1

AGARD-CP-452

AGARD-CP-452

AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92200 NEUILLY SUR SEINE FRANCE

AGARD CONFERENCE PROCEEDINGS No.452

Flight Test Techniques

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY
ON BACK COVER

AGARD-CP-452

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.452

FLIGHT TEST TECHNIQUES

Copies of papers presented at the Flight Mechanics Panel Symposium held at Edwards Air Force Base, California, USA, 17—20 October 1988

THE MISSION OF AGARD

According to its Charter, the mission of AGARD is to bring together the leading personalities of the NATO nations in the fields of science and technology relating to aerospace for the following purposes:

- Recommending effective ways for the member nations to use their research and development capabilities for the common benefit of the NATO community;
- Providing scientific and technical advice and assistance to the Military Committee in the field of aerospace research and development (with particular regard to its military application);
- Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;
- Improving the co-operation among member nations in aerospace research and development;
- Exchange of scientific and technical information;
- Providing assistance to member nations for the purpose of increasing their scientific and technical potential;
- Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field.

The highest authority within AGARD is the National Delegates Board consisting of officially appointed senior representatives from each member nation. The mission of AGARD is carried out through the Panels which are composed of experts appointed by the National Delegates, the Consultant and Exchange Programme and the Aerospace Applications Studies Programme. The results of AGARD work are reported to the member nations and the NATO Authorities through the AGARD series of publications of which this is one.

Participation in AGARD activities is by invitation only and is normally limited to citizens of the NATO nations.

The content of this publication has been reproduced directly from material supplied by AGARD or the authors.

Published July 1989

ISBN 92-835-0509-3

*Printed by Specialised Printing Services Limited
40 Chigwell Lane, Loughton, Essex IG10 3TZ*

PREFACE

These are exciting times for the aircraft test and evaluation community. Over the past 15 years, aircraft flight test emphasis has shifted from airworthiness and aerodynamics testing to avionics subsystem test and integration. Aircraft systems and aircraft test programmes havebecome highly integrated, increasing the technical and management challenges. The advances in weapons systems technology have had significant impacts on the test process as well as on the testability of our systems. Today's systems require new and innovative technical and management approaches. There is a need for greater use of simulators and other hardware-in-the-loop ground test facilities to accelerate the integration and checkout of software-intensive systems.

The Symposium addressed these critical isues; starting with overviews of a variety of military and commercial test programmes. The latest test methodologies for light dynamics and systems testing were reviewed. A report was made on a systems approach to flight test safety, with emphsis on payoffs to be realised in test safety. Presentations were given on state-of-the-art test instrumentation and facilities usedn support of flight tests. Innovative applications of information processing and display technologies to the real-time test decision-making process were reviewed.

The Symposium was structured to give attendes the opportunity to see and touch as well as hear about the latest in test techniques. Each day's agenda included tours and briefings on U.S. Air Force NASA test aircraft and facilities, and the Symposium closed with an airshow. There was consesus on the importance of the AGARD flight test community meeting regularly to review new techniques for flight test, instrmentation and data analysis to ensure that safe, efficient and timely testing is accomplished.

* * *

La communauté d'essais et d'évaluation aéronautiques vit une époque passionnante. Au cours des quinze dernières années l'aptitude au vol et les essais aérodynamiques ont dû céder la place à l'intégration et aux essais des sous-systèmes avioniques qui est devenu le nouveau centre d'intérêt des essais en vol. Les systèmes avion et les programmes d'essai aéronautiques sont désormais hautement intégrés et, par consequent présentent de nouveaux défis techniques et technocratiques. Les progrès réalisés dans le domaine de la tecnologie des systèmes d'armes ont eu un impact considérable sur la modalité des essais, ainsi que sur l'aptitude aux essais de nos systèmes. Les systèmes modernes appellent de nouvelles philosophies innovatrices sur le plan technique comme sur le plan du management. L'emploi plus intensif de simulateurs et d'autres moyens d'essai au sol à base de matériel informatique est demandé ain d'accélérer l'intégration et la mise au point des systèmes à forte composante logicielle.

Le Symposium a examiné ces questions de première importance, en commençant par un tour d'horizon des différents programmes d'essai civils et militaires. Les dernières méthodologies d'essai en dynamique du vol et en systèmes ont été revues. Un rapport a été presenté sur une méthode "systèmes" destinée à améliorer la sécurité des essais en vol. Des communications sur l'état de l'art de l'instrumentation d'essai et des servitudes utilisées ont été présentées. Des applications orginales des technologies de l'informatique et de la visualisation au processus de la prise décision en essais temps réel ont également été examinées.

Le Symposium a été structuré de façon à offrir aux participants l'occasion de voir de manipuler les appareils de test, ainsi que de s'informer sur les dernières techniques d'essai. Des visites et des briefings les avions d'essai et les installations de l'US Air Force NASA ont été organisés chaque jour. Les activités du Symposium ont terminées par une fête aérienne. Tous les participants s'accordaient à reconnaitre l'importance pour la communauté d'essais en vol de se réunir régulièrement afin d'examiner les nouvelles techniques d'a... essais en vol, d'instrumentation et des données, permettant de réaliser les essais en temps voulu, de façon ... dans les meilleures conditions de sécurité.

FLIGHT MECHANICS PANEL OFFICERS

Chairman: Dr Ing. A.Filisetti
Director
Combat Aircraft Group
Aeritalia
10146 Turin
Italy

Deputy Chairman: Mr R.C.A'Harrah
O.C.N.R.
Code OOJDL
800 N Quincy Street
BCT/1
Virginia 22217-5000
USA

TECHNICAL PROGRAMME COMMITTEE

Mr C.E.Adolph
Deputy Director
Defense R&E
(Test and Evaluation)
PSD, DDR&E
The Pentagon
Washington DC 20301-3110
USA

Mr J.Renaudie
Director
A320 Programme
STPA
26 Boulevard Victor
75996 Paris Armées
France

LOCAL COORDINATOR

Mr R.Hildebrand
Technical Director AFFTC/CA
Edwards Air Force Base
CA 93523-5000
USA

ACKNOWLEDGEMENT

The Flight Mechanics Panel wishes to express its thanks to the US National Delegates to AGARD for the invitation to hold this meeting at Edwards Air Force Base in California, USA and for the facilities and personnel which made this meeting possible.

Le Panel du Mécanique de Vol tient à remercier les Délégués Nationaux des Etats Unis près de l'AGARD de leur invitation à tenir cette réunion sur la Base Aérienne 'Edwards' en California, USA.

CONTENTS

* Not available at time of printing.

U.S. NAVY PRINCIPAL SITE TESTING CONCEPT AND THE F-18

by
Joe L. Dunn*
Naval Air Test Center
Patuxent River, Maryland 20670-5304
USA

ABSTRACT

In 1975, a decision was made by the Naval Air Systems Command (NAVAIRSYSCOM) to conduct the F-18 Full Scale Development (FSD) Program at a primary Navy location, the U.S. Naval Air Test Center (NAVAIRTESTCEN). Previous FSD programs had utilized multiple test locations (contractor and Navy) which resulted in significant program duplication from a facilities, logistics, and test data viewpoint. The principal site concept provides for a primary Navy location where all test assets are co-located. These assets include developmental aircraft, contractor and Navy test personnel, maintenance personnel, and all unique test equipment. The success of the F-18 principal site testing program paved the way for the AV-8B and LAMPS MK III programs at the NAVAIRTESTCEN. The principal site concept has had excellent benefits for the U.S. Navy in terms of improved test aircraft utilization, better visibility into the contractor's test program, elimination of redundant testing by utilizing a common data base, and improved utilization of government test facilities.

From the Navy's T&E community viewpoint, it is preferable to conduct the programs at a Navy test activity. Due to program funding constraints, this may not always be possible. With the F-14D and A-6F upgrade programs, the testing is being conducted at the contractor's facility as opposed to a Navy facility. In this case, the Navy has established a Navy Test Team on-site and is participating actively with the contractor during the FSD programs.

The impact of the principal site testing concept will be examined from the prospective of both the Navy and the contractor in terms of impact on the individual organizations.

Introduction

In the 1975 time frame, the U.S. Navy made the decision to principal site the F/A-18 Full Scale Development (FSD) program at the U.S. Naval Air Test Center, Patuxent River, Maryland. This represented a significant departure from the earlier philosophy of allowing the prime contractor to conduct the majority of the FSD test program at his own flight test facility with minimal involvement by the U.S. Navy test activities. With the F-14A FSD test program, which was conducted during the 1971-1974 time period, Navy involvement was limited basically to periodic evaluations (Navy Preliminary Evaluations) by the Naval Air Test Center at the contractor's facilities. These evaluations consisted of 15-25 test flights flown at approximately 6-9 month intervals during the FSD program. The F-14A test program was conducted simultaneously at three separate test sites. The flying qualities, high angle-of-attack, propulsion, performance, structural dynamics and structural loads program was conducted at the contractor's facility in New York. The carrier suitability test program was conducted at the Naval Air Test Center where catapult and arresting gear facilities are located, and the avionics and weapons integration program was conducted on the West Coast at the Pacific Missile Test Center, Pt. Mugu, California. The three test sites resulted in a large support infrastructure in terms of data reduction facilities, logistics support systems with lengthy pipelines, three separate flight test support organizations, and a large number of test assets.

Due to the complexities of the F-14A test program, the decision was made to principal site the F/A-18 at Patuxent River. Considerations that went into this decision process centered around the premise that if all the FSD testing was conducted primarily at one location, the number of test aircraft required could be reduced, the logistics and maintenance support requirements could be drastically reduced, both the contractor and the Navy could jointly share all data which would result in elimination of much redundant testing between the contractor and the Navy, and also the Navy in the role of the customer would have much better visibility into the overall progress of the contractor's test program. The view was held that by having better visibility into the contractor's test program, the Navy could do a better job of identifying deficiencies and getting them corrected early in the program. It should be noted that during this time period, most FSD contracts being awarded by the U.S. Navy were on a Cost Plus Incentive Fee Basis.

*Program Manager for Advanced Tactical Aircraft Program.

Another major consideration in the 1975 time period centered around improved utilization of existing Department of Defense test facilities. Given the large number of airframe manufacturers that the U.S. Navy buys aircraft from, it is impossible and unreasonable to expect the U.S. Government to underwrite the cost of unique test and evaluation facilities at each of the contractor's facilities. The argument was made that it made more sense and was most cost efficient to invest in state-of-the-art test and evaluation facilities at Department of Defense facilities and require the contractor's to utilize these facilities as opposed to each developing their own. These facilities typically include telemetry tracking systems, space positioning systems, real-time data computation systems, engine test facilities, avionics test laboratories, weapons firing ranges, and other associated infrastructure, to mention a few.

F-18 Full Scale Development Program

Immediately after the decision by the Commander, Naval Air Systems Command in 1975 to principal site the F-18 FSD program at Patuxent River, joint planning was begun by the Navy and McDonnell Aircraft Company as to how best to support and execute the program. The FSD program consisted of 11 FSD aircraft with first flight occurring at the McDonnell facility in St. Louis in November of 1978. After limited expansion of the flight envelope in St. Louis, the aircraft was ferried to Patuxent River in January 1979. Existing hangar and office facilities at Patuxent River were modified to meet the needs of McDonnell. At the peak of the program, over 600 McDonnell employees (including many local hires) were involved in the support of the FSD program. The Naval Air Test Center provided the majority of support effort for the program. This effort included telemetry tracking, range support consisting of space positioning, radar coverage, video coverage, target support, chase and target aircraft support, tanker aircraft support, search and rescue aircraft support, and extensive laboratory services support. During the period from November 1978 to February 1982, a total of 3,205 flights were flown for 4,799 flight hours. Extensive use of the Naval Air Test Center Real-Time Telemetry Processing System during the FSD program contributed to quick turnaround of the data. Nine of the 11 test aircraft were fully instrumented.

The F/A-18 Principal Site test program resulted in all the participants being co-located at one activity. This included McDonnell, General Electric, and Northrop from the contractor side. The Navy team included representatives from the Naval Air Test Center, Pacific Missile Test center, Naval Weapons Center, Board of Inspection and Survey, and the Operational Test and Evaluation Forces. The co-location of personnel allowed for good face-to-face communications, sharing of data, and a building of an overall team spirit. A series of Memoranda of Agreement (MOA) was established in writing to define the duties and responsibilities of all involved parties. These MOA's spelled out the various relationships necessary to accomplish FSD testing in a smooth and efficient manner. For example, the MOA with MCAIR spelled out to what extent the Navy could be present during MCAIR flight briefings and debriefs and to what level the Navy had access to contractor data during the development phase of the program. These MOA's also provided a means of continuity in procedures for personnel rotating into the test program.

Navy Test Philosophy

The Navy has now completed three major FSD programs (i.e., F/A-18, AV-8B, and SH-60B) utilizing the principal site test concept at government facilities. From the Navy's viewpoint, some of the major advantages to this concept have included:

a. High utilization of existing DOD facilities.

b. Reduced number of test assets required.

c. Significantly improved customer visibility into the contractor development program.

d. Early identification and correction of design problems.

e. Establishment of common contractor-Navy data base.

f. Elimination of redundant testing.

Even though the testing is accomplished primarily at a Navy facility, the Navy has been very careful to recognize the contractual authority of the contractor to be responsible for the successful development of the aircraft. In doing this, the Navy recognizes the need for the contractor to be in charge of the day-to-day control of his operation and to be able to make programmatic decisions accordingly. Although the contractor has control, experience on the F/A-18, AV-8B, and SH-60B has shown that it is important to involve the customer (i.e., U.S. Navy) early in the development effort. This involvement may include Navy flight test engineers attending the contractor's preflight and postflight briefs, monitoring the flight at the telemetry ground station, and having qualified Navy test pilots flying the chase aircraft for each of the test flights. The next step that has evolved is the limited participation of Navy test pilot into the contractor's flight program.

The amount of Navy pilot participation is determined in general terms prior to the start of the actual flight test program. This involvement is often referred to as the participatory approach. Based on our experience, this type of participation gives us a good insight into the development program, allows for early identification and correction of design deficiencies, and maintains flight proficiency in the test aircraft for a limited number of test pilots.

By maintaining this active involvement in the contractor's test program, the Navy has been able to eliminate a significant amount of dedicated testing that was accomplished under the previous way of doing business which allowed for blocks of dedicated Navy test time at specific intervals during the test program. The amount of contractor generated data utilized to satisfy Navy requirements has increased greatly under the participatory approach.

Under the previous way of accomplishing FSD programs, the contractor would conduct his envelope expansion to the 100% build-up point and would then repeat the point during a series of formal demonstrations which were witnessed by U.S. Navy engineers. Under the principal site concept, the Navy engineers will accept the completion of the 100% build-up point as meeting the intent of the formal demonstration if all the necessary specification conditions are satisfied.

Although the term "principal site" is used extensively, the Navy's position is that it will go off site as necessary to take advantage of other existing Department of Defense test facilities. For example, climatic testing is accomplished at the Climatic Test Facility at Eglin Air Force Base and cross-wing landing tests are conducted at Edwards Air Force Base. This ensures best utilization of existing Department of Defense facilities and keeps the facilitization cost for the program manager down to a minimum.

Navy/Contractor Perspective

There has been much discussion during the last 10 years concerning the overall benefits of principal site testing. From the Navy's perspective the concept has provided for best use of limited resources, resulted in the development of state-of-the-art test and evaluation facilities available to all contractors, increased productivity in terms of flight rate, eliminated much redundant testing between the contractor and the Navy, provided for a single set of facts to be utilized by all participants, and has provided up-front customer participation resulting in much better overall Navy visibility into the development progress.

Not all contractors share the Navy's perspective on the benefits of principal site testing. Some of the comments expressed by the contractor community when forced to utilize government test and evaluation facilities include concern over potential decreased productivity, longer concept-to-operation cycles, dislike of the military being too deeply involved, concern over sharing common facilities in terms of scheduling conflicts and lack of control over support resources, and too many government rules and regulations.

It is up to the Navy to convince the contractors that these concerns are not valid and that the Navy test and evaluation personnel are sensitive to their schedule and resource requirements from a contractual viewpoint. In terms of FSD program productivity, the F/A-18 FSD conducted at Patuxent River achieved the highest flight rate (approximately 17.5 hr/month/aircraft) of any recent Department of Defense tactical aircraft development program.

Measurements of Success

Although it is difficult to quantify absolute measures of success, it is the Navy's position that the principal site test philosophy has significantly contributed to the outstanding success presently being enjoyed by the F/A-18, AV-8B, and SH-60B in the fleet today. This success can be measured by the following factors:

a. Fleet Acceptance - All three weapon systems have been extremely well received by fleet users with a high degree of satisfaction.

b. Reliability and Maintainability We are seeing some of the best R&M statistics with these systems in the history of Naval aviation. These statistics translate directly into flight hour availability to the fleet commander.

c. Safety Record - These aircraft are enjoying a safety record unparalleled by any previous comparable weapon system. The attrition rate of each of these aircraft is running well below the predicted loss rate.

d. Configuration Changes - Due to the high state of maturity of these aircraft at the time of fleet introduction, the need for major engineering changes and updates have been minimal when compared to their predecessors. This translates into significant dollar savings in terms of life cycle costs to the Navy.

Impact of New Navy Acquisition Strategy

Recently revised Navy acquisition policy states that Full-Scale Engineering Development will be conducted under a firm fixed-price contract. Recent programs in which this policy has been implemented are the V-22, F-14D, A-6F, and T-45 programs. This is in contrast to the cost plus incentive fee type contract used on previous FSD programs. Because the contractor is in the position of having to accept a greater financial risk, the Navy has in turn given the contractors more latitude in determining how and where the FSD programs will be conducted. In the case of the V-22, F-14D, and A-6F programs, the contractors have elected to conduct the majority of their test programs at their home facilities. Navy facilities will be utilized for testing for which the contractor does not have the necessary facilities such as carrier suitability and missile firing testing. The Navy test activities will support the FSD programs at the contractor location by locating Navy test teams at the contractor's facilities during the FSD program to preserve as much Navy involvement in the programs as possible. It is the Navy's intent to maintain the same type of involvement and participation at the contractor's facilities that it would at Navy test and evaluation activities. The Navy position is that regardless of test site location, we will maintain a full time involvement throughout the FSD programs. This will include on-site monitoring by Navy engineers and participatory flying by Navy test pilots. This will be accomplished fully recognizing the contractor's responsibility to develop the new aircraft and to control the day-to-day decision making process during the FSD program.

In summary, the test site location decision process will be made based on economics, projected test productivity, available support facilities, flight safety considerations, and technical capability.

Summary

From the Navy's viewpoint, the principal site has been successfully proven with the F/A-18, AV-8B, and SH-60B FSD programs. The concept has helped to improve utilization of DOD test facilities by reducing facilitization of contractor facilities at government expense, has provided improved customer visibility into contractor test programs, and has provided a better product to the fleet in terms of high fleet acceptance, improved reliability and maintainability, improved safety records, and minimal in-service configuration changes.

ATTAS FLIGHT TESTING EXPERIENCES

by
H.-H. Lange, D. Rohlf, A. Zach, H.-L. Meyer *
Institute for Flight Mechanics
* Flight Operations
Deutsche Forschungs- und Versuchsanstalt
für Luft- und Raumfahrt e.V.
Flughafen, D-3300 Braunschweig, FRG

SUMMARY

An overview of recent development and flight test experiences of the DFVLR's flight test vehicle ATTAS (Advanced Technologies Testing Aircraft System) equipped with a digital fly-by-wire/light flight control system is presented. System design, multiprocessor communication management, parallel data processing, redundancy management as well as software development and validation are summarized.

Further, the role of ground based system simulation for development and testing, flight test procedures and interesting flight test results are dealt with in several examples.

1. INTRODUCTION

In the last six years a modern flying simulator and demonstrator aircraft called ATTAS (Advanced Technologies Testing Aircraft System) has been developed by DFVLR and MBB supported by the Ministry of Research and Technology of Germany [1]. ATTAS is based on a MBB developed twin-turbofan, short haul passenger aircraft VFW 614 (figure 1), which is ideally suited for this purpose due to it spacious cabin, test equipment loading capability, flight performances and excellent handling qualities.

Over the next 15 to 20 years ATTAS will serve as the primary DFVLR flight test vehicle for research and development to demonstrate and validate new methods and technologies in the area of flight control, flight guidance, navigation, man-machine interactions and in-flight simulation (figure 2) [2,3].

In addition to method oriented research application ATTAS provides a wide integration and testing capability for aircraft equipment as it is summarized in figure 3. Beside ground based simulation and advanced computation methods flying testbeds for in-flight simulation techniques have increasingly gained attention because the overall pilot/aircraft system can be investigated under real environmental conditions [4,5].

Within the DFVLR research programs ATTAS will mainly be used as flying simulator [6] in a broad sense. In this role ATTAS is able to represent the dynamic behavior of model aircraft or systems under real environmental conditions in total missions providing the pilot with exact visual and motion cues in an early stage of a development. To fulfillall these testing capabilities ATTAS was modified and equipped with a powerful fly-by-wire/light flight control system.

is system, designed by DFVLR, fulfills also the requirements to be very easily adapted to changing flight tests and to give the experimenter clear and simple to handle software and hardware interfaces [7].

The fly-by-wire/light flight control system (FBW-system) development, integration and flight test experiences will be dealt with in this paper.

