.

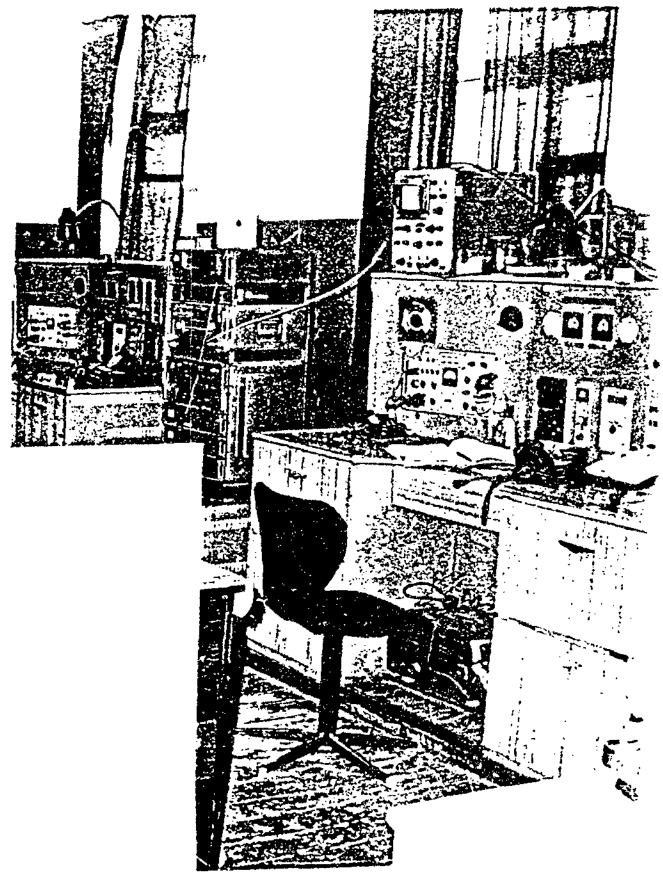
Basis methodological principles of test technique for aeronautical radiocommunication complexes and means

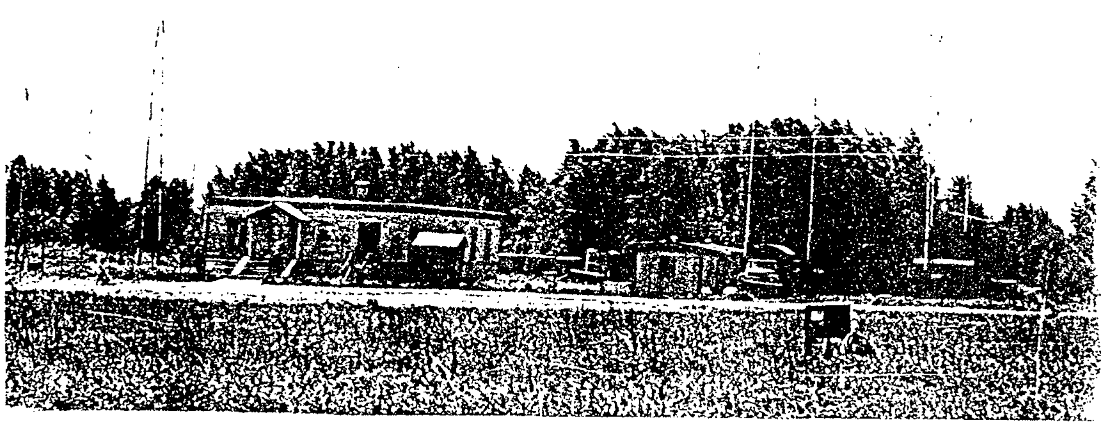
— Combination of simulation and flight tests by the results of the experimental work:

Simulation — 80 %	**mathematical modeling — 60 %**
	hardware-in-the-loop simulation — 20 %
Flight tests — 20 %	**on a flying test-bed — 10 %**
	on an aircraft — 10 %

— Provisions for convergence of simulation and flight tests results
— Simultaneous development of radiocommunication means, hardware and software test support, preliminary development of radio link mathematical models
— Provisions for quick acquisition and analysis of flight test results by using the capabilities of on-board radiocommunication equipment computers.