2. Aircraft Modifications and Equipment

The main aircraft modifications, test equipment [8] and features are summarized as follows (see also figure 4):

- right hand seat safety pilot with conventional control system,
- left hand seat evaluation pilot with full axis fly-by-wire controls,
- freely programmable flight instruments/displays, CRT's (figure 5),
- fly-by-wire controls/column or sidestick with adjustable force feel system,
- dual channel digital on board computer system with fiber optic data bus providing freely programmable control laws and flying qualities,
- duplex inertial reference and digital air data systems,
- comprehensive on board data acquisition system, recording and PCM-telemetry,
- 15 electro-hydraulic self monitored actuators, partly duplex linked by MIL-BUS 1553B to the FBW-system,

- antennas installation provisions,
- dual redundant hydraulic system,
- dual redundant electrical system,
- fly-by-wire actuators for
  - o elevator,
  - o stabilizer,
  - o rudder,
  - o both ailerons (also with symmetrical deflection capability),
  - o both engines,
  - o landing flaps,
  - o six direct lift flaps,
- *on board operator consoles (four places),*
- nose boom with α, β- and TAS- probe.

2.1 Additional Control Capability

To give ATTAS a 5-DOF simulation capability five independent control surfaces must be available. Therefore, ATTAS was equipped with a *specifically developed 'Direct Lift* Control' System (DLC) for pitch/heave motion decoupling and gust/load control. For low frequency DLC operation the basic VFW 614 landing flap system can electrically be deflected between 1 to 14 degrees. The rear part of the landing flaps have been divided in six (three on each wing) fast moving flaps having about 85 deg/sec flap rate and ±35 degrees flap deflection capability for high frequency direct lift modulation. Both lift devices can simultaneously be used between 1 to 14 degrees landing flap position. Further, DLC flap pairs can be individually controlled. The permissible flight envelope is also shown in figure 17 demonstrating that DLC can also be used in the high speed region up to 285 kts, compare 3.4.

Another important feature of the ATTAS FBW-system is the superimposed symmetrical aileron actuation capability which will be used for wing bending mode control. Because the ailerons are mechanically connected in the basic aircraft's flight control system, the symmetrical aileron deflections, "flaperons", are compensated on pilot's wheel by using a differential gear.

2.2 Data Aquisition System

The aircraft is equipped with all sensors which are necessary to measure the aircraft body rates, accelerations and attitudes as well as all control surface positions, pilot command inputs and engine data. Air data are calculated by two air data computers, inertial data by two *laser gyro inertial reference* units (LTN 90). Analog sensor outputs are conditioned (amplified, filtered etc.) in a DFVLR developed signal conditioning system [9,10]. A very important feature of this system is that all parameters for each channel can be set, checked, and electrically calibrated from an on board master computer (figure 6).

2.3 Electro-Hydraulic Actuators

The electro-hydraulic actuators have been developed for ATTAS by Liebherr Aerotechnik, LAT, in Germany.

In total 15 self-monitored high bandwidth actuators are used to drive all the actuators mentioned above.

The actuators are installed in parallel to the basic aircraft control system. They are linked by integrated electro-hydraulic actuated clutches which can be opened in any situation by pressing a switch on the safety pilots control wheel or by introducing a certain amount of force in safety pilot's controls.

The actuator electronic is fully digital using duo-duplex microprocessor configuration for failure detection and failure handling. The complete actuator electronic is included in four boxes. Data communication with the FBW/Light computer system is realized by a duplex MIL-BUS 1553B. Due to safety reasons elevator and rudder actuators are doubled providing duo-duplex failure behavior. For DLC control surface redundancy was applied by using three independent pairs of DLC flaps which are in addition monitored by a separate device as it is shown in figure 7.

2 Fly-By-Wire/Light Control System

[illegible] *Fly-by-Wire/Light Flight Control System as heart of ATTAS* has to provide all [illegible]onal functions needed for

- normal fly-by-wire operation (control laws, mode switching),
- interfacing external devices,
- data processing and recording,
- monitoring and error detection,
- computation of experimental functions.

The performance has to meet the requirements for

- computational cycle time of less than 20 ms for all functions,
- required redundancy for flights in critical maneuvers (take off and landing etc.),
- airborne equipment and interfacing of aircraft systems (ARINC 429, MIL BUS 1553 B, etc.),
- freely programmable capacities for user applications in a high order programming language.

To meet these requirements the system has been designed as a two channel computer network consisting of four processors in each channel with one common central processor for communications and data recording (figure 8). All on board computers are of MIL-Spec. LORAL/ROLM types (MSE/14 and Hawk/32) which are software compatible to the commercial Data General Eclipse S/140 and MV/8000 series [11].

The network in each channel is based on a ring structured serial fiber optical bus system providing actual data rate in each channel of 150 kWords/sec. Network redundancy is used for failure detection by comparing exchanged input and output data. In the case of exceeding given amplitudes and time thresholds the system will be passivated and enabling the safety pilot to convert to the basic aircraft control by providing smooth transients. Software is identical in both channels.

The data processing system operates A/D converters and digital inputs and outputs, interfacing the signal conditioning system, the ARINC 429 BUS connected to the air data computer, inertial reference and avionic systems, as well as MIL BUS 1553B for data exchange with the actuator electronic units (AEU). In total more than 400 input and 300 output signals are processed in each channel.

The computational power of the system is about 3.1 Mega Floating Point Operations (MFLOPS) in each channel. 1.4 MFLOPS are available for experimental functions in the Hawk/32 32-bit computer with up to 8 MB of memory.

System and software design allow an easy system integration of experimental functions and additional hardware components.

2.5 Fly-By-Wire Control Functions

Operation of ATTAS is performed in the following three principal modes, as illustrated in figure 9:

- BASIC Mode
- Fly-By-Wire Mode (FBW-Mode)
- Simulation Mode (SIM-Mode)

In the BASIC Mode ATTAS is operated using the basic mechanical controls of the safety pilot on the right hand seat in the cockpit.

In this mode of operation ATTAS is used for all the experiments in which electrical flight control is not required. Nevertheless on board computation, data aquisition and data recording capability is fully available.

In the FBW-Mode the evaluation pilot on the left hand seat has control to the FBW-system by connecting the electro-hydraulic actuators to the basic aircraft control system. Flight operation will be performed identically to the basic controls (FBW 1:1 function). Figure 10 illustrates the FBW control functions installed for the elevator as an example.

In the SIM-Mode the aircraft also operates under fly-by-wire control but the functional connection between pilot inputs and actuator commands are given by user defined functions such as used for in-flight simulation control laws (IFS) where the evaluation pilot flies an aircraft model built in the computer program. Model following control laws generate commands for all the actuators to force the test aircraft to follow the on board computed equations of motion.

Primarily the fly-by-wire system provide control functions for the actuators of the primary controls
- elevator,
- ailerons and
- rudder,

and the secondary controls
- stabilizer,
- engines,
- landing flaps and
- DLC flaps.

Further, special functions for mode switching and for minimizing transient effects based on mode switching are provided.

For electro-hydraulic actuator engagement a coincidence function moves the actuator piston into the relevant position before clutches are closed.

Limitation functions reduce the actuator command signals in maximum rates and amplitudes according to safety and flight conditions.

An automatic elevator trim system operates the stabilizer in such a way that the aircraft is always in trim condition. In cases where the safety pilot reconverts to the basic control system the resulting column and aircraft transients will be negligible.

Evaluation pilot's displays and instruments are controlled by the FBW-system to represent standard symbology (FBW-Mode). In the SIM-Mode display representation is freely programmable due to user purposes. For system checkout and preflight check procedures special functions are included in the FBW/Light software program.

2.6 ATTAS Simulator

Essential part of the total ATTAS is the ground based ATTAS simulator (figure 11) which was designed and developed in parallel to the aircraft. The ATTAS simulator is a complete copy of the flying system simulating all system functions in real-time on the ground. This facility is the primary tool for software and functional testing.

Further, the simulator assists the ATTAS operation by validating user applied software (experiments) and by preparing flight test procedures. The ground simulation consists of an identical ATTAS cockpit, a nearly identical computer configuration based on commercial Data General computers which are software compatible to the LORAL ROLM on board system, a hybrid computer EAI 600 and the high speed multiprocessor system AD 10. These both computers simulate the complete aircraft flight dynamics, actuator response as well as engine and sensor informations of the aircraft. All sensor data are connected to the ground based FBW-system via connector identical interfaces (A/D, D/A, ARINC 429, MIL-BUS 1553B), so that flight test hardware can be plugged into the simulation. By this all in-flight computer programs can be developed and checked out on the ground under real time conditions before they are transferred via magnetic tape on board the aircraft. Further, for 'hardware in the loop' simulation and testing purposes ATTAS can be linked with the ground simulator via fibre optic bus extension while standing on the ramp or in the hangar. This situation is illustrated in figure 12.

2.7 Software Development and Data Handling

According to the ATTAS equipment with a central FBW-system software development was a main project of about 60 man years at DFVLR [12]. Software development was organized in a 'top down' stepwise method and had to be considered for hardware and software together. So from the beginning of the system design software functions were analyzed and designed in parallel to the hardware design in progress. With the structure of the data processing system the functional structure of the software was defined and resulted in the Software Requirements Document (SRD).

Mainly the involved subsystems and their interfaces (ARINC, MIL BUS 1553B) specified the requirements to be fulfilled by the software. Also user requirements for minimum controller cycle time and computing power with reusable software had to be considered.

2.7.1 Software Structure

According to the design of the data processing system three main software functions can be identified to be embedded in the computer network:

- Input/Output Operations (Terminal Computer Functions)
- Fly-By-Wire and Experimental Functions
- Communication Functions with different subsystems

Input/Output Operations are installed in the Cockpit and Tail Terminal Computer (CTC, TTC). They include the interface drivers, data checks with the second channel, scaling and data communication to the parallel processors in the network. Synchronization of all components is done with a Programmable Interval Timer (PIT) interrupt in the cockpit terminal computer program.

Fly-By-Wire Functions are mainly installed in the FBW Computer (FBWC) to fulfil all operations of the FBW control law. The software structure has to consider timing requirements and switching problems as well as special monitoring functions.

User functions are installed in the Experimental and Control Computer (ECC) to give free space and full performance for experimental programs. With the chosen program structure user programs can be included without changes in the operating system and the runtime structure.

Communication Functions include the interfacing to the telemetry system, data recording on magnetic tape, quicklook presentation on displays, program loading and offline handling. Functions, being not essential for fail passive behavior of the FBW control system, are installed in the single Central Communication Computer (CCC) which communicates with all computers in both channels in the duplex data processing system.

2.7.2 Software Development, Integration and Validation

Handling the ATTAS software project as a medium size project and the requirement to maintain the system for at least 10 to 15 years special procedures and tools were mandatory.

The TOP DOWN design method was used in 6 steps. Each step ended with a review on the final documents:

(1) Software Requirements Document
(2) Software Design (rough) Document
(3) Software Detailed Design Document
(4) Coding and Modul Testing Source Code
(5) Integration and Verification - Program Test Results
(6) Validation (Ground and Flight Test) - System Test Results

Test requirements had to be developed during the software design, giving the opportunity of checking programs in further steps against the requirements.

Software development was done on the ATTAS ground based data processing system using a super mini computer MV/6000 of Data General.

The test concept for ATTAS software development was applied in two phases:

- Modul Test
- Validation

Offline tests are required for any module used *in different programs*. This is fulfilled with test programs to be run on the development system where also the test results are recorded.

The design of the ATTAS ground system followed the idea of testing all software functions of the target system in the ground simulation. Besides the capability of testing interface modules the major advantage is that validation tests can be performed on the ATTAS ground system prior to the on board system.

Test functions generating command inputs are available to be recorded with all data available as measured inputs or program outputs. Identical software test procedures are used in 'firstflight' and 'preflight' tests in the aircraft.

2.7.3 Data Recording and Handling

PCM signal recording and transmission to the ground station with the telemetry system is available. Besides this data recording is done on a digital magnetic tape (800 bpi, 75 inch/sec) attached to the data processing system. The format of data recording is the same as the data format used in the computer system.

Based on the header information a magnetic tape with raw data can be processed in the ground system converting the data into a format to be exploit on the main frame computer system with a standard evaluation program, DIVA - a dialogue interpretation program.

3. Flight Tests

Special test procedures were evaluated for ATTAS according to the basic aircraft maintenance and test instructions. The ATTAS test concept therefore divides into two phases, 'Ground' and 'In-Flight' tests.

Standardized ground tests are performed in periodical intervals or in preflight checks. In-flight tests are normally included in the evaluation program for user flight tests.

3.1 Ground Test Requirements

The ground tests are performed as so called 'Firstflight' and 'Preflight' checks.

For Preflight checks the data processing system is involved in all parts needed for fly-by-wire control functions. The program functions cover all modes and switching conditions to obtain full scale of test conditions. The GTR's - 'Ground Test Requirements' documents - give detailed information how tests have to be performed and how they fit in the basic aircraft checks and operations.

Firstflight checks are also defined in GTR documents organized in different parts to check out single components of the equipment independently. An advantage of using the on board computer system in check out procedures results in data recording and presentation and of course both, hardware and identical software can be checked in connection.

An example for system checkout with software functions generated in the on board computer system is given by the frequency response of the elevator actuator system, figure 13, being exited by a sweep function. The results give a quick overlook how specified performances are met.

3.2 Operational and Certification Aspects

After implementing the data processing system different vendor systems were tested and tuned to reach an operational status. Also experiments for parameter identification and calibration of the measurement equipment were carried out.

ATTAS based on the MBB/VFW 614 aircraft was certified according to FAR 25 including amendment 32 by the German authorities (LBA). Additional certification basis for ATTAS took the modifications into account. ATTAS will always be operated under a preliminary certification. DFVLR is authorized to develop and verify modifications for their own.

The ATTAS safety system is based on the safety pilot and his basic hydraulic mechanical control system. During flight tests it was demonstrated, that the safety pilot in the right hand seat could under any condition and without any problem take over control. This safety concept is referring to the highest authority of the safety pilot. This is assured by limitations in the actuators forces so that in a runaway failure case the most critical resulting flight loads, attitudes and rates do not lead to a flight situation reaching the envelope boundaries. Presently the certification is based on a time delay for the pilots reaction of 2 seconds in fly-by-wire operation above 500 ft GND safety altitude. The simulated hardovers showed typical reaction time of about .5 sec with rather small transients. With the duplex FBW system operation below 500 ft GND shall be tested after more experiences.

3.3 Flight Test Results

The FBW control system acceptance and validation flight test program carried out in 1987 covered the mode switching functions, flight control functions, switching transients, recovery procedures due to actuator hardovers, failure mode evaluation and the fly-by-wire operation in general. Switching and transient functions occurring by switching between FBW- and BASIC-mode operation have been of main interest.

Therefore, the FBW-mode was switched off during various aircraft maneuvers such as steep turn, ' ;h roll rates, pull ups, during landing flap operations with and without the auto st.,.ilizer trim system. An example of the resulting transients during FBW-off switching is shown in figure 14 demonstrating smooth transient response, which was very well accepted by the safety pilot.

The FBW operation was tested throughout the complete flight envelope, which is presently limited to a minimum altitude of 500 ft above ground due to safety reasons.

Aircraft control behavior and control system response was evaluated by pilot induced high frequency inputs. A typical result is shown in figure 15 where the ailerons are excited. Due to the complete measurement of different elements of the control system a detailed analysis is possible. Test results of show precise aileron response at a frequency of about 1.7 Hz and a time delay of about 25 ms.

3.4 System Identification

Special flight tests were carried out to obtain precise information about the complete ATTAS aircraft system. The obtained data were evaluated using a sophisticated DFVLR developed Maximum Likelihood (MC) Identification Procedure for the estimation of parameters in nonlinear systems [13]. Besides the analysis of flight test data special notice is put on the optimization of control inputs and the conduction of the flight test itself to obtain suitable data for identification and save flight test time. The well known 3-2-1-1 input signal (compare with figure 17) is a result of this optimization effort now being generated as standard test function in the FBW system.

3.4.1 Aileron Control System

The electro-hydraulic actuators connected to the control tabs on each side and with mechanical springs and aerodynamic forces the ailerons itself are deflected, where left and right ailerons are still connected via cables, also with the control wheel.

The time histories in figure 16 show outout results of joint evaluation of five single flight tests performed at increasing speeds. The figure shows the comparison of the measured aileron outputs with signals of different mathematical models describing the control system dynamics. For a precise fitting a second order model with time delay and an influence on the characteristics frequency, depending on c..odynamic tab forces.

3.4.2 DLC Flap System

The control input signal was identical to all six independent actuator systems. One DLC flap step response is presented in figure 17 together with the 3-2-1-1 input signal. DLC flap situation with actuator being connected to the landing flap is presented with the flight envelope for DLC flap operations.

For the identification of the DLC flap system, the mean value of the six independent flap deflection angles were calculated. Again different mathematical models were checked, figure 18. Finally best results were obtained with a nonlinear second order system with rate and deflection limitations due to the force limitations of the actuators for structural safety reasons.

3.4.3 DLC Control Effectiveness

The aerodynamic efficiency of a control surface is only identifiable using the entire dynamic response of the aircraft, since the aerodynamic forces and moments are not directly measurable. To determine nonlinear control effectiveness it is necessary to excite the system with inputs of several different amplitudes or with inputs at different reference control surface positions. In figure 19 time histories represent one run for DLC flap operations in different reference positions for vertical acceleration pitch rate and angle of attack, together with the result of a lift coefficient, reaching ± . 35 g vertical acceleration as maximum.

4. CONCLUSIONS

The flight control concept designed and developed for ATTAS proved as suitable and effective to fulfill the requirements of a powerful and flexible testbed for research applications as it was demonstrated in flight.

The design of ATTAS test equipment on board and the operational concept turned out to be suitable for applications at DFVLR. Further, the software development concept and software development system configuration choosen for ATTAS development and operation was very well accepted by software design engineers and proved to be very powerful for applicational programming. Using a high order language for real-time processes was a big help for software validation for experimental programs.

Main improvements for the future are seen in computer aided software development methodology and related tools to improve mainly system/software specification and software maintenance.

Results from the flight tests showed that the required performances could be achieved. Handling the system turned out to be very acceptable for the pilots in use for in-flight simulation. ATTAS proved to be an acceptable testbed for flight control systems and applications for 5-DOF in-flight simulations.

5. References

1. Hamel, P., Krüger, H., "ATTAS - Der neue Erprobungsträger", 4. BMFT- Statusseminar, München, 28.-30. April 1986.

2. Hanke, D., "Der Fliegende Simulator und Technologieträger ATTAS der DFVLR", Luft- und Raumfahrt, Jahrg. 7, Heft 1/86, 1986, S. 4-8.

3. D. Rohlf, D. Schafranek, K. Wilhelm, "Recent Results With ATTAS In-Flight Simulator", AIAA "Flight Simulation Technologies" Conference, Atlanta, GA, 7.-9.9.1988

4. D. Ranz and C.F. Berthe, "Boeing 7J7 Fly-By-Wire Control Development", The Society of Experimental Test Pilots, 31st Symposium Proceedings, 1987 Report, pp. 33-50.

5. R.H. Hoh, "Advances in Flying Qualities", AGARD Lecture Series LS 157, Paper 5.

6. Hanke, D., Wilhelm, K., Meyer, H.-L., "Development and Application of In-Flight Simulator Aircraft for Flying Qualities Research at DFVLR ", IEEE 1986 National Aerospace and Electronics Conference NAECON 1986, Dayton, May 19-23, 1986, Volume 3, pp. 490, ISSN 0547-3578.

7. Hanke, D., Lange, H.-H., "Flight Evaluation of the ATTAS Digital Fly-by-Wire/Light Flight Control System", ICAS-Congress 88-3.6.1, Jerusalem, August 28 - September 2, 1988.

8. Lange, H.-H., Hanke, D., Nietschke, W., "Einsatz eines Rechnerverbundsystems mit optischer Buskommunikation im DFVLR-Flugversuchsträger ATTAS", DGLR Jahrbuch 1986 I, DGLR Jahrestagung 1986, München, 8.-10 Oktober 1986.

9. Zach, A., Gandert, R., "A Progammable Data Aquisition System with Integrated Test and Calibration Facilities", Proceedings of the International Telemetering Conference 1984, pp. 155-172, Las Vegas, USA

10. Kevenoglu, M.F., Klewe, H.-J., "Testing Facility and Procedure of the ATTAS On-Board Data Aquisition System" in Proc. ETC Telemetry Conference, Aix en Provence/Fr., 1987, pp. 389-402.

11. Hanke, D., "Data Processing Concept for the DFVLR Research Aircraft ATTAS", Proceeding of the Third European Rolm Users Group Conference, Braunschweig, May 7-9, 1985.

12. Lange, H.-H., "Software Development under AOS/VS for ARTS Target Systems", Third European Rolm Users Group Conference, Bad Soden, May 7-9, 1985.

13. Rohlf, D., Mönnich, W., "Identification of the DLC-Flap System of the Research Aircraft ATTAS", Proceeding of the 18th Annual Symposium "Flight Testing World Wide", Amsterdam, September 28 - October 2, 1987.