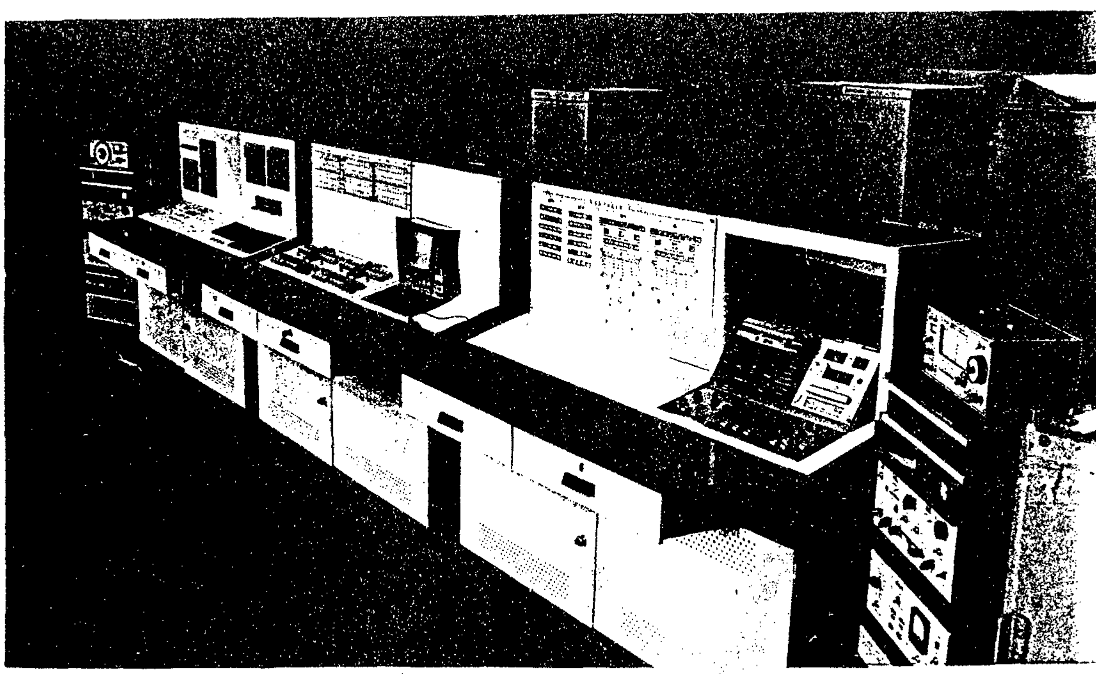
HARDWARE TEST-MATURIZATION COMMUNICATION COMPLEX





Ground test-maturation communication complex:
— occupied square 25 hectares;
— standard antenna fields;
— radio stations utilized in civil and military aviation;
— digital radio exchange hardware;
— confidential telephone and digital communication aids;
— measurement-recording complex.

HARDWARE TEST-SUPPORT



Hardware-in-the-loop simulation complex for aeronautical radiocommunication systems:

- — channel-formation and terminate equipment for aeronautical radiocommunication;
- — control computer;
- — simulator of aeronautical communication links, interference, communication aerials;
- — measurement-recording equipment.

SOFTWARE TEST SUPPORT

Application package of radio link mathematical models for HF, VHF, satellite aeronautical communication and evaluation program of data communication links by "effectiveness—cost" criterion.

INTEGRATED TECHNIQUE APPLICATION

Comparison Data on Communication Reliability Evaluation Received During Simulation and Flight Tests HF–radio link in the data exchange mode.

Data on effectiveness Evaluation of Aircraft On-board Complexes:

Compliance level of the prototype complex characteristics predicted in the Specification Requirements		
Complex type	Generalized complex effectiveness criterion E_c	Generalized criterion of technical level indices E_t
Heavy aircraft complex	0.6–0.75	0.55
Light aircraft complex	0.52–0.7	0.32

The presented material is based on the test results of radiocommunication complex for the An-124, Tu-160, Su-27 aircraft.