6. Figures

Figure 1. ATTAS in flight

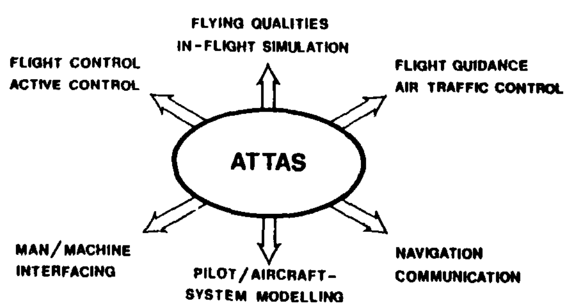


Figure 2. ATTAS R & D utilization

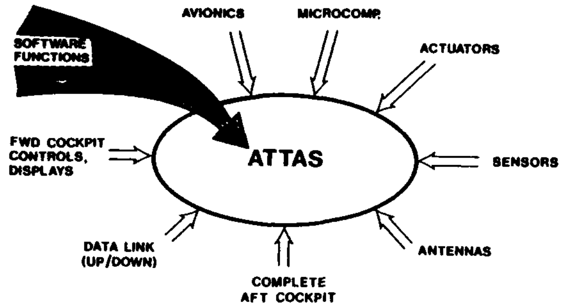


Figure 3. Hard-/Software testing capability

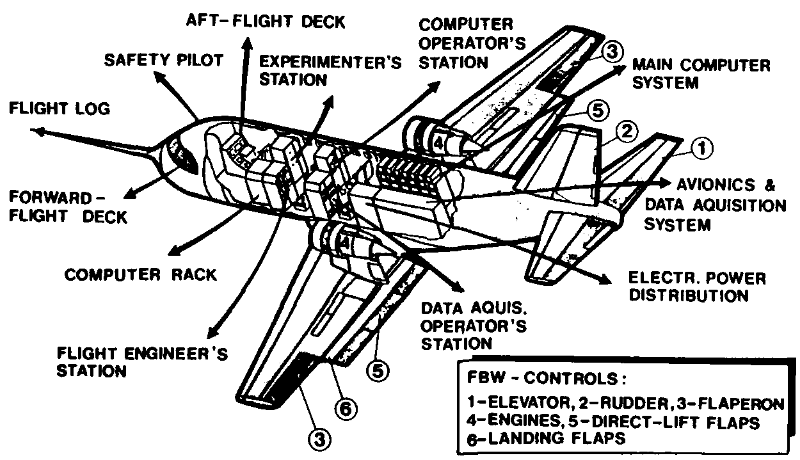


Figure 4. ATTAS modifications

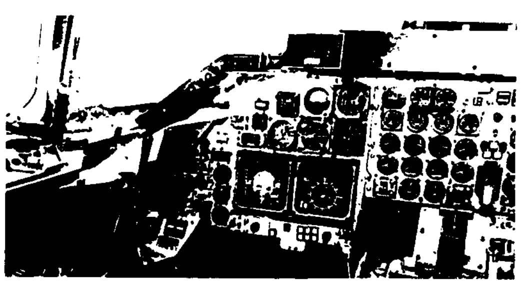


Figure 5. Cockpit instrument panel

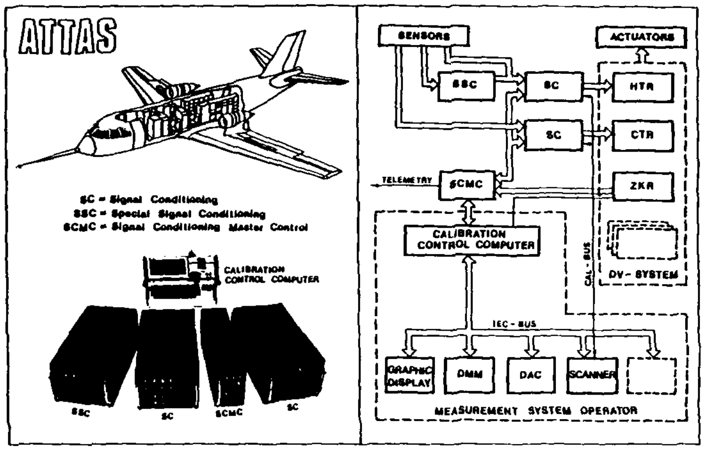


Figure 6. Data aquisition system

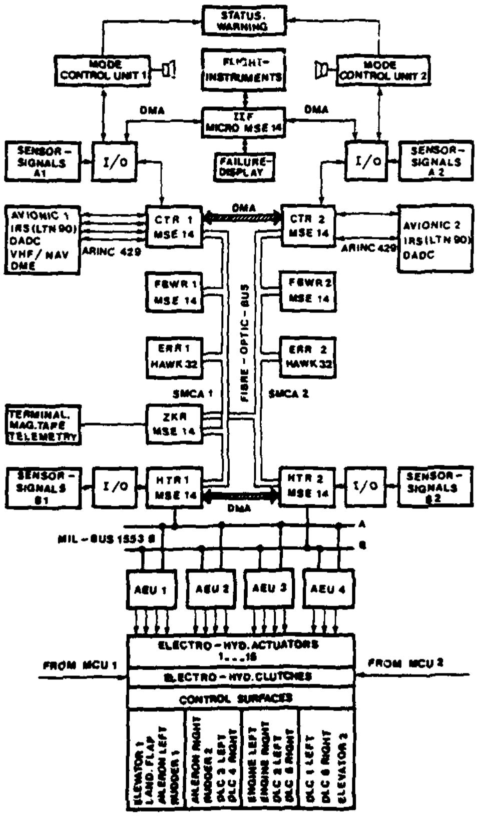


Figure 8. ATTAS dual redundant fly-by- wire/light flight control system

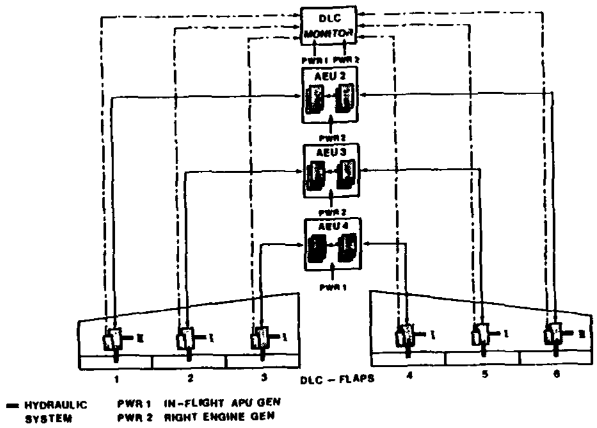


Figure 7. DLC redundancy concept and Actuator electronic units

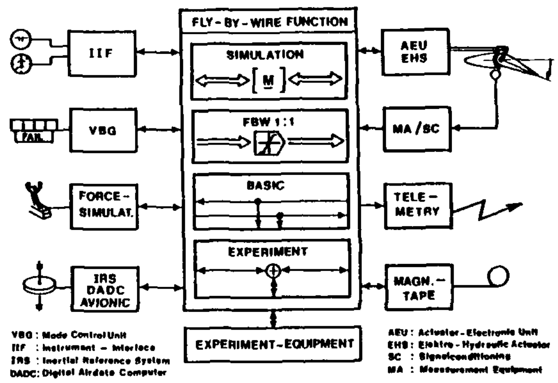


Figure 9. Fly-by-wire control functions

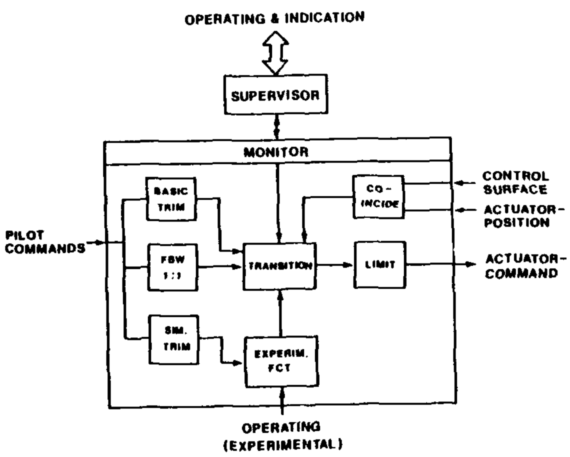


Figure 10. FBW function for elevator control

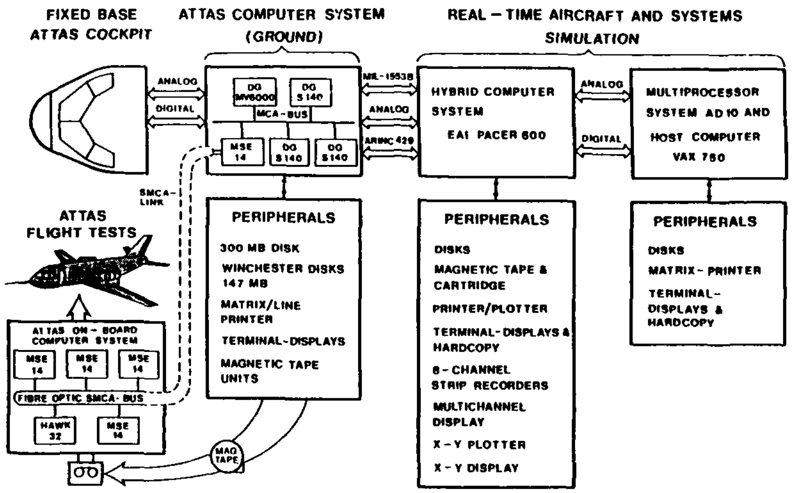


Figure 11. ATTAS ground based simulator

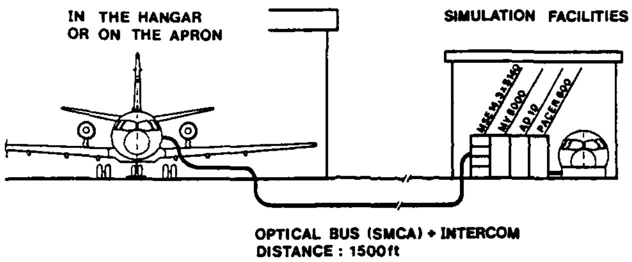


Figure 12. ATTAS fibre optic link to ground based simulation

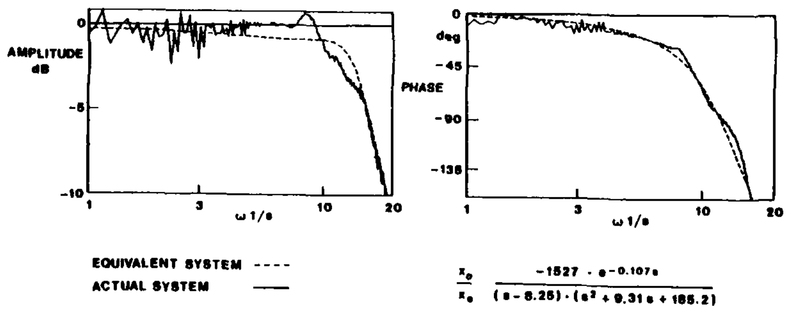


Figure 13. Elevator actuator system frequency response

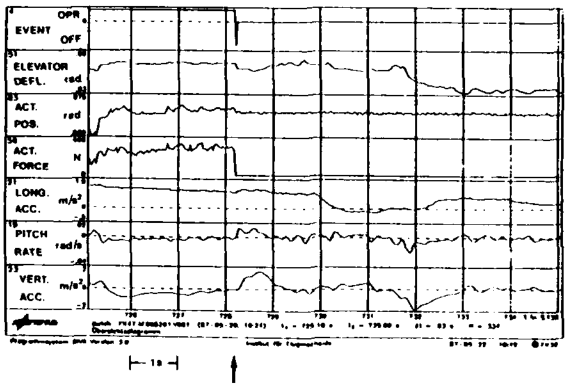


Figure 14. Fly-By-Wire Transient functions histograms

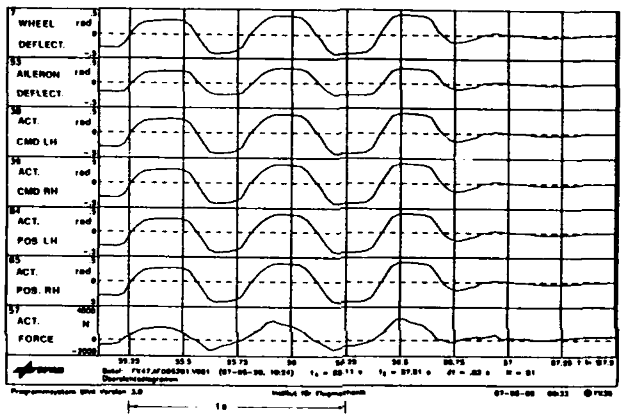


Figure 15. Aileron control system frequency response

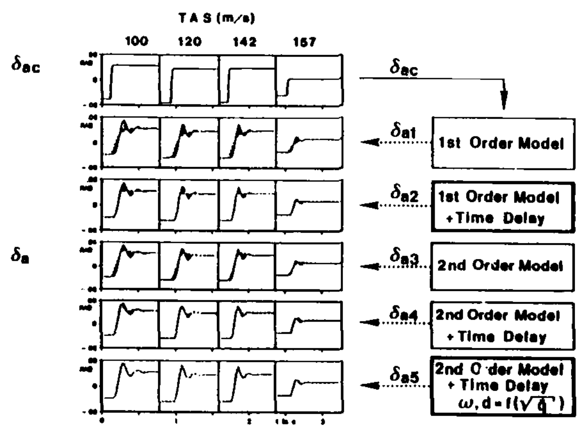


Figure 16. Aileron system identification results

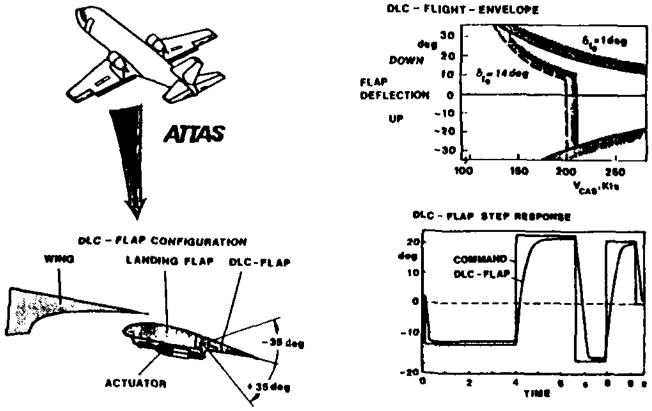


Figure 17. DLC flap step response

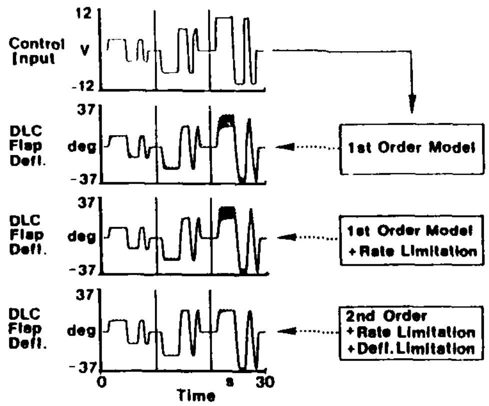


Figure 18. DLC actuator identification results

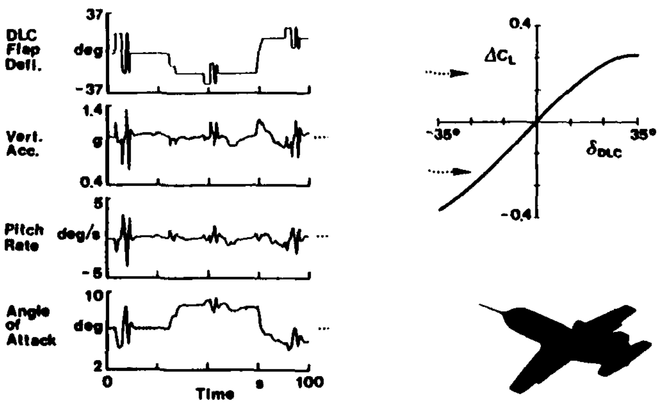


Figure 19. DLC flap system identification results

P-180 AVANTI - Project and Flight Test Program comprehensive overview

R. de' Pompeis - E. Traini
Chief Flight Test Dept. - Chief Flight Test Pilot
I.A.M. Rinaldo Piaggio S.p.A.
Via Cibrario, 2
16154 GENOVA SESTRI - ITALY

SUMMARY

Among the turboprops in advanced flight testing phase, the Piaggio P-180 AVANTI (see fig. 1) possesses exciting features.
Very unconventional shape, characterized by 3 lifting surfaces, pusher props, mid wing very rearback mounted, is coupled with outstanding speed and range performances.
The project and flight test programs planning has required much care for the peculiar characteristic of the aircraft.
Areas of extensive in-flight investigation were:

- flutter expansion
- stability and control at high angle of attack
- laminar flow investigation
- flap system tuning
- propeller stress

This paper focuses on the P-180 project and flight test programs presenting the project concepts and the test results achieved to date.
The flight test program, actually in the certification phase, has achieved more than 600 flight hours and R.A.I./F.A.A. initial certification is expected for mid 1989.

1. HYSTORICAL BACKGROUND

In 1979 I.A.M. Rinaldo Piaggio S.p.A. launched a research program for the study of a new generation F.A.R. 23 category turboprop aircraft.
The only design goals clear in the engineer's mind were near-jet cruise speeds, high fuel efficiency, short runway capability and large cabin dimension for optimum passenger comfort. The different configuration initially under study are shown in fig. 2.

Piaggio settled on a totally brand new aircraft featuring three lifting surfaces (3LS) and pusher props rear mounted on high efficiency laminar flow wing.
The project code was P-180 and the commercial name chosen for the aircraft was AVANTI.
Gates Learjet Corp. entered the program in 1982 with a 50% partnership with the task of producing the aircraft fuselage.

The AVANTI novel aerodynamic formula was extensively studied spending more than 5000 hours in wind tunnel testing (fig. 3) at different facilities in the United States such as the University of Kansas, the Boeing Aerospace and General Dynamics/Convair.
Low speed wing tunnel tests were made in Italy at Piaggio's Finale Ligure factory and additionally at Aermacchi S.p.A. Varese plant.
In the same time, the Center for Research and Development Corp. based at San Diego, CA conducted a finite element analysis for the prediction of the aircraft structure behaviour under all conditions of static and dynamic loading.

While the theoretical analisys and the wind tunnel tests went on, in 1983 Gates Learjet started the fuselage production for the first two prototypes at the Wichita, KS firm facilities.
In the summer 1985, Gates announced its dropping out of the AVANTI partnership for financial reasons.
Since that date, Piaggio proceeded alone toward the first prototype roll-out on the 19th April 1986.
After few months, on the 20th August 1986, during high speed taxiing trials, the chief test pilot Enzo Traini, with the chief flight test Roberto de' Pompeis, lifted-off the aircraft for a little jump in the air, but the officially recorded first flight happened later on, on September 23, 1986.

While the first prototype underwent the first development tests, the second one flew for the first time on May 14, 1987.
After the necessary Development Test Phase (DTP), the two prototypes started with the scheduled F.A.R. 23 Certification Test Phase (CTP) on March 3, 1988.

In August 1988 Italian Air Force Flight Test Center (R.S.V.) flew the P-180 for a preliminary evaluation as short to medium range transport and advanced trainer aircraft.

Up to now more than 600 hours have been dedicated to achieve Federal Aviation Administration and Registro Aeronautico Italiano certifications both expected for the spring of the 1989.

2. THE AVANTI CONCEPT

2.1 AIRCRAFT GENERAL DESCRIPTION

The Piaggio P-180 AVANTI is a single pilot general aviation business aircraft with a 10810 lbs (4900 Kg) MTOW.
It is powered by two PT6A-66 Pratt & Whitney of Canada turboprop engines, each with a maximum rating of about 1450 shp (1081 KW), flat rated to 850 shp (634 KW) at standard sea level up to about 25000 ft (7620 m).
Two counter-rotating Hartzell five blades aluminum propellers push the AVANTI up to 400 kts (740 km/h) top true airspeed.

The aircraft maximum operating speed (Vmo) is 300 KEAS (555 km/h) up to 20000 ft (6561 m). From here up to the 41000 ft (13000 m) maximum operating altitude the mach limitation is 0.67 Mmo (fig. 4).
The landing configuration minimum stalling airspeed is 85 kts (157 km/h).

The passenger pressurized cabin offers great internal comfort thanks to its outstanding dimensions, 72 in (1.84 m) wide and 69 in (1.75 m) high, and to the 8.4 psi pressure differential.
Low level cabin noise and quite complete absence of vibration are achieved thanks to the rear propeller position, to the high laminar flow extension on the fuselage and to the special attachment system of the engines.

The composite materials have been used not extensively so that only about the 20% of the airframe is epoxy made while aluminum alloy was used for the rest.
The major composite sections are the tail, nacelles and forward wing all items out of pressurized area and not in contact with fuel (fig. 5).

2.2 AERODYNAMIC CONCEPT

To achieve their outstanding goals Piaggio' engineers used advanced aerodynamic concepts, some of them never seen before on production general aviation aircraft (fig. 5):

- three lifting surfaces (3LS)
- natural laminar flow (NLF) wing
- pusher propellers
- flaps on both main and forward wing
- delta fins

In many of the aircraft flight conditions, the 3LS layout provides both trim and induced drag lower than they would be with a conventional 2LS design.
Infact during cruise, both main and forward wing produce upward quite self balancing forces and only a negligible amount (about 1-5% of the total load) of negative lift is generated by the horizontal tail.
Since the forward wing's balancing force acts in the same way as main one, this last has to produce a lift lower than a conventional 2LS layout so that the induced drag lift generated is lower too.
Thanks to the low tail load and to the lift sharing between main and forward wing, the 3LS configuration permits smaller lifting surfaces than they would be in a conventional 2LS design thus reducing in part profile drag too.
Setting back the main wing as in the 3LS layout allows a cabin with no internal obstruction and a mid-fuselage wing positioning that is another low drag aerodynamic solution for it reduces interferences between the wing and the fuselage.
Finally the 3LS concept also allows a wider center of gravity travel than a conventional 2LS configuration.

The 3LS layout is responsible for the AVANTI's low cruise drag as well as the existence of natural laminar flow (NLF) on both aircraft wing and fuselage.
To achieve low drag in cruise the main wing has been provided with NLF airfoils studied by the Ohio State University.
The rearloaded profiled used on the wing also give the aircraft the capability to reach a very high critical Mach number (0.71).
Special studies on the fuselage shape and cross sections allowed to keep laminar flow over the nose of the cabin.

In the attempt to keep as low as possible the AVANTI's drag, Piaggio's engineers applied the pusher configuration concept to the aircraft: there being no propwash on the main wing, the laminar flow is enhanced.
The pusher configuration also provides better flying qualities since the rear propeller position gives a stabilizing effect.
To give the aircraft no critical engine the propellers are counter-rotating.
Locating the props rear the wing finally improve the aircraft crashworthiness too.

Flaps on both main and forward wing allow the AVANTI to reach stalling speeds in take-off and landing configuration lower than those that should be achieved with a single surface flap design.
The contemporary deflection of the flaps on the forward and main surfaces keeps the aircraft in trim for every flap configuration change and tuning the deflection time-law with care allows nor pitching moment variation neither pilot action on the stick during transient.

Delta fins fixed on the fuselage aft portion, out of wing or fuselage wake, provide the AVANTI with a longitudinal stabilizing force at low, near-stall speed keeping out the aircraft from deep stall.
They also provide a strong directional stability contribution in every flight condition.

2.3 HYDRAULIC AND ELECTRIC SYSTEMS

The hydraulic system of the P-180 provides hydraulic power for landing gear extension and retraction, nose wheel steer and wheel brake function.
An electric motor driven hydraulic power generating unit, whose main components are a variable displacement pump, a reservoir, a low pressure filter and a landing gear selector valve, is the main item of the system.
In case of lack of pressure from the hydraulic power unit, due to components or line failure, the aircraft shall be brought to a safe landing through a hand pump and an emergency independent ducting system for the brake operation.

The DC generation and distribution systems are mainly composed by:

- a nikel-cadmium 25.2 Volts, 36 Ampere-hour battery that allows the engine start without External Power Unit
- two generators (30 Volts - 400 Amperes) working in parallel with the battery and functioning also as starters
- two control units controlling the engine starts in manner to keep constant the generators output voltage and checking in general the two generators functionability in order to prevent critical conditions due to overvoltage, or inverted currents or overexcitation
- 6 different buses sharing the systems loads in accordance with the systems importance

All the DC systems are protected against accidental failures thanks to left and right systems automatic separation operated by overload sensors, pilot buses cables protection and block diods on the various multiple alimentation lines.

AC generation and distribution is achieved through two static inverters of 250 VA installed in the aircraft radome and controlled by a control unit.
Each inverter, able to supply all the necessary utilities, in connected to a 115 V (AC) bus and to a 26 V (AC) 400 Hz bus.
In case of primary inverter failure, the control unit set automatically the utilities on the secondary inverter that loose the capability of supplying its own utilities.
Major utilities are protected by mean of breakers.

2.4 FLIGHT CONTROL SYSTEMS

The primary flight control systems (F.C.S.) are not power-assisted and utilize traditional

cables and pulleys.
Only ailerons have been rigidly connected through rods because a traditional cable connection should not ensure the sufficient stiffness on such a rearloaded wing; in fact the subsequent not perfect ailerons positioning during high speed cruise would destroy the original pressure distribution over the wing itself.
All the movable surfaces are mass and aerodynamically balanced.

The secondary FCS such as longitudinal, lateral and directional trims are electrically servo-assisted.
In particular the longitudinal trim has a fail safe actuator with both electrical and mechanical redundancy.

The AVANTI's flap system is composed of the following subsystems:

- a main wing outboard fowler flap (MWOF) located on the part of the wing outboard of the engine nacelle.
  Left and right wing flap are moved by a single electrical motor since they are mechanically interconnected.
- a main wing inboard slotted flap (MWIF) located in the part of the wing between the engine nacelle and the fuselage. Also this subsystem is moved by a single electrical motor since mechanically interconnected.
- a forward wing slotted flap (FWF) which is not mechanically interconnected and each one, left and right, is moved by a single electrical motor.
- an Electronic Control Unit (E.C.U.) receives the input from the flap lever, enables the proper movements of the flaps, checks the proper operations of the system supplying the electrical power to the motors and finally phasing the MID position of all the flaps.

2.5 ENGINES

Piaggio chose to install, on the AVANTI, engines of low specific fuel consumption, with high altitude flying capability (at least 35000 ft) and belonging to a vary widely in-flight proved engine family.
That is why AVANTI is powered by two lightweight free turbine PT6A-66 P&W of Canada, each rated at 850 shp at standard sea level up to about 25000 ft.
The PT6A-66 engine has infact the following characteristics:

- low specific fuel consumption (SFC=0.5 lb/hr/hp)
- rear gearbox optimised for the AVANTI
- modularity
- low operation costs
- ease of maintenance
- more than 1000000 hours flown all over the world (all the versions)

2.6 AVIONIC SYSTEM

The system installed on the aircraft, though representing the state-of-the-art avionic technology, is quite simple and uncomplicated to use being studied for a F.A.R. 23 aircraft.
The design criteria for the avionic system were:

- single pilot operation
- low pilot workload
- long range I.F.R. navigation capability
- full integration with other aircraft system
- full provision for growth and development in the future

All the avionic systems installed on board are powered by a conventional electrical supply system.
The prototype No. 2 cockpit is shown in fig. 6 where it is possible to see the following systems:

- 2 Communication VHF
- 2 Navigation VHF
- 2 Transponder
- Weather Radar
- ADF

3. DEVELOPMENT AND CERTIFICATION TEST PLAN

The overall program has been structured in such a way as to cope with the different customer requirements, to provide for a low risk for the manufacturer, and to design, develop and manufacture the aircraft in a cost effective way.
The DTP and CTP phases have been organized in manner to show compliance with the normal F.A.R. 23 rules and the additional special requirements (Special Conditions) issued by F.A.A. on the following items:

- composite structure fatigue damage tolerances
- doors and exits
- forward and main wing flap system interconnection
- forward wing loads
- propeller ground clearance
- propeller markings
- propeller ice protection and exhaust gas impingement
- cockpit smoke evaluation
- buffet onset envelope
- contamination on laminar flow
- inadvertent excursion beyond max operating speeds.

To achieve these goals, the planning of the P-180 development and certification tests have required 5 test articles:

- 1 static test article
- 2 flying prototype aircrafts
- 1 flying production aircraft
- 1 fatigue test article

The planning and main milestone are shown in fig. 7. The DTP phase has lasted 30 months while the CTP's end in expected for mid 1989.

4. FLIGHT TEST PROGRAM

The flight test programs mainly based on two flying prototypes while the first production aircraft will be utilized only for confirming all the configuration changes either retrofitted on the prototype or ex-novo installed (like the brand new two pieces windshield).
Up to date more than 600 flight hours have been totalized on both prototypes (fig. 8) so showing the aircraft good reliability.
The flight test program schedule is based on 15 productive flights per month per test aircraft.
The test program has been prepared in agreement with the following criteria:

- utilization of the prototype No. 1 mainly for the flight envelope expansion while the prototype No. 2 is dedicated to powerplant and systems testing
- keeping on ground one aircraft at a time and not more than 4 weeks
- enlargement of the flight test envelope through handling qualities checks and structural flutter clearance so that step by step crew familiarization with the aircraft characteristics is possible
- in-flight testing of the proposed experimental modifications in order to identify the satisfactory solutions to be included in the final production configuration
- utilization of the multimission concept to effectively evaluate and integrate the total aircraft systems in lieu of individual component or system testing
- extensively use of model data and simulation results before testing extreme flight conditions

The main areas tested and the tests sharing between the two aircrafts are shown in fig. 9.

Since all the modifications have been selected, the aircraft configuration has been frozen and the test program is proceeding toward certification.
In fact, following Piaggio's own evaluation, R.A.I. accepted to fly the aircraft, on the 3rd March 1988, as having achieved an adequate standard for certification.
As already stated, according to the actual test program schedule Piaggio expects to reach R.A.I. initial certification for the spring 1989 while F.A.A. certification is scheduled to be achieved not long after.

5. FLIGHT DATA ACQUISITION

The flight data acquisition system was designed to satisfy the needs for a rapid exposure

and understanding of problems.
Data measurement, recording and processing systems were geared in terms of capacity, accuracy, resolution and rapidity of data production to facilitate and ensure expedient tactical decisions relating to remedial action or redirection of the flight program as required.

5.1 FLIGHT TEST INSTRUMENTATION

To carry out the flight test program as scheduled in a cost effective way, Piaggio's engineers provided both the 1st and 2nd prototypes with a number of F.T.I. facilities appropriate to the test program for that aircraft, plus those required by the back up role. These facilities include data acquisition and on board recording system, a single channel telemetry for the total PCM data stream transmission mixed with intercom speech, video camera system, flutter excitation system, time correlation and syncronization systems. AVANTI's data acquisition system (fig. 10) actually allows the measurement of about 250 parameters that roughly include:

- 65 pressure measurements
- 25 motion/displacements measurements of the aircraft control surfaces or of any other aircraft component
- 85 temperature measurements
- 15 electrical signals measurements
- 3 aircraft linear accelerations (body axis Jx, Jy and Jz)
- 3 aircraft angular velocity measurements
- 54 miscellaneous (flows, forces, torques, events, vibrations)

During both the development and certification tests, the parameters number has been changed from test to test depending on the test purpose.
The total sample rate generally used for data acquisition is 2048 sample/second.
Any single parameter can be sampled from a maximum of 228 s/s to a minimum of 0.5 s/s.
The total sample rate of the system can be enlarged up to 32 times so permitting a maximum sample rate of 65536 s/s.

Specific tests such as flutter or airloads survey tests have required specific F.T.I. installations.

Flutter test F.T.I. consists in 18 pyrotechnical bonkers located on forward and main wing, horizontal and vertical tail able to excite any airframe vibration mode.
In addition inertial shakers are placed on the two engines between the flange and the crandle of each one to study eventual coupling between airframe and engines themselves.
The airframe free response is detected by 36 pre-amplified accelerometers whose signals are first low pass filtered and then recorded, together with the pilot's voice on a FM magnetic tape recorder.

Due to the different geometric and aerodynamic characteristic of the three lifting surfaces, their pressure distribution was checked using different measurement means for each one.
Spanwise and chordwise pressure distribution on forward wing was determined via 15 skin pressure transducers while airloads acting upon the wing have been checked by means of 19 strain gauges.
Vertical tail was controlled through 40 pressure taps connected to a scanivalve, while the stabilizer and the elevator were instrumented both with 15 skin pressure transducer and 40 scanivalve pressure taps.

5.2 GROUND DATA PROCESSING FACILITIES

Real time test monitoring is provided by a ground station linked to the test aircraft via a telemetry system with a computer controlled paraboloid antenna.
The computerized ground station showed in fig. 11 allows to receive, record on back-up tape, decode, display and analyse in real time the data acquired while testing the aircraft.
64 parameters among the about 250 data transmitted via telemetry are available both in digital and bar-graphs format on three color monitors all connected to thermal printers.
Furthermore it is possible to select 16 of the 64 parameters available and record them on two 8-tracks paper recorders.

Dedicated leds properly located on the front consolle in the ground station allows the flaps and gear position checking while a line of green light leds give information on the state of the pyrotechnical charges installed on the aircraft for specific flutter tests.

Telemetry antenna aiming is controlled by means of a HP-71 pocket computer that analyse the aircraft signal quality and set the best receiving antenna angle.

Post flight data processing include primary PCM into secondary computer tape conversion process plus total flight time history plotting.
The flight test recorded data can then be processed by each specialist via a terminal network managed by a Digital PDP 11-44.
Compared with the flight test program complexity, the AVANTI's post processing facilities allows a quick turn-around of data for an efficient flight program proceeding.

6. SAFETY EQUIPMENTS

Special care has been spent in the determination of the safety equipments to be installed on the two aircrafts.
Both the aircrafts have been provided with an expecially studied egress system enabling pilot and co-pilot to quickly reach the exit doors and bail out.
Since the first prototype has been dedicated to the flight envelope expansion it has been equipped with a tail parachute which can be used during both flutter or high AOA investigations.

7. FLIGHT TEST RESULTS

7.1 FLUTTER

AVANTI's high operating and dive speed coupled wiht T-tail configuration have required much care in reaching flutter envelope borderlines.
Before flutter flight testing, extensive Ground Vibration Test (GVT) were accomplished by French ONERA and German DFVLR at Piaggio's facilities in Genova.
First flutter investigation flight was performed at 140 keas (260 Km/h) and 14000 ft (4270 m), on the 31-10-86 and going ahead with a 10 kts step finally on the 15-2-88 the flutter envelope has been successfully opened up to 300 keas (555 Km/h) Vmo and 0.7 Mach Mmo at 30000 ft (9144 m).
Fig. 12 shows the test point flown for the flutter envelope investigation.

Thanks to the great extent of on ground tests and computer analysis no surprise came out from the flight test and the frequencies and the dampings found are in agreemnt with the calculated ones.
Fig. 13 shows the comparison between calculated and experimental dampings/frequencies for different surfaces vibration modes.

Though the maximum operating speed/Mach has been already cleared, dive speed/Mach is still restricted pending completation of the airframe ongoing static tests and windscreen ground testing.
Piaggio's engineers actually do not foresee any problem to reach the VD/MD as scheduled.

7.2 STALLS

High angle of attack handling qualities and stall characteristics were extensively investigated in the Piaggio's low speed wind tunnel to provide the aircraft with a quite usual stalling behaviour.

Initial studies indicated a 4° forward wing angle of incidence, with respect to the airplane reference line, to obtain the forward wing stall before the main one in order to avoid the hazard of pitch up at high AOA.
Moreover stalling the forward wing before the main one set back the aircraft neutral point so that better longitudinal stability is also achieved and higher stick force is required to stall the aircraft.
Initially experienced stall behaviour was not satisfactory because nose down was not sharp and the main wing dropped laterally when reaching more than 25° of AOA in clean configuration.
Positive considerations came from the fact that stall natural warning was well clear in clean and landing although in take-off configuration it seemed to disappear for a while during the deceleration.
Longitudinal, lateral and directional controls were always fully effective before, at and after stall.

Reducing the elevator deflection from 25° down to 22° allowed to reach lower stall angle of attack and introducing partial leading edge stall strip on the forward wing improved stall behaviour since resulted in a sharper nose down. Unfortunately undesired wing dropping was still experienced.

In order to reduce the dynamically reached stalling angle of attack, the elevator deflection was reduced progressively from 22° down to 16°. Flight tests showed that wing stalling was not yet completely satisfactory.

The outboard main wing was therefore fitted with a small stall strip located in the proximity of the engine nacelles.
Moreover delta fins were enlarged in the attempt to enhance their stabilizing directional and longitudinal forces at high AOA.
In the same time, to unload the low aspect ratio forward wing in order to improve the aircraft cruise performance, the canard angle of incidence was reduced to 3°.
To get the same forward wing stalling characteristics achieved with the 4° angle of geometric incidence, a full span strip was installed on.

Final configuration testing showed satisfactory stalling characteristics no wing lateral drop with any flap and gear position regardless of the center of gravity position and in every flight condition required by the F.A.R. rules.

P-180 stall can be actually defined quite usual, with full control authority, with a strong unmistakeable natural warning, with full longitudinal control deflection and strong stick force employment.

7.3 STABILITY

Despite of its unconventional desing, P-180 has shown conventional stability and controllability throughout its flight envelope (see fig. 14a)
Both dynamic and static longitudinal stability resulted within the required F.A.R. 23 limits and were satisfactory judged by the test pilots in all the flight conditions.
In particular it must be noted that at near stall speeds, both stick force and elevator deflection to pull the aircraft are enhanced since the forward wing early stall sets back the aircraft neutral point so increasing the stability margin (see fig. 14b).
During the development phase, static lateral-directional stability was improved installing a new fairing on the fin in order to reduce the surface angle of swept, to increase, in same time, the rudder surface and to achieve also as secondary effect the reduction of the wake behind the top of the tail.
In particular to enhance the lateral directional stability in landing configuration the aircraft was provided with brand new nose landing gear door of reduced surface (see fig. 14c)
In addition to the above mentioned modifications, to improve both lateral/directional and longitudinal stability in all the flight conditions and expecially at high angle of attack the delta fins were enlarged.
Final configuration includes all the modification previously outlined and it has shown very satisfactory characteristics throughout the flight envelope.

7.4 PERFORMANCES

The test technique utilized for the cruise performance determination was the one of the level flight stabilized points.
Because of the high speed/Mach achievable in level flight, the method used in the performing of the tests was that named constant "W/δ" typically used for jet aircraft.

Predicted speed performances of the aircraft were fully verified since top true airspeed reached during flight test, cruising at 30000 ft (9144 m), was 403 kts (746 Km/h).
The in-flight measured specific range was slightly lower than expected (3% less).
Flying a typical I.F.R. mission as provided by the National Business Aircraft Association (N.B.A.A.) at 39000 ft (11887 m) and 300 kts (556 Km/h) of true airspeed, the maximum range was 1510 nm (2800 Km) carrying 4 passengers and 2700 lbs (1225 Kg) of fuel including fuel reserves for alternate airport at 100 nm (180 Km) and 30 minutes of loiter at 5000 ft (1520 m).
Climb performance tests were accomplished using the sawtooth climb technique.
Test results showed a 3500 fpm (18 m/s) two engine climb rate and a 1000 fpm (5 m/s) one engine out climb rate achieved at standard sea level and MTOW.
The calibrated one engine level flight airspeed achieved was an outstanding 200 kts (370 Km/h) at I.S.A. 5000 ft (1520 m) and MTOW.

Take-off and landing performances have been determined using the French STRADA trajectographic laser system at the Centre d'Essais en Vol homed at Bretigny sur Orge (France). The total field lenght required for take-off the P-180 at MTOW in a standard day was 2500 ft (760 m) while, at Maximum Landing Weight the distance necessary to land and stop the aircraft without use of reverse was 2000 ft (610 m).

7.5 AERODYNAMIC INVESTIGATION

The aerodynamic investigation on the P-180 had mainly three tasks:

1) the verifying of the effects of contamination of laminar flow on different airframe areas such as main and forward wing, horizontal and vertical tail, fuselage, engine nacelles;
2) the checking of the NLF real extension on the full scale aircraft and in real operating condition;
3) the measurements of the pressure distribution over the forward wing, the main wing, the horizontal and vertical tail.

In particular, checking the effects of contamination on the AVANTI's NLF wing airfoils and the pressure distribution measurements over the lifting surfaces were requested respectively by two F.A.A. Special Condition, SC-3 and SC-5.

7.5.1. NLF visualization

Special flow visualization test on the full scale aircraft were executed to study NLF development and transition mode on the lifting surfaces, the engine nacelles and the fuselage nose.
NLF visualization was performed utilizing sublimating chemical paints.
The visualization method chosen is related to the fact that higher heat exchange inside the turbulent boundary layer enables the chemicals to sublimate before the chemicals painted on laminar flow surfaces.

Since chemicals colour is white, the aircraft areas to be checked were painted in black (see fig. 15). Chemicals sublimating in the turbulent boundary layer area was detected by the fact that those surfaces progressively became black loosing the paint while laminar flow areas appeared white coloured.

Flight test technique included smooth take-off and a high speed direct climb up to the test altitude where airspeed stabilization was performed for almost 15-20 minutes depending on the air temperature and on the chemical used.
Camera pictures and video recordings were taken from a chase airplane and from the inside of the test aircraft too.
Choosing with care the test sequence it was possible to take significant recordings of more than only one point in the same flight.
The relatively low sublimation speed of the chemical used and the accomplishment of an opportune descent and landing have allowed to recognize the transition line of the last stabilized point also after landing.

In order to choose the chemicals to be used, two flights were performed applying diphenil on the right wing and acenaphtene on the left one.
Test results showed that for the typical test altitudes and airspeeds acenaphtene behaviour was better than diphenil.
After the tuning of the chemical type, the flight test program covered some typical cruise conditions as presented in the table No. 1.
The results achieved in terms of extension of laminar flow on the surfaces investigated are reported in table No. 2.
It must be pointed out that NLF extension reported have been measured after landing.

It is significant to note that NLF on main wing reaches the 60% on the upper and lower surfaces of the wing tip, while on the inboard it decrease down to about 40% because of the less favourable airfoil characteristic and a higher turbulence level due to the forward wing wake.
An on-ground taken picture (fig. 16) shows the flow pattern over the inboard wing obtained using acenaphtene. Small protuberances, such as insects or dust, generated vortexes that caused the spikes visible in the picture.
The NLF investigation will be completed to cover all the aircraft flight envelope.

7.5.2 Effects of contamination

Effects of contamination over AVANTI's NLF airfoils were simulated installing a minimum size adhesive tape strip (trip strip), 0.12 in. (3.12 mm) wide and 0.006 in. (0.15 mm) thick, at the 10% (m.a.c.) of the mean aerodynamic chord of the main wing. Small size trip strip was selected in order to reduce the effect of the tape itself on drag.
Tests were repeated installing the trip strips on the other lifting surfaces (i.e. forward wing, vertical and horizontal tail) and on the engine nacelles and the radome too.