INTEGRATED TECHNIQUE APPLICATION

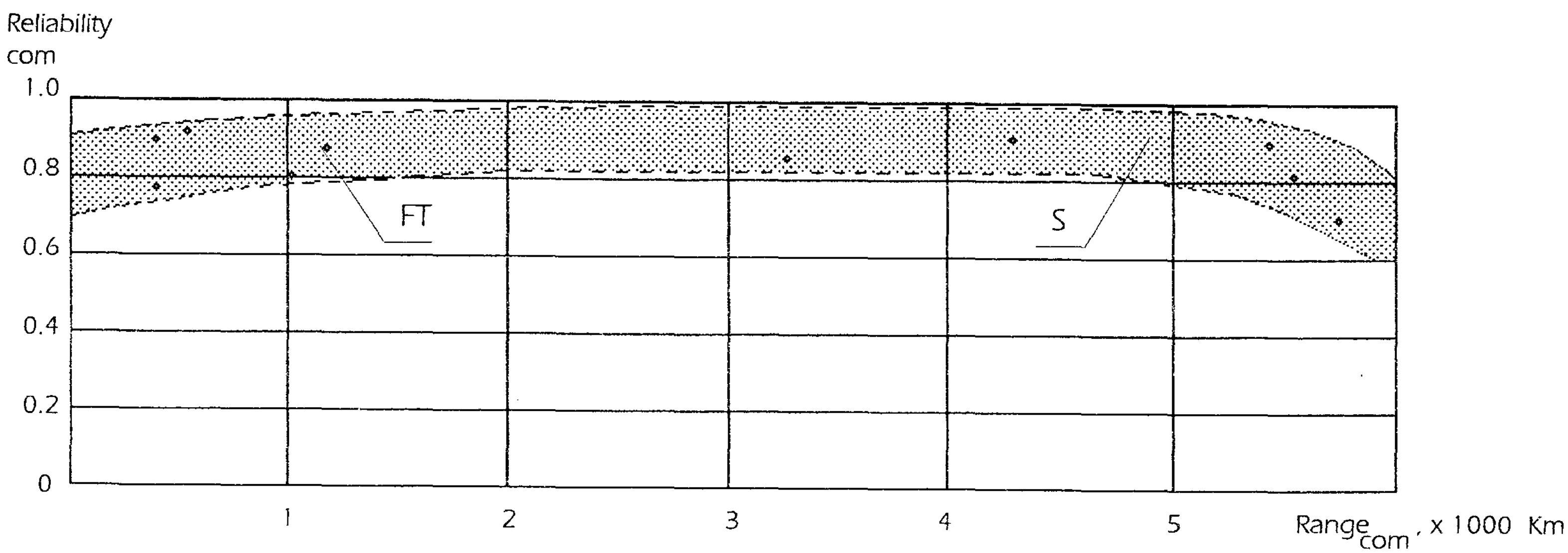


Figure 1.

* *Comparison Data on Communication Reliability Evaluation Received*
Duaring Simulations and Flight Tests (FT) Hf- radio link in the data exchange mode

METHOD FOR DETERMINING A DISTURBING FACTORS VALUE THAT PERMITS SPACEPLANE GUIDANCE AND LANDING CONTROL FROM THE GROUND CONTROL POST

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The Ground Control Post (GCP) is one of the possibilities to enhance the reliability of spaceplane control function. The GCP takes upon a particular importance in the phase of preliminary and aerospace tests.

The GCP is primarily designed to identify abnormal situations and to form recommendations on how to minimize their impact on the control function execution.

But characteristics of the control post itself and of the systems that assure its functioning must allow the realization of this main task.

In the present paper an effort is made to determine, as an example, the regions of allowable variation of some GCP characteristics for the phases of pre-landing maneuvering and approach. In particular, the following issues are considered:

— Measuring errors for the position vector of a subject to be controlled and velocity vector errors as a rule functionally related to them;

— Power of information flow on spaceplane attitude which may be interpreted as step-type behavior of measurements updating; and

— Total track lag of measurements due to the processes of reception-transmission, coding-deciphering and processing to obtain command signals in GCP spaceplane closed control loop.

To illustrate the method we shall restrict our consideration to the phase of pre-landing maneuvering.

Let us suppose that there is a mathematical model of GCP spaceplane closed loop. It includes a model of spaceplane center-of-mass motion, a model of center-of-mass motion control algorithm and a model of data measurement and communication system.