Fig. 17 shows a comparison between AVANTI's speed power polars with the wing tripped at 10% m.a.c. of upper and lower surfaces and with the complete aircraft tripped at 10% m.a.c. of every lower and upper lifting surfaces, while the fuselage and the nacelles were tripped at 7.5 inches (19 cm) from their noses.
Artificial tripping on the wing only caused an increase of 3.6% of necessary power for level flight, while tripping the complete aircraft the necessary power showed a 7% increase.
No significant change in the longitudinal or directional stability or controllability was detected during the tests.

For final certification purposes, tests will be again repeated with a wider and thicker trip strip along all the surfaces leading edge in order to simulate the worst contamination conditions.

7.5.3. Airloads survey

Regardless of computer analysis and wind tunnel testing, F.A.A. believed that the P-180 contains sufficient novel features in terms of aerodynamic design and construction that flight test loads verification on the full scale aircraft was required.
In addition to concerns about load distribution, the full scale airplane was believed to respond differently than predicted.
For those reasons, F.A.A. formally requested Piaggio, as stated in Issue Paper C-2, to conduct in flight airloads measurement in order to verify the correctness of the data used to calculate the aircraft loads.

The test planning included a great amount of test points since the flight conditions to be performed were:

- level flights at 6000, 20000, 30000 ft, clean and landing configuration;
- climbs and descents at 20000 ft, clean and landing configuration;
- sideslips up to 15 deg of skid;
- coordinated turns up to 60° of bank.

In addition to the tests above mentioned, F.A.A. requested the measurement of the top roll moment on the aircraft tail by strain gauges.
Initially pressure measurements were performed on both port and starboard lifting surfaces to check airloads simmetry.
Later on, when airloads simmetry was verified, only port side was instrumented.

While scanivalve on vertical tail and pressure transducers on forward wing gave very good results in every flight condition, slightly lower quality results were experienced with the horizontal tail pressure transducer.
In order to obtain pressure measurements of good quality, horizontal tail was so fitted with pressure taps connected with another scanivalve, leaving in site the pressure transducers previously installed.

Example of the pressure readings obtained on forward wing via pressure transducer is shown in the fig. 18a for a typical cruise condition (i.e. 20000 ft, 260 Kias).
For the same flight condition, horizontal tail scanivalve data are presented in fig. 18b.
Tests regarding the determination of the top-roll moment on the AVANTI's T-tail are now underway.

7.6 FLAP

7.6.1 Flap system schedule tuning

AVANTI's unconventional flap system configuration has required an intensive investigation to determine the optimum flap schedule.

Initial behaviour of the aircraft was found unsatisfactory since an inversion of stick force was required to balance the pitching moment generated during an UP to MID flap maneuver.

All the other possible flap deflections caused no problems since the force to apply to the stick was without uncomfortable inversions.
In order to reduce the undesirable effects during UP to MID maneuver, the canard flap schedule time offset was modified.
Acting directly on the system ECU, the FWF deflection was anticipated of about 3 seconds in order to reduce the initial outboard wing flap pitching effect.

Though the desired effect was achieved in the UP to MID maneuver, the attempted solution gave as secondary result an increasing of the stick force necessary to balance the pitching moment variation during MID to DOWN deflection when a stronger influence is exercited by the MWOF and MWIF.

In order to identify the best deflection schedule the test crew was abled, by a special manual control, to choose the best deflection sequence (fig. 19) with quite negligible stick force application or pitching moment variation.

Besides reducing FWF deflection angles from 18° down to 13° in take off position and from 38° down to 32° in landing, the positive effect of the flight test determined schedule is shown in fig. 20 where it is possible to appreciate the lower stick force employment in the UP to MID critical maneuver.
The flight test determined extension/retraction law was then fitted on both the prototypes and will be fitted on the production aircrafts too.

7.6.2 Flap system failure effects.

Beyond the FAA/RAI rules, AVANTI's flap system have to meet two additional requirements: the SC-7 Special Condition that states requirements about the forward and main wing flap interconnection and the C-6 Issue Paper that requires the demonstration of the aircraft capability to be safely landed in any combination of synchronized or unsynchronized extreme position between MWOF and FWF using special flight procedures.

To demonstrate compliance with the above Special Conditions, the prototype No. 1 was equipped with a flap failure simulation box consisting in four switches connected to the power supply of each flap sub-system.
Such a system allowed to simulate both asymmetric and symmetric flap failure in any desired instant of the deployment/retraction maneuver.

Though all the tests have been not completed yet, the actual test results show that all the most critical failures involving flap position asymmetries between any intermediate flap position are controllable and trimmable and allow a safe aircraft landing with both forward and aft center of gravity position.

Some of the possible failures during a flap maneuver or retraction involving extreme position require a peak stick force not greater than 77 lbs (35 kg) and constant stick force not greater than 44 lbs (20 kg) allowing the pilot to reconfigure the aircraft in the last symmetric flap position.

In any event hazardous attitudes are not reached and safe landing is always possible for any failure simulated without, in general, adoption of particular flight procedures.
Only in case of failure during extreme deflection it is necessary reduce the flap asymmetry and accomplish the landing with an higher (from 5 to 25 kts) than normal approach airspeed depending on the type of asymmetry.

7.7 POWERPLANT

Both experimental and production PT6A-66 engines have demonstrated their reliability since the beginning of the program.
Because of the pusher configuration the engines have been installed just the contrary of the usual way. So the pusher installation of engine required some tunings. In fact 4 different nacelles were tested in order to improve the engine bay cooling flow. The repositioning of oil cooler NACA inlet and the reshaping of the nacelle rear side and of the exhaust tubes finally allowed to reach a satisfactory configuration.
Since this engine model never flew before at altitude higher than 35000 ft (10670 m), Piaggio experienced oil lubrification system disturbances at high altitude.
The oil troubles were solved thanks to minor modifications to the engine reargearbox and to the oil pipes size.

The airflow behaviour at engine inlet was extensively checked using tufts and by means of pitot probes throughtout the aircraft flight envelope, specially during slams and chops transients. In all the flight conditions tested the engine inlet has shown to be able to satisfy the engine demand.

Since the high maximum operative Mach and airspeed of the aircraft, many concerns existed on the propeller stress levels.
The Hartzell five blades propeller that was initially tested when the flight envelope of the aircraft was limited to 210 kias (390 Km/h), showed infact stress levels very near the limits. Thus Hartzell provided AVANTI with new reinforced propellers.
New stress survey performed up to 305 kts (565 Km/h) have indicated that the new model stress was well under the limits for the infinite life (fig. 21).

8. CONCLUDING REMARKS

Currently the AVANTI flight test program is proceeding as planned and no further development problem has been identified.
Though unconventional and therefore requiring more careful flight testing, the 3LS formula confirms itself as an aerodynamic layout capable to furnish very outstanding performances coupled with exceptional handling qualities.
That is the reason why so many new projects of general aviation aircrafts are characterized by the 3 lifting surface layout.
The general aviation market, so severe and competitive, will give the answer if this layout is destined to a successfull destiny or has to remain an exciting mind exercise only.

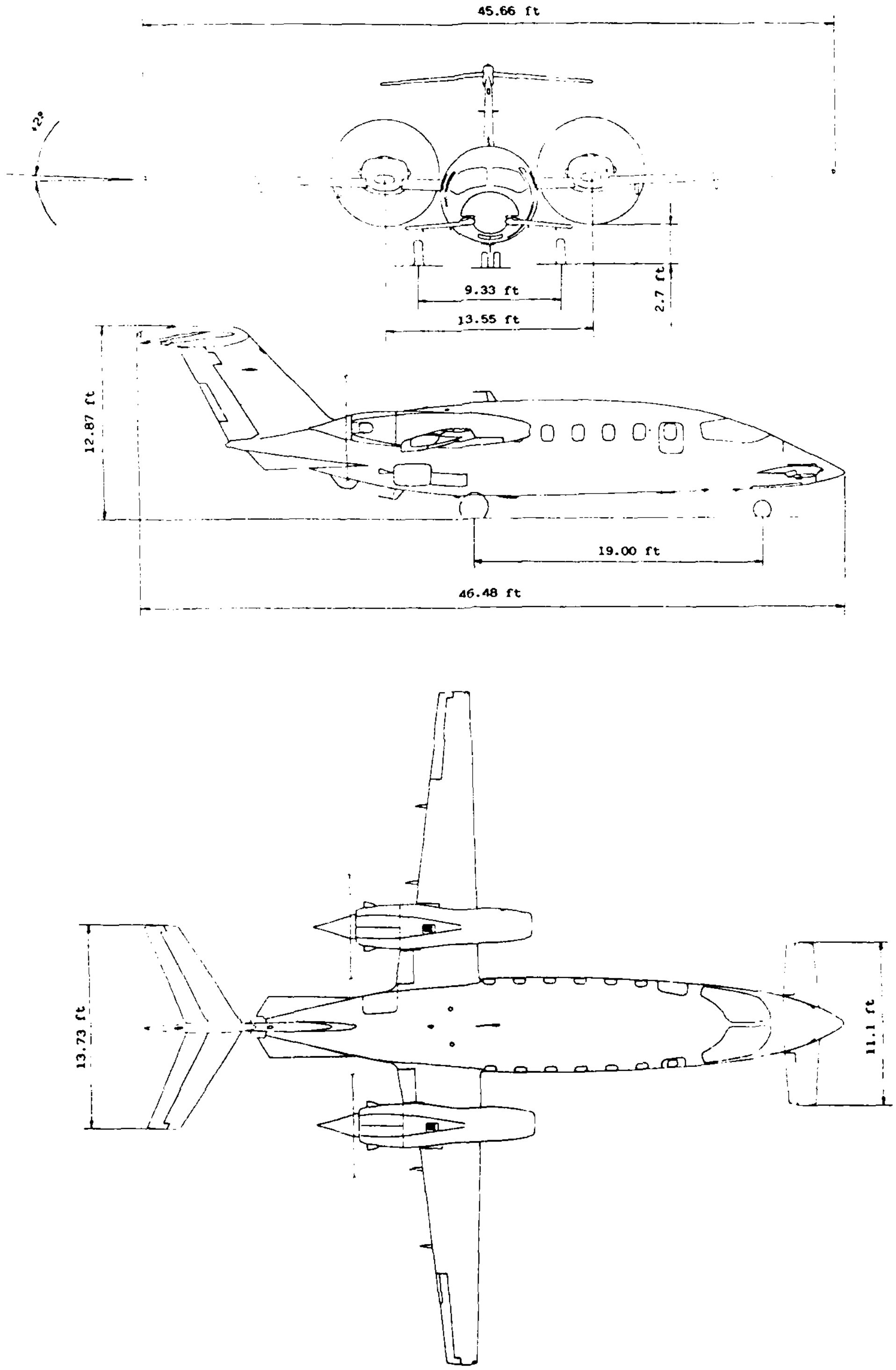


Fig. 1 - P.180 Three sides view with external dimensions

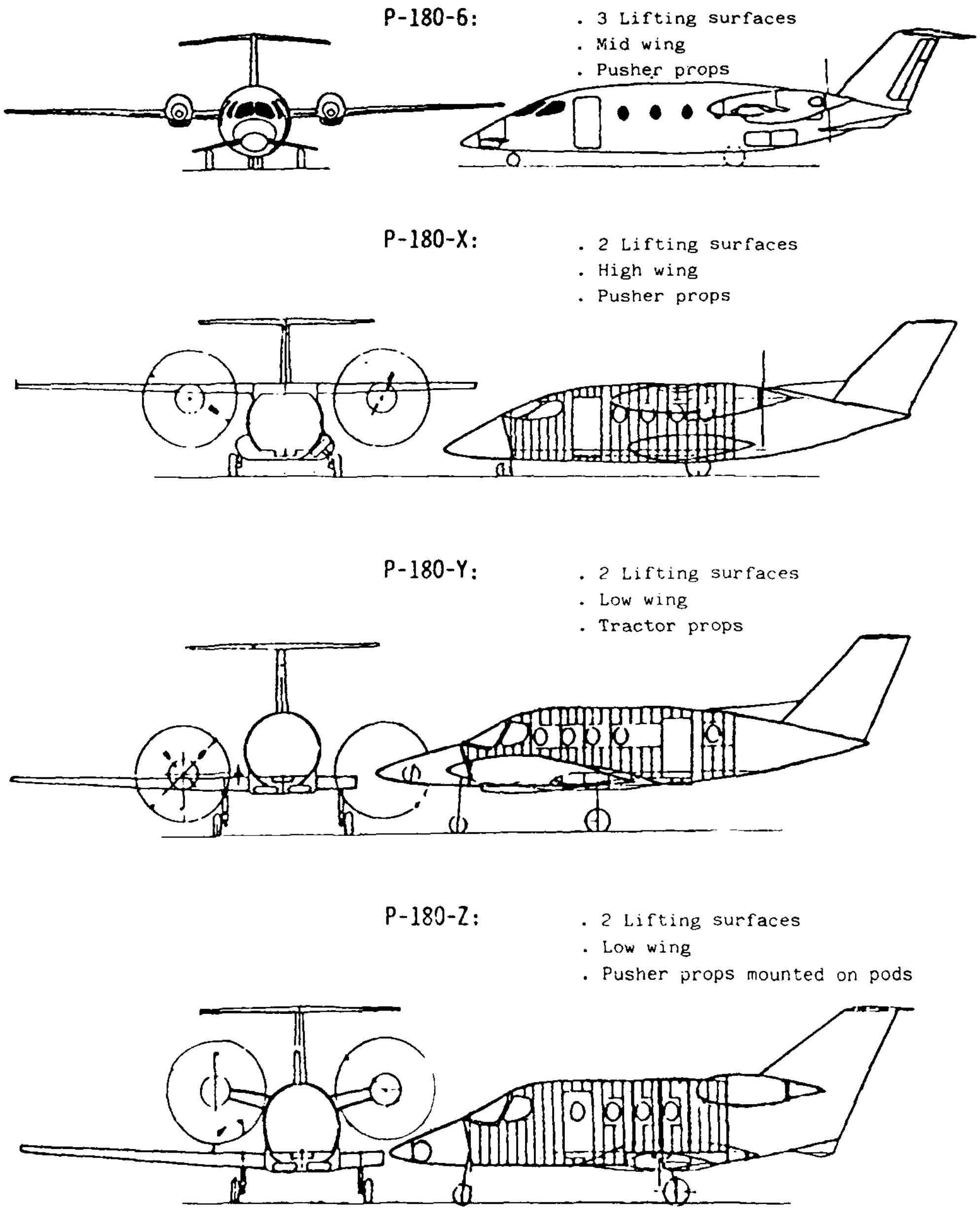


Fig. 2 - Layout comparison between components of the P.180 family

Fig. 3 - P.180 wind tunnel testing

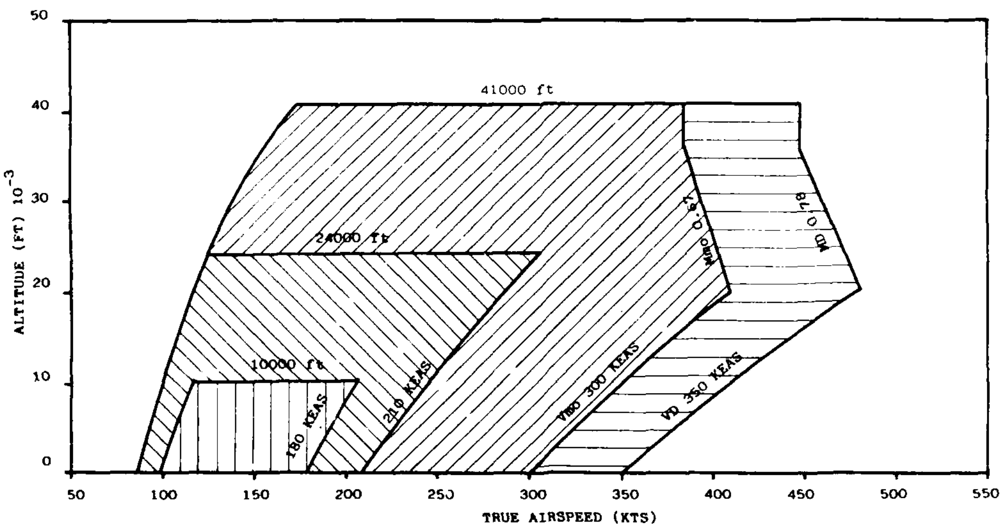


Fig.4 - P.180 Flight envelope

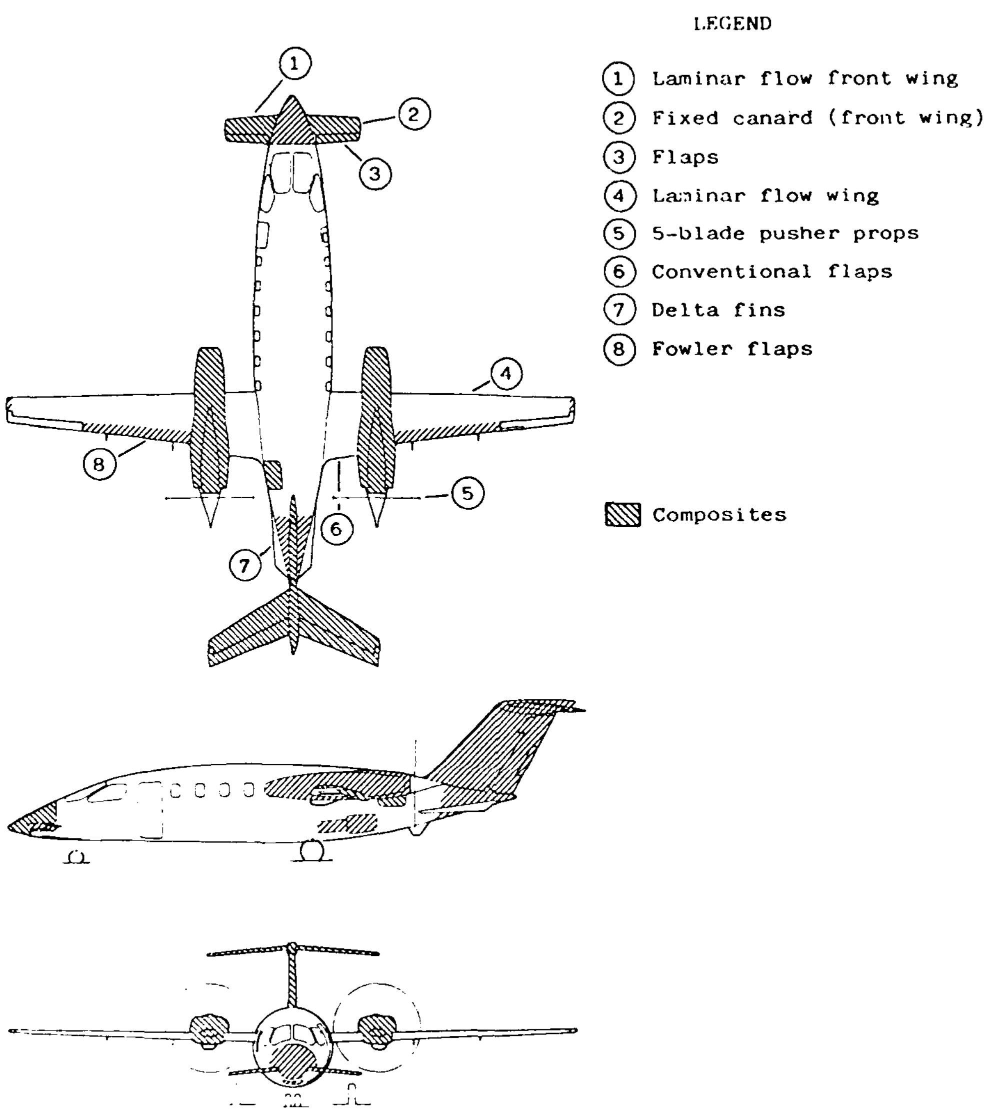


Fig. 5 - P.180 Aerodynamic concepts and composite structures

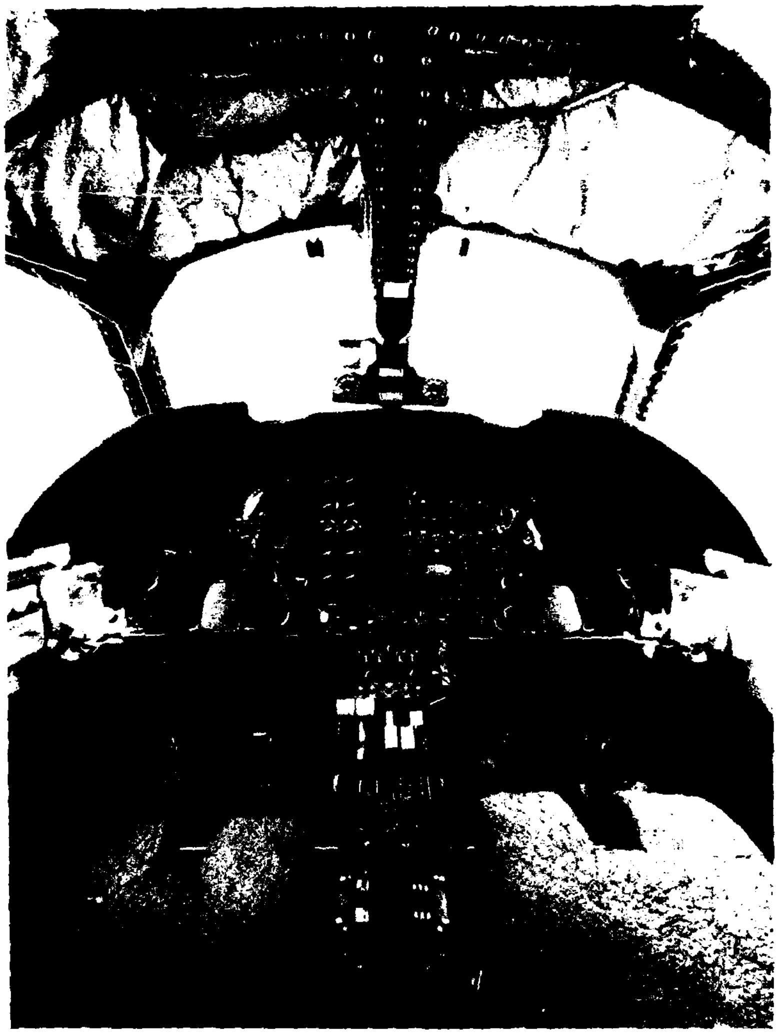


Fig. 6 - P.180 Prototype No.2 cockpit

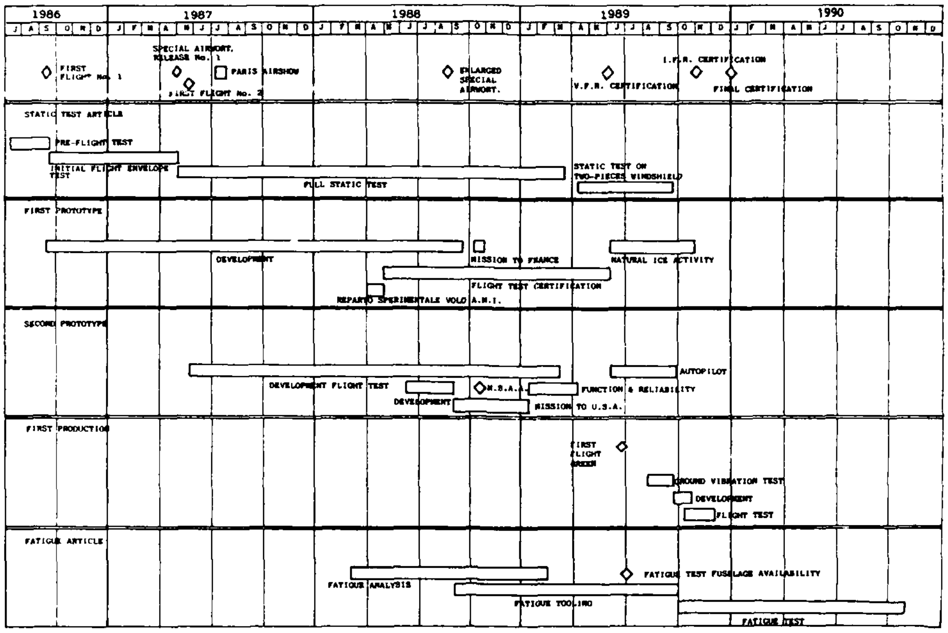


Fig. 7 - P.180 Development and certification test plan

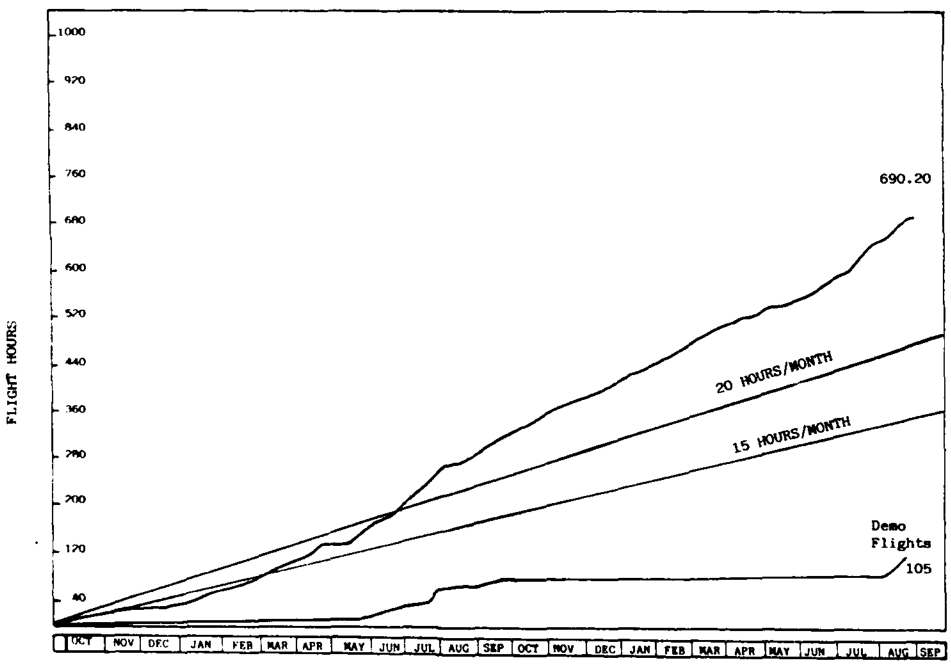


Fig. 8 - P.180 program flight hours

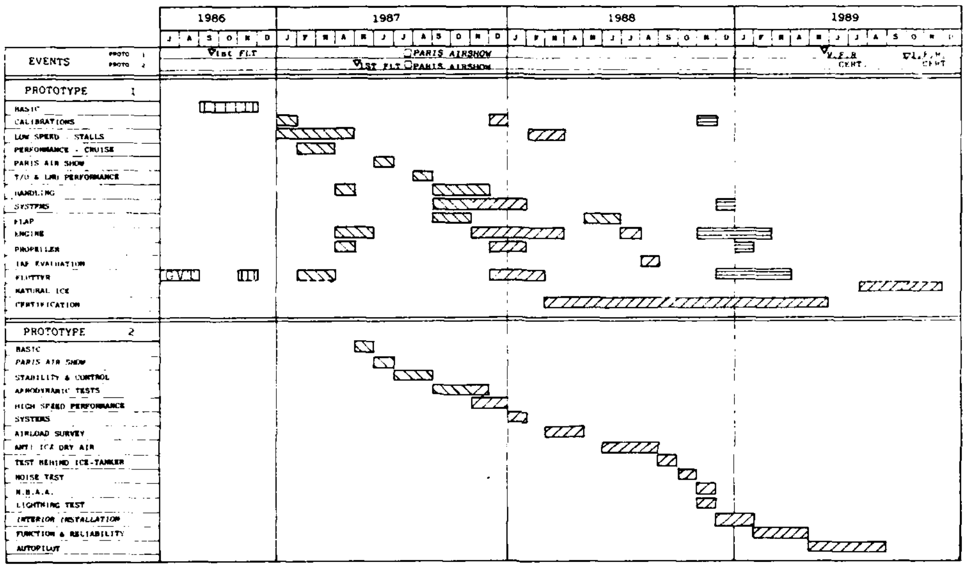


Fig. 9 - P.180 Flight test activity

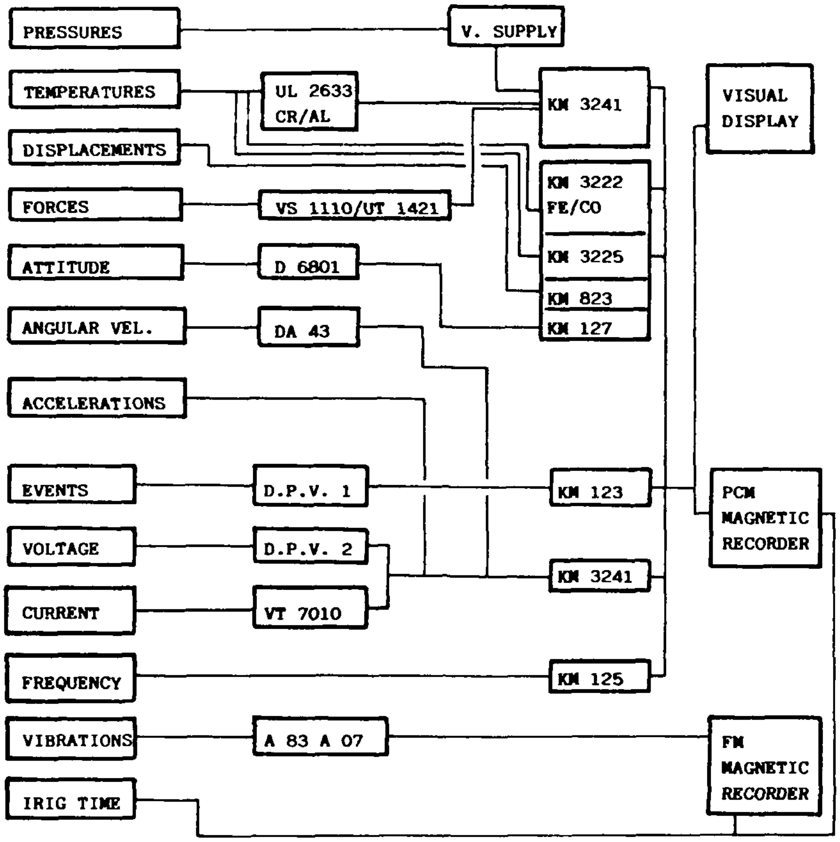


Fig. 10 - P.180 Flight Test Instrumentation

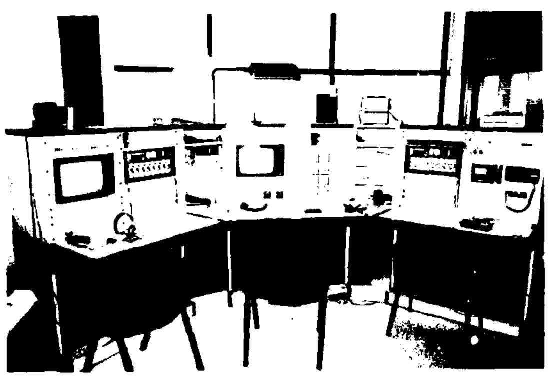


Fig. 11 - P.180 Ground Station

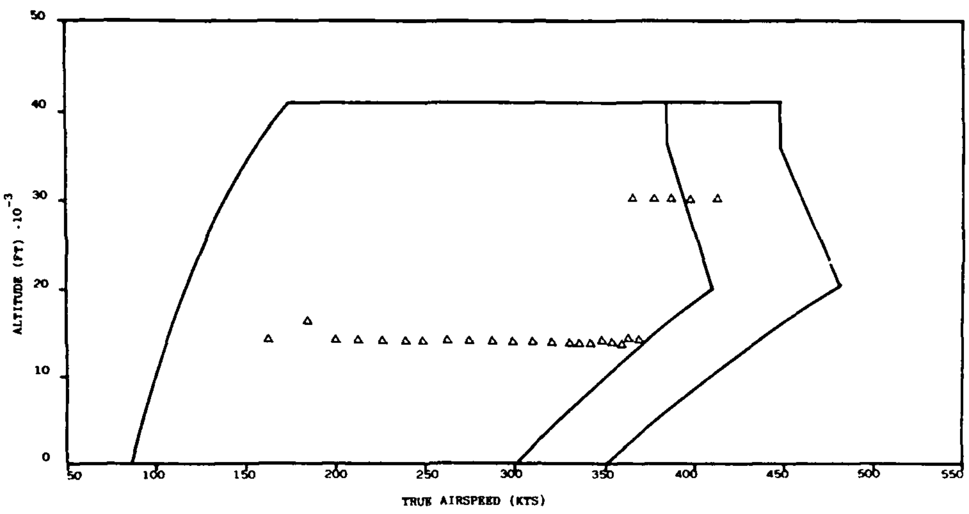


Fig. 12 - P.180 Flutter flight envelope

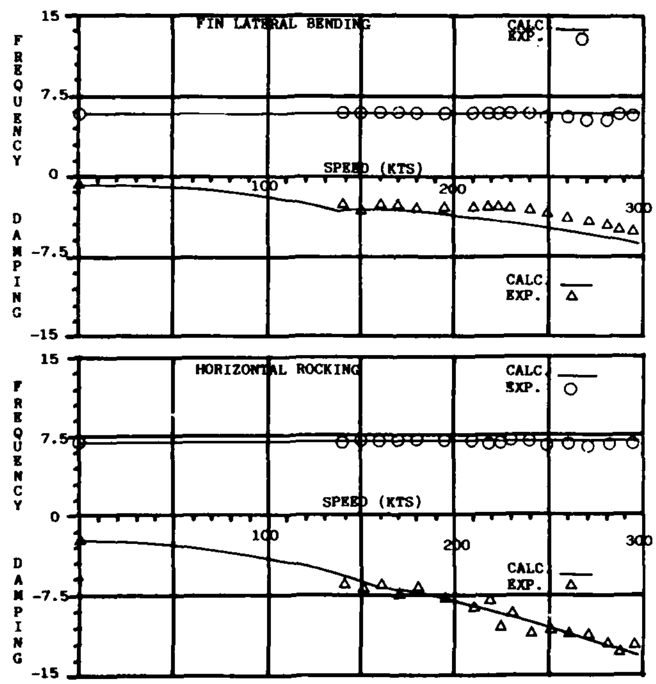


Fig. 13 - P.180 Flutter results and theoretical data

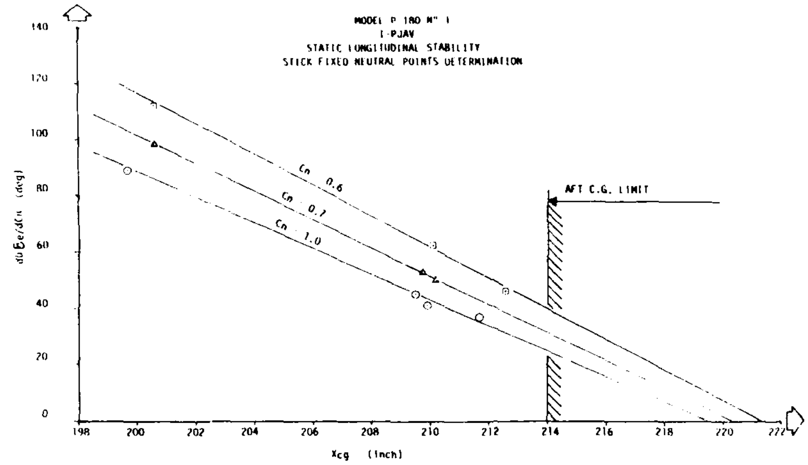


fig. 14a

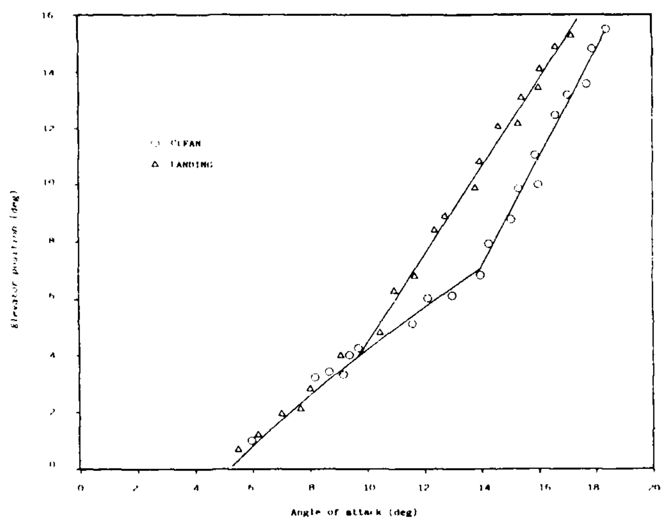


fig. 14b

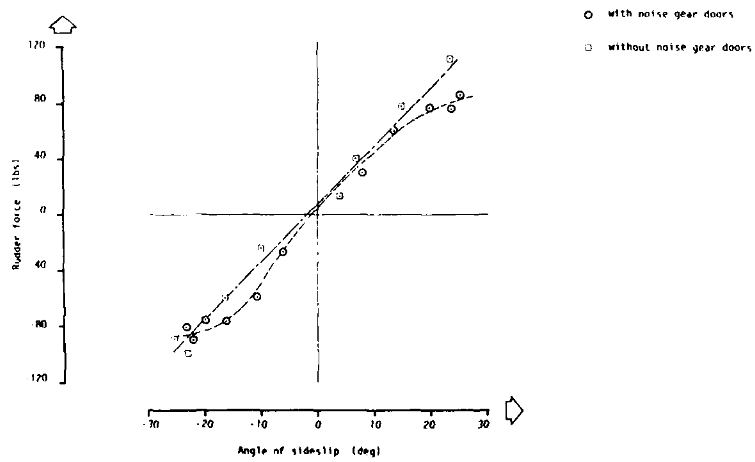


fig. 14c

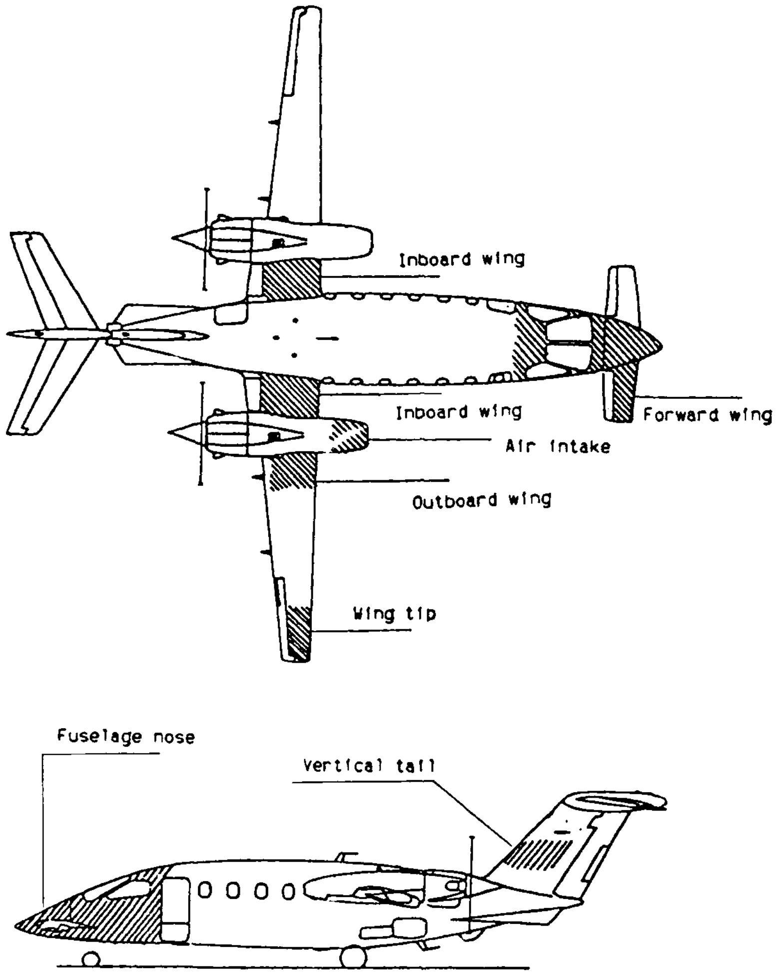


Fig. 15 - P.180's areas of NLF investigation

Table 1: P.180 FLIGHT ENVELOPE INVESTIGATED FOR NLF

Test No.	Altitude ft	CAS Kts	Mach	Re/ft 1/ft	CL	AOA deg
1	8000	220	.384	2.19	.357	1.9
2	25000	230	.558	1.89	.333	1.2
3	28000	240	.617	1.90	.306	1.0
4	8000	220	.384	2.19	.349	1.9

CAS Calibrated airspeed (knots)
Re/ft Reynolds Number per unit lenth (1/ft - millions)
CL Aircraft Lift coefficient
AOA Angle of attack

Table 2: P.180 NLF DEVELOPMENT ON MAIN AND FORWARD WING

Test No.	WING sta. 1400 (% chord) upp./low.	WING sta. 2600 (% chord) upp./low.	WING sta. 6000 (% chord) upp./low.	Fwd WING (% chord) upp./low.
1	40 / n.a.	n.t. / n.t.	n.t. / n.t.	n.t. / n.t.
2	40 / n.a.	n.t. / n.t.	n.t. / n.t.	n.t. / n.t.
3	44 / 14	30 / 21	59 / n.a.	17 / 23
4	26 / 24	33 / 25	60 / 60	20 / n.a.