The motion model represents a system of sixth-order non-linear differential equations in the wind-body coordinate system. Limits are imposed on the following motion parameters: roll angle $|\gamma|$, roll rate $|\omega_x|$, equivalent airspeed derivative $|dV_i/dt|$, equivalent airspeed range versus flight altitude.

The data measurement and communication model contains data on spaceplane attitude in the form of range — D, azimuth — A_z, elevation angle — E, which arrive with step-type behavior — t and gaussian centered noise along the appropriate axes σ_D, σ_{A_z}, σ_E. The total time lag between spacecraft attitude measurement and control action in response to the signal is $\_C$.

Course angle (Ψ_C) and equivalent airspeed (V_i) are obtained by processing the measurements ($\{D, A_z, E, t\}$ $\{H, X, Z, t\}$ $\{V_x, V_y, V_z\}$ $\{\Psi_C, V_i\}$).

To derive the state vector (X, Z, H, V_x, V_y, V_z) from the measured position vector (X, Z, H) an algorithm based on a finite memory digital differentiating filter was used. It should be noted that the delay due to the filter is equal to zero if the glide is steady in roll and equivalent airspeed. In case of active spaceplane maneuvering in roll and equivalent airspeed the filter can produce a delay up to 6 sec. By the control system model is meant an algorithm of spaceplane state estimation and algorithm of control action formation.

When constructing the GCP/spaceplane closed control loop an integrated estimate of spaceplane center-of-mass position in (X, Z, H, V_x, V_y, V_z) space is used, this is altitude excess.

The altitude excess (ΔH) is a function of the preset altitude of the extreme trajectory termination. The extreme trajectory carries the spaceplane from the current phase state to some final phase state with minimum altitude loss.

The extreme trajectory is calculated by the estimation algorithm every time the spaceplane attitude measurements are updated.

The control algorithm producing control actions keeps track of the predetermined motion program in the plane "altitude excess-altitude" and forms the control parameters: preset course $\Psi_{pr.}$, preset equivalent airspeed $V_{i_{pr.}}$, preset airbrake deployment angle.

The algorithm functioning is based on the main property of the extreme trajectory. When moving along the trajectory the spaceplane altitude loss remains constant, whereas at the motion along any other trajectory the altitude loss decreases. As the extreme trajectory is unique, its basic characteristic — altitude loss — can be used in forming control actions. Apart from altitude loss, the data on all the components of the state vector is use in the control algorithm.

Depending on the values of input vector elements one of three trajectory types can be realized: open-type and close-type trajectories and that of precise guidance to the check point.

Method for determination of an admissible level of disturbing factors

Let us enter the notion of guaranteed programs area (GPA) as a set of programs in the plane $(H, \Delta t)$. When keeping check of these programs the remote control algorithm brings spaceplane from the multitude of admissible initials states into the specified final state.

The GPA is formed in the following way. 11 points uniformly distributed throughout the area of admissible initial states are selected for the initial altitude in a random manner.

The motion program is given by the piece-linear function of the form:

$$\Delta H = \begin{cases} K \cdot (H - H_{const}), & K = const, \quad H > H_{const} \\ 0 & when \qquad H < H_{const} \end{cases}$$

Then we choose a reasonably great value K and simulate closed-loop system operation for all the preselected initial points. If the spaceplane final state obtained in each realization corresponds to the specified area the program is considered belonging to the GPA. Then K is reduced and the simulation is performed once again. The procedure is repeated until the spaceplane final state stops corresponding to the preset one, if only for one realization. In this case the GPA calculation is over.

To avoid the impact of random nature of a single realization the described procedure is repeated several times. Thus, we have in the plane $(H, \Delta H)$ some restricted area in the form of a sector.

The sector contains ΔH_{pr} (H) programs , following which the algorithm puts the spaceplane into a specified final state among all the admissible initial states.

The size of the GPA obtained in this way depends on the level of disturbances (in this case: information channel noise, total control loop lag, control action interval). It is precisely this property of the GPA that was used in determining the disturbances area within which the GCP spacecraft closed loop is able to solve the set problem. The heart of the method is to find a level of disturbing factors at which the GPA takes measure "zero".