n.t. not tested
n.a. not available

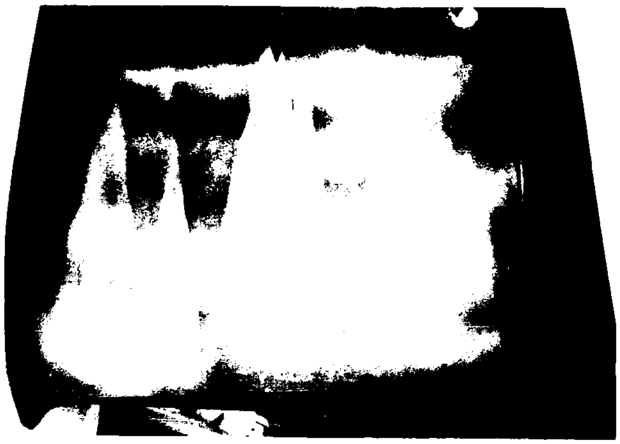


Fig. 16 - P.180 inboard wing flow visualization

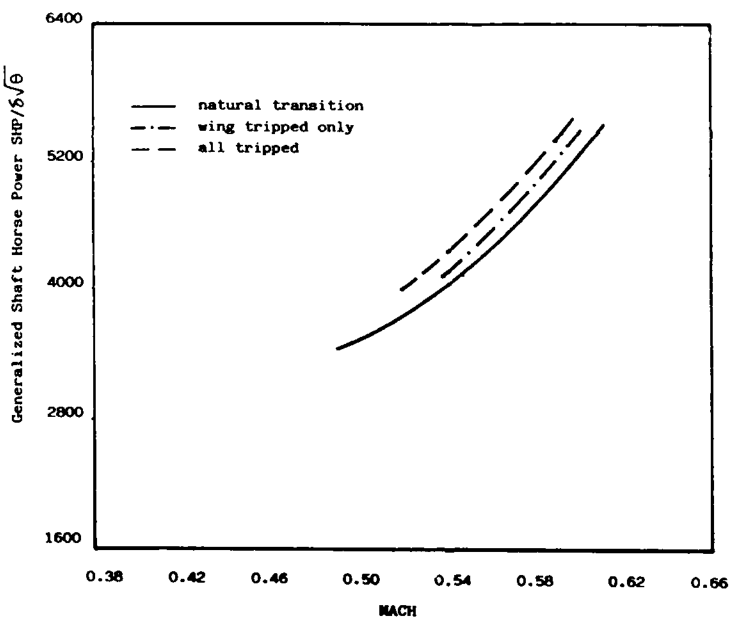


Fig. 17 - P.180 Effect of contamination on speed-power polar

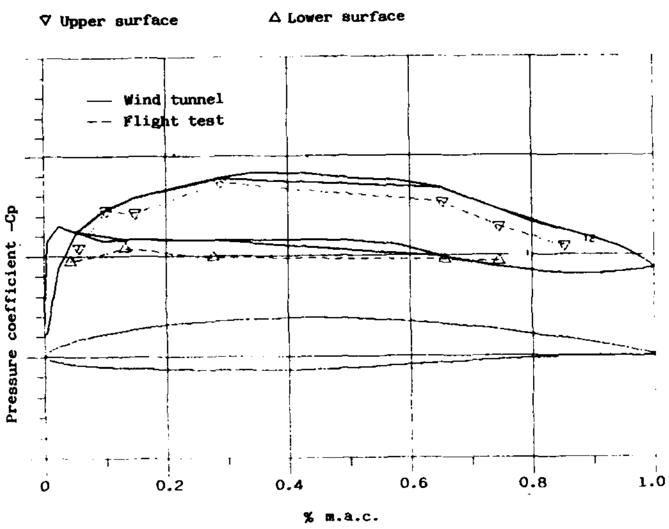


Fig. 18a - Pressure distribution on the forward wing at 20000 ft, 260 KIAS

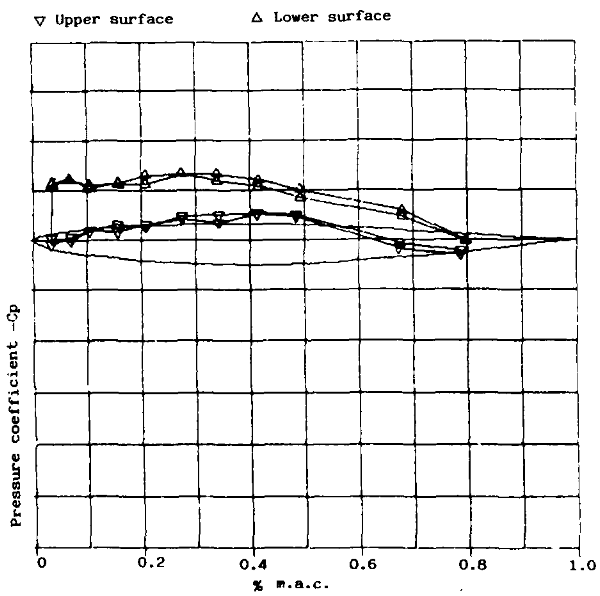


Fig. 18b - Pressure distribution on the horizontal tail at 20000 ft, 260 KIAS

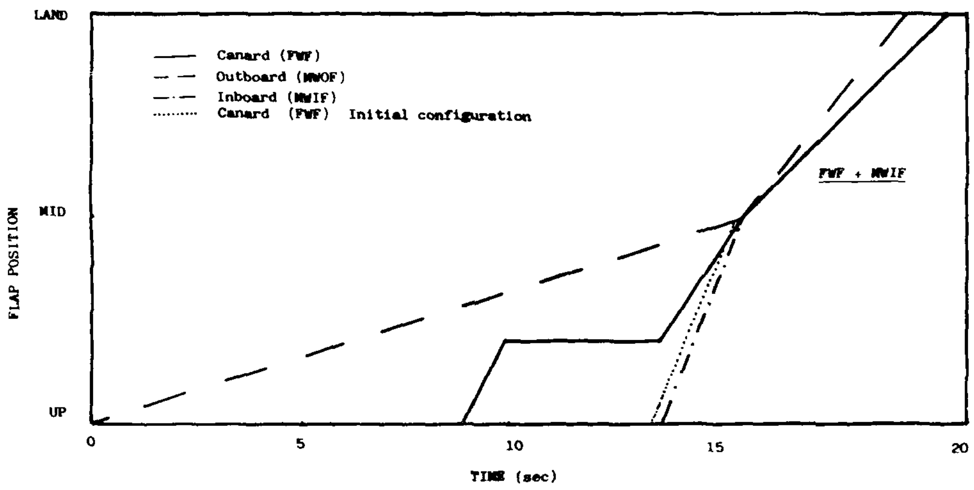


Fig. 19 - Flap schedule

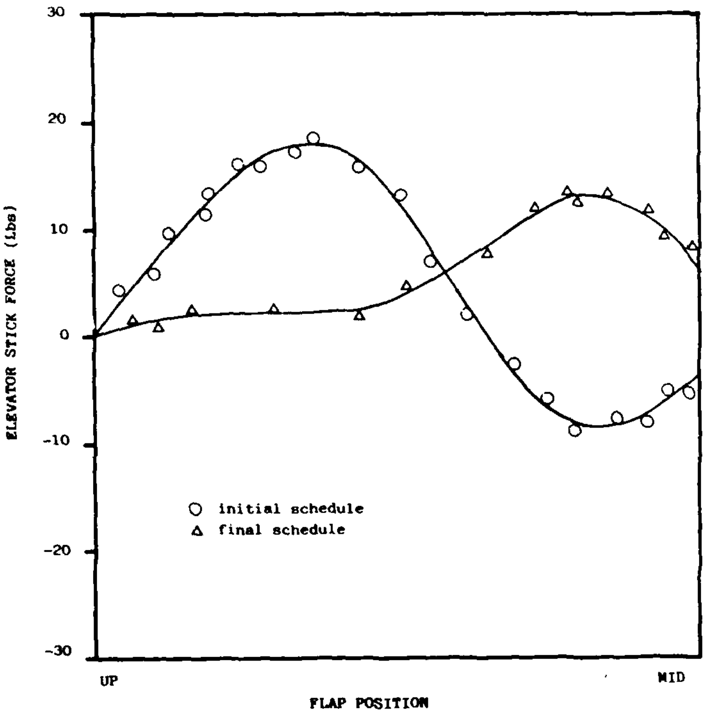


Fig. 20 - P.180 Stick force during UP-MID flap maneuver

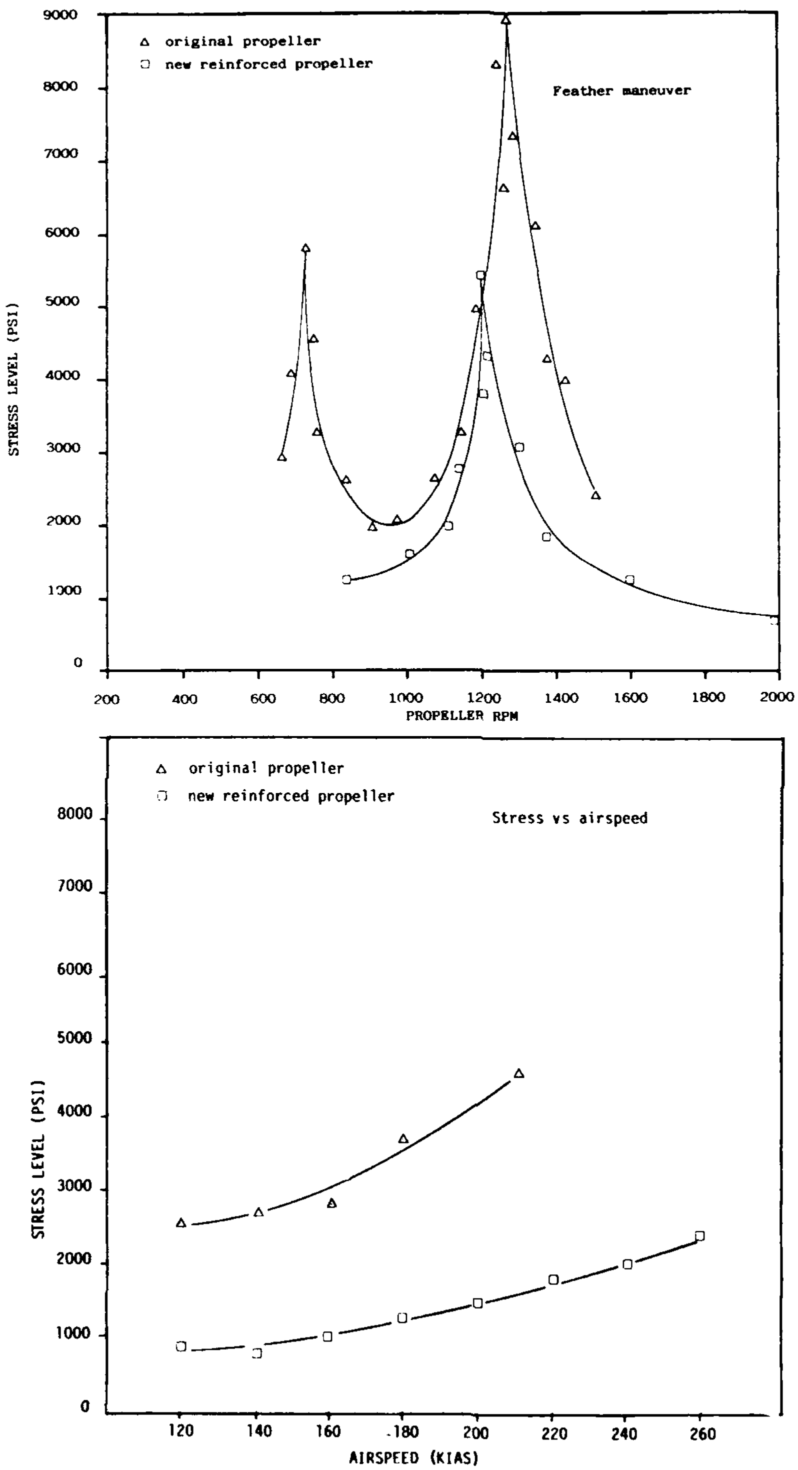


Fig. 21 - P.180 propeller stress

ESSAIS EN VOL A320

PARTICULARITES ET INNOVATIONS

par

Gérard GUYOT
Directeur du Département Essais en vol

AIRBUS INDUSTRIE
1 Rond Point Maurice Bellonte
BP 33 31707 Blagnac Cédex
FRANCE

L'Airbus A320, produit par AIRBUS INDUSTRIE, est le premier avion commercial à commandes de vol électriques et son cockpit révolutionnaire est également le plus avancé du monde. Malgré le pas technologique considérable que représente cet avion, la certification de la première version A320-100 a été obtenue après douze mois d'essais au cours desquels mille quatre cent quatre vingt (1480) heures de vol ont été effectuées sur 4 avions de développement.
Pour arriver à ce résultat remarquable, des moyens particuliers ont dû être mis en oeuvre :

- télémesure très puissante pour diminuer les temps d'exploitation
- système embarqué permettant de modifier en vol la configuration de certains calculateurs
- moyens permettant de vérifier que l'avion n'est pas sensible aux rayonnements électromagnétiques.

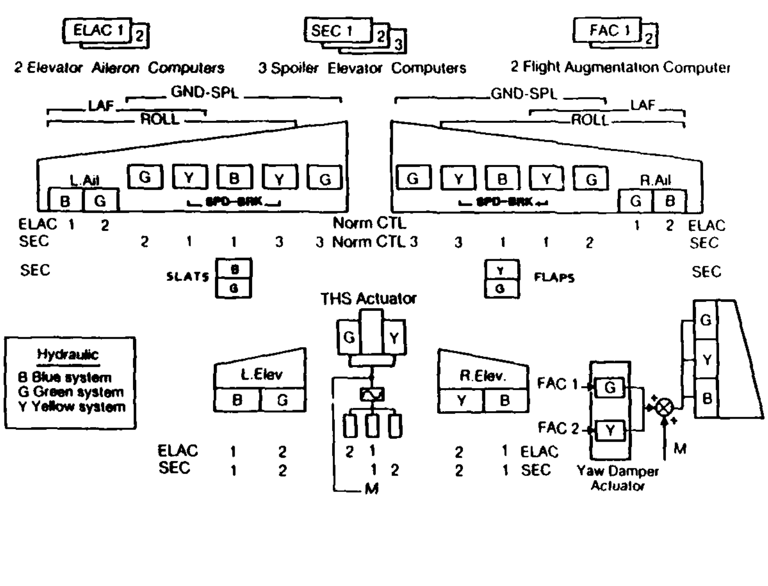
Il n'est pas nécessaire de parler de la télémesure qui est depuis quelques années un moyen d'essais classique, mais il faut par contre insister sur les moyens particuliers utilisés pour la mise au point et la certification de l'A320.

SYSTEME SPATIAL

Le système de commandes de vol électriques de l'A320 (EFCS : Electrical Flight Control Panel) est organisé autour de trois types de calculateurs :

- Deux ELAC (Elevator Aileron Computer) qui assurent :
  . le contrôle des ailerons
  . le contrôle normal de la gouverne de profondeur et du plan horizontal reglable.
- Trois SEC (Spoiler Elevator Computer) qui assurent :
  . le contrôle des spoilers
  . le contrôle en secours de la gouverne de profondeur et du plan horizontal réglable.
- Deux FAC (Flight Augmentation Computer) qui assurent le contrôle de la gouverne de direction.

FLIGHT CONTROL SYSTEM GENERAL ARCHITECTURE



Afin de pouvoir effectuer la mise au point des commandes de vol dans des délais très courts, le système SPATIAL a été créé et permet trois fonctions :

1) Emission de paramètres internes aux calculateurs sur bus, Essais en vol, permettant l'analyse du fonctionnement en cas de problèmes.

2) Envoi vers différents calculateurs d'ordres de changement de configuration (gains, valeurs de braquage, etc...)

3) Injection de sollicitations de différentes formes dans les calculateurs EFCS et AFCS (sinusoïdes, rampes, créneaux, balayages).

Huit types de calculateurs sont abonnés au système SPATIAL :

- SEC (3) Spoiler Elevator Computer
- FAC (2) Flight Augmentation Computer
- ELAC (2) Elevator Aileron Computer
- BSCU Braking System Control Unit
- FCDC (2) Flight Control Data Concentrator
- DMC (3) Display Management Computer
- FWC (2) Flight Warning Computer
- FMGC (2) Flight Management and Guidance Computer

Les configurations sélectables en vol sont préprogrammées dans les calculateurs et vérifiées systématiquement au simulateur avec des calculateurs EFCS identiques à ceux montés sur l'avion.
Naturellement, des précautions particulières ont été prises pour minimiser les risques d'erreur au moment des changements de configurations par l'ingénieur navigant et pour revenir rapidement à l'état initial des calculateurs en cas de problèmes.
La procédure de changement de configuration des calculateurs est la suivante :

- sélection du calculateur à modifier au moyen d'un clavier avec contrôle de l'opération par visualisation sur écran
- sélection du numéro de programme correspondant à la configuration demandée
- vérification de la configuration sur écran
- validation du numéro de programme si configuration correcte.

A ce moment-là, la nouvelle configuration est prête à être introduite dans le calculateur ; pour activer celle-ci, il faut encore :

- enfoncer le bouton poussoir correspondant aux calculateurs concernés
- enfoncer le bouton poussoir général d'activation.

En cas de problème de contrôle de l'avion après activation de la configuration d'essais, l'ingénieur navigant peut couper la commande générale d'activation, mais le pilote a également la même commande à sa disposition en cas d'urgence.
Lorsque le programme comporte des sollicitations automatiques au moyen du générateur de signaux, une simple action sur le manche latéral par le pilote suffit pour désactiver le générateur et interrompre ainsi les sollicitations ou le balayage en cours.

EXEMPLES D'UTILISATION DU SPATIAL

1) Mise au point EFCS :

a) dans la configuration initiale des aérofreins, le niveau de buffeting, jugé trop important, était dû à l'influence de l'écoulement aérodynamique sur l'empennage de l'avion lorsque le SPOILER interne était braqué :
   - le SPATIAL a permis d'essayer au cours d'un même vol différentes combinaisons de braquage des spoilers pour retenir celle donnant le meilleur compromis performance/faible niveau de buffeting.

b) à l'atterrissage, les essais ont montré que certains pilotes avaient tendance à mal contrôler la phase d'arrondi à cause de la loi normale de pilotage en facteur de charge, mal adaptée dans ces conditions.
   Une nouvelle loi de stabilité en assiette longitudinale a été rapidement mise au point grâce au SPATIAL et introduite manuellement par l'ingénieur navigant, avant de l'être automatiquement à 50 ft par un signal radio-altimètre.

2) Identification de l'avion :

Jusqu'à présent, les sollicitations sur les commandes de vol nécessaires à l'identification des divers coefficients aérodynamiques de l'avion sur ses trois (3) axes étaient effectuées manuellement par le pilote, ce qui conduisait à une certaine imperfection sur la qualité des entrées (créneaux, rampes, etc...) et sur leur répétitivité.
Par le SPATIAL et son générateur de signaux, les sollicitations sont effectuées automatiquement, ce qui garantit un meilleur accès aux fonctions de transfert des réponses de l'avion aux signaux d'entrée.

3) Réponse structurale

a) les lois de commandes de vol de l'A320 assurent une fonction de diminution des charges de la voilure en cas de rafale par braquage vers le haut des ailerons et des spoilers 4 et 5 (LAF : Load Alleviation Function).
Pour analyser la réponse de l'avion lors du braquage de ces surfaces et optimiser le fonctionnement du système, des signaux d'entrée avec différents gains ont été envoyés par le SPATIAL.

b) Le générateur de signaux du système SPATIAL permet également d'effectuer des braquages de gouverne d'une manière sinusoïdale, avec balayage en fréquence, pour les essais d'excitation forcée.
Pour effectuer les essais d'excitation forcée en dehors du domaine opérationnel VMO/MMO le SPATIAL a été utilisé également pour inhiber la protection grande vitesse qui existe sur A320.

4) Certification des performances

Sur un avion conventionnel, un certain nombre de vitesses opérationnelles ne doivent pas être inférieures à une vitesse minimale qui est définie par un facteur les associant à la vitesse de décrochage VS_{MIN} (par exemple, V2 = 1.2 VS, V_{REF} = 1.3 VS, etc...)

L'avion A320, grâce aux commandes de vol électriques, a une protection d'incidence qui l'empêche de pouvoir aller jusqu'au décrochage.
Si les vitesses opérationnelles, avaient été associées à la V_{MIN} démontrée en vol avec la protection d'incidence, l'A320 aurait été de ce fait pénalisé par rapport aux avions conventionnels.
Il a donc été admis par les Autorités de Certification Européennes et Américaines que la limite de la protection d'incidence pouvait être décalée vers le haut pour pouvoir démontrer la vitesse minimale Vs1g qui sert de référence pour les vitesses qui lui sont associées.
Ce décalage est introduit en vol d'une manière très rapide au moyen du SPATIAL avant d'effectuer les essais.

5) Introduction d'informations nécessaires aux lois de pilotage

Les lois de pilotage utilisent des gains qui sont fonction du centrage qui est calculé automatiquement par le FAC.
Au cours de la phase de la mise au point, les calculs n'étant pas encore optimisés, les valeurs de centrage étaient introduites dans les calculateurs au moyen du SPATIAL.

PROTECTION CONTRE LES RAYONNEMENTS ELECTROMAGNETIQUES

Tous les avions sont exposés à la foudre et aux rayonnements électromagnétiques en général. Les commandes électriques ont été introduites progressivement sur les avions AIRBUS avant d'arriver à l'A320 et des précautions importantes ont été prises depuis le début au niveau de la définition et de l'installation de ces commandes pour qu'elles soient immunisées contre les rayonnements électromagnétiques.

Des essais particulièrement sévères ont été effectués sur l'A320 pour vérifier cette immunité.

Trois types principaux de rayonnements électromagnétiques sont identifiés :

- transitoires électriques ou électrostatiques, comprenant la foudre en particulier.
- champs électriques et magnétiques de la génération électrique de bord (400 hz).
- fréquences radio-électriques, comprenant émission radar, radio-diffusion et télévision, entre autres.

En plus de l'application du règlement européen JAR en matière de protection, AIRBUS INDUSTRIE a démontré que l'avion pouvait résister aux agressions les plus sévères considérées par le Département de la Défense des Etats-Unis qui a été consulté par la FAA (EEAC-Décembre 87), bien qu'il ait été trouvé discutable d'appliquer des conditions d'essais plus sévères parfois que celles du NATO pour un avion civil.

Trois types d'essais ont été effectués :

1) Essais en laboratoire :

Les signaux électriques sont introduits au niveau des câblages électriques et des calculateurs de l'avion sur un ban d'essais qui simule l'installation des éléments hydromécaniques et électriques des commandes de vol.