Research Results

The first phase covered the investigation of individual impact of each factor on the system. The numerical value of a disturbing factor increased to the point that the GPA degenerated into a line. The values: $\sigma_D^{''}, \sigma_{A_z}^{''}, \sigma_E^{''}, \_C^{''}, \_t^{''}$ gained by this means were taken to be normalizing in the investigation of overall disturbances impact.

The joint distribution of $\sigma_D^{''}, \sigma_{A_z}^{''}, \sigma_E^{''}$ admissible values for the fixed $\_C$ = 1 sec, $\_t$ = 1 sec is described by the following equation:

$$\left(\frac{\sigma_D}{45}\right)^2 + \left(\frac{\sigma_{A_z}}{4}\right)^2 + \left(\frac{\sigma_E}{4}\right)^2 < 0.65^2$$

$$\sigma_D^{''} = 45m, \sigma_{A_z}^{''} = 4', \sigma_E^{''} = 4',$$

$$\sigma_D > 0(m), \sigma_{A_z} > 0(\text{min}), \sigma_E > 0(\text{min}),$$

This result is supported by 3000 simulations of control process.

Adding the lag resulted in a four-dimensional area of admissible disturbances:

$$\left(\frac{\sigma_D}{45}\right)^2 + \left(\frac{\sigma_{A_z}}{4}\right)^2 + \left(\frac{\sigma_E}{4}\right)^2 + \left(\frac{\_C}{1.25}\right)^2 < 0.5^2$$

$$\_C^{''} = 1.25\,\text{sec}, \sigma_D > 0(m), \sigma_{A_z} > 0(\text{min}), \sigma_E > 0(\text{min}), \_C > 0(\text{sec})$$

The given result is confirmed in 3300 control process realizations.

The above-mentioned results were obtained for the limits:

$|\omega_x|$ < 10 deg/sec and $|dV_i/dt|$ < 2>5 (km/h)/sec

When extending maneuvering capacity on the terminal lag (roll rate $|\omega_x|$ < 50 deg/sec and equivalent airspeed variation $|dV_i/dt|$ < 9 (km/h)/sec), the area of admissible characteristics increased:

$$\left(\frac{\sigma_D}{70}\right)^2 + \left(\frac{\sigma_{A_z}}{8}\right)^2 + \left(\frac{\sigma_E}{8}\right)^2 + \left(\frac{\_C}{2.25}\right)^2 < 0.5^2$$

$$\sigma_D^{''} = 70m, \sigma_{A_z}^{''} = 8', \sigma_E^{''} = 8', \_C^{''} = 2.25\,\text{sec}$$

$$\sigma_D > 0(m), \sigma_{A_z} > 0(\text{min}), \sigma_E > 0(\text{min}), \_C > 0(\text{sec})$$

This results are supported by 1100 control process realizations.

Added to step-type behaviour of measurements as a disturbing factor, the equation took the form:

$$\left(\frac{\sigma_D}{70}\right)^2 + \left(\frac{\sigma_{A_z}}{8}\right)^2 + \left(\frac{\sigma_E}{8}\right)^2 + \left(\frac{\_C}{2.25}\right)^2 + \left(\frac{\_t}{4.25}\right)^2 < 0.2^2$$

$$\sigma_D > 0(m), \sigma_{A_z} > 0(\min), \sigma_E > 0(\min), \_C > 0(\sec), \_t > 0 \sec$$

For more stringent limitations on roll and equivalent airspeed the equation of allowable errors of the enumerated factors is of the form:

$$\left(\frac{\sigma_D}{45}\right)^2 + \left(\frac{\sigma_{A_z}}{4}\right)^2 + \left(\frac{\sigma_E}{4}\right)^2 + \left(\frac{\_C}{1.25}\right)^2 + \left(\frac{\_t}{1.9}\right)^2 < 0.25^2$$

$$\sigma_D > 0(m), \sigma_{A_z} > 0(\min), \sigma_E > 0(\min), \_C > 0(\sec), \_t > 0 \sec$$

This result is attested to by 600 control process realizations.

So, any parameter to be numerically simulated can be added in the disturbance area in a similar manner.

The proposed method allows with reasonable facility to construct the area of admissible values for any set and number of parameters to be mathematically simulated. Using the area of disturbing parameter admissible value will enable a system developer to balance the level of some disturbances by the others.