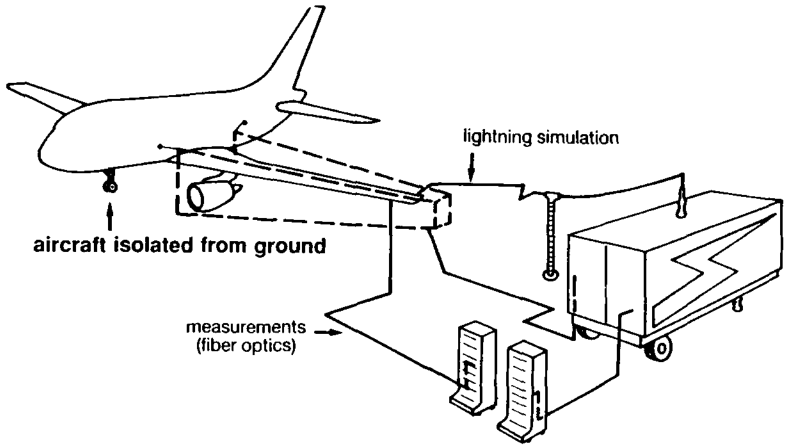
2) Essais au sol sur avion :

Des essais de simulation de foudre sont effectués en injectant des signaux électriques allant jusqu'à 200 KA au niveau de la voilure et de la nacelle, avec mesure des courants induits en différents points du câblage avion.

Les mesures sont effectuées à un niveau inférieur à celui des conditions réelles de la foudre, puis extrapolées aux conditions les plus sévères estimées.
La baie de mesure est isolée électriquement de l'avion et reçoit ses informations par fibres optiques.

A320 -EM COMPATIBILITY

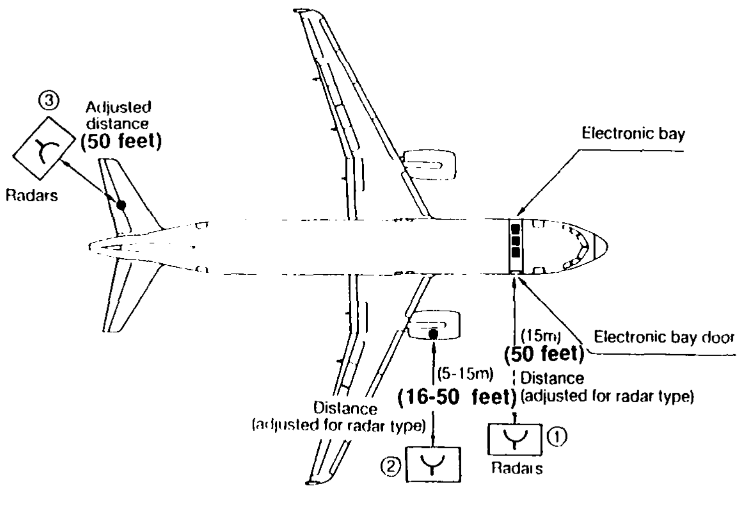
-Test Installations.



L'avion est également exposé, dans un centre d'essais de la Marine Nationale, aux champs de radars et d'émissions radio-électriques HF d'un niveau très élevé.

A320 -EM COMPATIBILITY

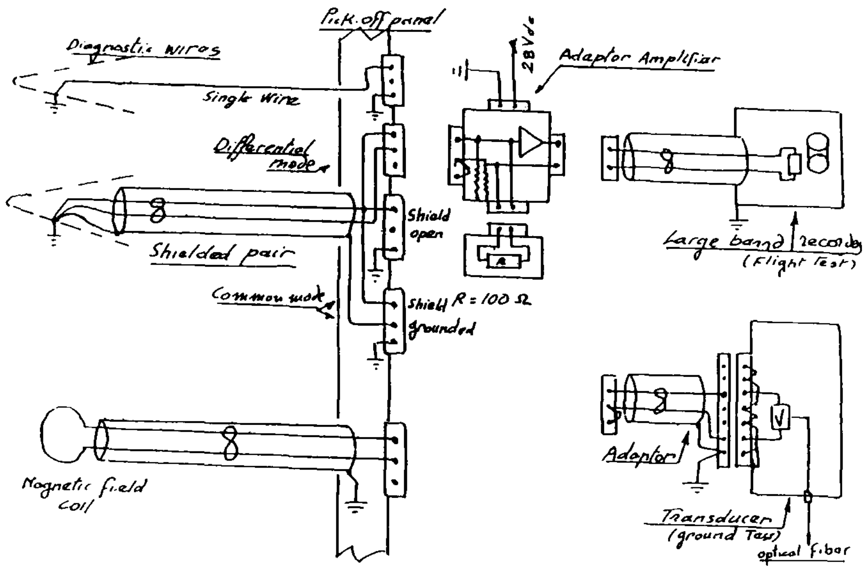
-A 320 Tests



3) Essais en vol :

Quinze canaux d'enregistrement sont réservés à la mesure du champ magnétique en six (6) points de l'avion et de neuf (9) paramètres électriques. Ils sont analysés chaque fois que l'avion est foudroyé pendant les essais en vol, ce qui arrive fréquemment en particulier au cours de la recherche de givrage.

A.320 – LIGHTNING INDUCED TRANSIENT MEASUREMENTS
DIAGNOSTIC WIRES INSTALLATION (TYPICAL)



CONCLUSION

L'évolution de la technologie associée à des délais de certification qui doivent être de plus en plus courts pour minimiser les coûts de développement, obligent les Bureaux d'Etude et les Services d'Essais en Vol à faire preuve d'imagination pour pouvoir faire dans ces délais les essais nécessaires pour garantir la sécurité à la mise en service des avions.

Le système SPATIAL a parfaitement rempli son rôle qui sera encore étendu pour les programme A330/A340.

Sur les avions futurs, la protection contre les rayonnements électromagnétiques restera une préoccupation légitime et les Autorités de Certification doivent définir des critères clairs et réalistes, sans perdre de vue les méthodes de démonstration qui doivent être éprouvées pour éviter des improvisations tardives au cours des programmes d'essais.

THE EXPERIMENTAL AIRCRAFT FLIGHT TEST PROGRAMME

by

R.A. Hartley
British Aerospace PLC
Military Aircraft Division
Warton Aerodrome
Preston PR4 1AX
United Kingdom

SUMMARY

The Experimental Aircraft Programme has grown out of the studies carried out over the last decade to investigate the needs for the next generation of fighter.

These studies resulted in the definition of a large number of new technologies for which it was recognised that there would be significant benefits to be gained by their integration into a demonstrator aircraft.

This paper describes the objectives and progress of the flying programme for this demonstrator and how it has lead on to the European Fighter Aircraft.

1. INTRODUCTION

In the late nineteen seventies, British Aerospace in common with many other Military aircraft companies, was analysing and assessing the fighter aircraft needs for the Air Forces in the nineties and the next century. From this it was determined that a major requirement would be for a highly manoeuvrable aircraft for close and medium range air combat, and with a secondary, but effective, capability for air to surface battlefield support.

This would entail a lightweight single crew aircraft capable of carrying a wide variety of stores and twin engined to give a high level of survivability.

To produce this within an acceptable cost would require the extensive use of new technologies, many of which were still in the concept or early stages of development.

For example:

- o The manoeuvrability requirements would need an unstable aircraft controlled by a digital fly by wire system

- o To keep the weight and size of the aircraft to a minimum and ensure as small as possible a target for detection and acquisition by enemy weapons; the extensive use of new materials and advanced structural design concepts would be required.

- o For the weapon system to be capable of management by a single crew would need extensive automation, requiring an interactive network of computers, controlling the aircraft systems, sensors and weapons. These would provide the pilot with only essential information, via multifunctional controls and displays and thus relieve him of all but the essential tasks.

It was also declared UK government policy that as far as possible all future major military projects should be in collaboration with other European partners. As there was already a highly successful collaboration in progress on the Tornado it was a natural tendency for these partners which included BAe, to consult on this requirement. This rapidly evolved into combined studies to define an aircraft to meet a perceived common European requirement.

In 1982 the Agile Combat Aircraft (ACA) was announced as a collaborative design programme involving AIT, BAe and MBB. At the time the Governments and MOD's in Europe were still defining their requirements for the aircraft of the 1990's, therefore this programme was in the main privately funded by the companies involved. It was however foreseen that to establish a good base for the development of any future aircraft it would be necessary to build a demonstrator, to pull together all of the advance technology areas which have been referred to and thereby shorten any prototype development phase. However, the manufacture of such an aircraft was considered to be beyond the private resources of the Companies without Government assistance.

2. THE EXPERIMENTAL AIRCRAFT PROGRAMME

In June 1983 a partnership contract was signed between BAe and the UK Government to build a Demonstrator aircraft to pull together and investigate as a whole the technologies required for a future agile fighter.

This recognised that AIT and MBB plus a number of European equipment manufacturers were also partners in the project. Also it was intended that a second demonstrator should be built by MBB but funded in Germany. In the event the funding for the second demonstrator could not be made available and this led to the withdrawal of MBB from the project at the end of 1983.

This partnership, called the Experimental Aircraft Programme was essentially to design and manufacture an aircraft, and within that it would integrate the required advanced technologies and developed them into a viable complete concept. However, it also recognised that to follow, it would be necessary to assess and prove the viability of these in flight. Therefore a number of objectives were defined for a flight assessment programme.

- o A progressive increase and clearance of the flight envelope during which the air vehicle would be assessed to establish confidence in the overall design including the use of new materials.

- o Assessment of the advanced aerodynamics by a progressive increase in the manoeuvring envelope to identify the high incidence characteristics and demonstrate the agile handling qualities including:-

  High angle of attack including the determination of engine intake behaviour and its effect on engine performance.
  Use of a departure prevention system.
  The effectiveness of a control system design for non linear interference between a canard and wing.

- o Measurement of performance during transonic acceleration, supersonic flight and on the airfield, particularly during landing; and the confirmation of the design for low drag weapon carriage.

- o Determination of the low level ride qualities.

- o Demonstration of active control technology by optimisation of the FCS during the above aerodynamic assessments.

- o Demonstration of the use of digital control techniques by the safe and satisfactory operation of the aircraft systems and avionics controlled by 1553 data busses.

- Demonstration of a modern electronic cockpit design using multifunction controls and displays with a wide angled head up display and the use of voice to provide the pilot with information and warnings.
- Demonstration of stealth design techniques for low Radar and Infra Red signatures.
- Demonstration of integrated digital engine control.

3. AIRCRAFT DESCRIPTION

The aircraft designed and built to meet these objectives was advanced both in its structure and equipment (Fig.1).

The airframe is an inherently very unstable configuration with moving foreplanes (canards) and a cranked delta wing with full span trailing edge flaperons and extending leading edge droop.

Carbon fibre composites are widely used for the manufacture of the forward fuselage, foreplanes and the wings, which use an advanced manufacturing process of co-bonding of the skin to the internal wing structure.

The unstable configuration requires a flight control system (FCS) using active control technology in a quadruplex, full authority computer control system without mechanical back up (Ref.1). In its primary mode it is designed to provide full "carefree" handling. This system has been developed from the experienced learned during the Jaguar Fly By Wire programme (Refs 2 & 3).

The utilities systems e.g. fuel, hydraulics etc, of the aircraft are monitored and controlled by a microprocessor based utilities systems management system (USMS). The processors and other system constituents communicate via a MIL-STD-1553B data bus. Information on this data bus also includes equipment/system failure codes which are stored in a single unit for subsequent access by the groundcrew via a Maintenance Data Panel (MDP).

Avionic equipment e.g. comms, navigation etc also communicate between each other and the cockpit displays and controls via a separate 1553B data bus. All the data buses are interfaced with one another to exchange information as required (see Fig.2) and pass it to the pilot.

The cockpit is ergonomically arranged with three multifunction displays (MFD) and controls arranged on a main flying panel (see Fig.3). Each MFD can present in full colour selected displays of flight information e.g. attitude, position etc, or systems status information. Two typical displays are shown in Figs. 4 & 5.

Surrounding each display are 24 keys which are mixed with a few "hard" i.e. fixed function keys and the remainder as "soft" keys having a multifunction capability depending on, and identified by, the display selected. Above is a wide angled head-up display (HUD) and below and around are "get you home" (GUH) instruments with essential information in the event of a failure associated with the main avionics data bus. Built in the left hand glare shield is a manual data entry facility, with selection and moding keys for the control of communications and navigation inputs. The right hand glare shield incorporates associated indicators.

Above the glare shield are temporary fit flight test instruments providing the normal indications and controls associated with a high incidence handling programme. The side consoles also are a mixture of switches and controls associated with Flight Test instrumentation and facilities and the back up functions required by the aircraft as a new technology demonstrator. However, the control stick and throttle is designed on the principle of HOTAS (Hands-On-Throttle-And-Stick) concept with a multiplicity of switches for all essential in-flight functions.

The concept of the demonstrator programme entailed the use of an existing high performance engine. For this the RB199 - Mk.104D was selected. This is in use in the Tornado ADV and the only major modification required, was the removal of the thrust reverser system. The engine is digitally (DECU) controlled and this was directly interfaced with the aircraft USMS data bus. To provide high quality (low distortion) air flow to the engine at high angles of attack, two chin mounted intakes are provided. These have a fixed throat but with a variable lower lip, controlled by the FCS, for high incidence and supersonic flight.

4. THE EAP DEMONSTRATOR FLIGHT TEST PROGRAMME

To make an overall assessment of this aircraft and the previously defined objectives, including the development of "carefree" handling and expansion of the envelope to supersonic speeds, a programme of approximately 200 flights over a period of two years was proposed (Fig.6). That meant we had to achieve an overall flying rate of in excess of 8 flights per month.

To achieve that it was recognised that a highly serviceable aircraft and the minimum of lay-ups for modifications including software updates, would be required. In the event a number of software updates e.g. FCS Carefree Handling software were identified before first flight. These were planned so as to be available with the minimum of disruption to the flying programme by extensive rig assessment and the minimum of on-aircraft tests.

The data handling and analysis would also have to keep up to avoid the risk of abortive or wasteful flying. For this the already existing Warton Flight Test facilities of telemetry, which was to be used on all programmed flights from Warton, and data reduction were considered adequate but with the extension and development of real time computer links for flight mechanics and loads analysis. The aim was for the objective of the flight to be cleared by basic analysis before the pilot debrief. In the main this was achieved (in many cases before the pilot had left the cockpit).

For this it was essential for the aircraft to have a complete instrumentation system. This consists of a PCM processing unit handling data from the three aircraft data busses, the FCS, Avionics and Utilities (USMS) plus direct FTI sourced parameters. On average approximately 600 parameters are recorded each flight with a typical mix as follows:

FTI	sourced	200
FCS	"	125
USMS	"	175
Avionic	"	100

at sampling rates between 1 and 64 samples per sec. Additionally up to 80 FM/FM multiplexed channels are available in 4 selectable groups of 20.

All of the above are recorded on-board and telemetered to the Flight Test ground station.

Up to 10 high frequency channels are also available for on-board recording in addition to the above.

5. INITIAL ASSESSMENT

The basic demonstrator design manufacturing programme was achieved in just under three years and the aircraft rolled out at a ceremony at Warton on 16th April 1986.

Following that the normal pre-flight activities of systems checkout, engine runs and taxy trials took place leading up to a first flight on 8th August 1986.

This occasion marked the end of the contract between BAe and the UK government. The programme now continuing as a collaborative Industry programme between BAe, AIT and the European equipment manufacturers.

The first flight by Dave Eagles (Executive Director of Flight Operations) demonstrated the achievement of a successful design and the meeting of the manufacturing objectives, by carrying out a detailed handling assessment as well as the normal shakedown functions concerning engines systems etc. Mach 1.1 was achieved with a defect free landing after a flight duration of 67 mins.

This was the start of an intensive flight test programme to expand the aircrafts manoeuvring envelope. Due to the need for having an accurate confirmation of the FCS air data for incidence and sideslip before the fully scheduled control system could be invoked; the first period of flying had to take place with the system in the reversionary mode. The first task was to measure the air data and to confirm the basic handling and manoeuvring envelope as well as the capability of the engines, systems and avionics. This was also coupled with flying the aircraft at a precautionary mid, but still unstable c of g of 34.6% s.m.c.

The handling envelope was then expanded, to provide a rolling clearance between 0 and 3g and assess the variety of manoeuvres required in air combat or for demonstration purposes. This was achieved in 20 flights and over a period of as many days. In the same time the flight envelope was expanded to Mach 1.33/450kts and surge free engine handling confirmed up to 35000ft (Fig.7).

This provided more than adequate confidence for a participation and an extensive demonstration of the aircraft at the Sept'86 SBAC Show at Farnborough. The show, flown by Chris Yeo (Chief Test Pilot) was completed without problem with the aircraft performing on each of the designated days and showing its exceptional handling qualities. It also demonstrated the very high standard of serviceability possessed by the aircraft from it's first flight and which continues up to today. On returning from Farnborough a total of 38 flights had been completed in the 35 days from first flight.

The handling envelope expansion was then continued together with commissioning and clearance of an Emergency Power System and engine assessment at high altitude, to provide the information and envelope clearance for the start of a full high incidence programme to clear"Carefree" handling with the full FCS (see Fig.8). Over the same period an initial measurement of aircraft performance was completed together with a qualitative appraisal of the low level ride qualities. The latter was completed by flying down to 300ft on local low flying routes over mountainous terrain. The ride qualities were considered to be "surprisingly good" considering the low wing loading of the delta configuration and the fact that we had no deliberate ride control features yet installed in the FCS.

By now a total of seven test pilots had flown the aircraft, 5 from BAe and 2 from AIT and all had found the introduction a very easy and pleasant experience.

In November 1986, 3 months after first flight the aircraft was laid-up for a work programme to prepare for a high incidence handling assessment· the essential prerequisite to a carefree handling clearance at subsonic airspeeds. This involved the fitting of a spin recovery parachute as well as a FCS software update and moving the c.g aft to its design point of 36.5% smc.

The FCS software update was necessary to bring the full system into operation but additionally a precautionary "spin recovery" mode was also introduced to work in association with the spin recovery parachute. This mode was for use only in the event of a departure from controlled flight. It permitted a "direct" stick to flying controls link allowing a controlled recovery of the aircraft in conjunction with the spin recovery chute.

6. CAREFREE HANDLING

Following this preparation, the aircraft flew again in April 1987 to assess the aircraft at high angles of attack and clear carefree handling. This was achieved in a period of one month, 25 flights, including the precursors of assessing the spin recovery parachute and other associated system testing. The result was a fully carefree handling clearance between 200kts and 400kts/0.9M. Below 200kts the general clearance is still "carefree" but snatch pulls, in particular re pulls, should be avoided to prevent an incidence overswing.

The outstanding tasks associated with the major aspects of the objectives were then completed prior to the aircrafts attendance at the Paris, Le Borget Airshow in June. During this show the aircraft achieved its 100th flight birthday and continued its by now well established record of high reliability and demonstration of exceptional manoeuvrability.

On return from Paris the major part of the objectives for this phase had been achieved. This was marked by an invitation to the UK and Italian MOD to send their pilots to "evaluate" the aircrafts features of new technology. An assessment was performed by three pilots, 2 UK and 1 Italian over 4 flights in a period of eight days, which included all ground school, briefing and the actual assessment flights.

Following this the aircraft was laid up again; this time to prepare for new tasks aligned with assessment directed towards the design of the European Fighter Aircraft - EFA.

It had achieved a total of 116 flights and 79 flying hours in a period of 11 months. This represented an average flying rate of 10.5 flts per month overall, but 18 flights per month for the flying periods between major lay-ups (Fig.9).

A total of 10 test pilots had flown the aircraft made up of:

Company pilots	BAe	5
	AIT	2
Official pilots	UK	2
	Italy	1

The aircraft had completed the major objectives and envelope conceived by the initial proposal; in about half the time planned, and in the process had proven the following features, among others, for future use:-

- o There were no major problems inhibiting the design or building of an aircrafts critical structure i.e. wing foreplane and fuselage using Carbon fibre composite.
- o The combination of foreplane/delta wing aerodynamics with a quadruplex digital FCS was completely successful in controlling a fully unstable aircraft and providing excellent performance with ideal handling qualities.
- o Carefree handling had been demonstrated as not only a worthwhile but attainable feature.
- o The use of microprocessors connected by digital data busses were shown to be viable and the indications are that these will provide a much higher order of reliability and maintainability over conventional engineering and electrical systems.
- o The cockpit was liked by all the pilots who flew the aircraft and, although not fully developed, the concepts of moding, layout and control have been accepted for the next generation of fighter aircraft.
- o Although a basically "off the shelf" engine, the RB199-Mk104D engine intake interface showed an excellent combination giving carefree, surge free handling throughout the envelope.

7. FUTURE PROGRAMME

The aircraft is still engaged on technology development with trials progressing to further expand its envelope, particularly in the direction of full carefree handling and development of the FCS to expand this into the supersonic areas. It is however now committed to do this in association with the European Fighter Aircraft (EFA) development programme and many of its trials are being arranged to provide information in direct support to EFA development. These will include trials to expand the use of new controls and systems and expand the capability of the cockpit by further automation to reduce pilot workload e.g. use of Direct Voice Input. It will also be used to develope techniques to reduce mass and/or improve confidence in the assessment of prototype aircraft and the clearances processes required for Release to Service.

For the latter the flight envelope has been further expanded (Fig.10) and the aircraft is now well forward in the development of a technique for full scale measurement of aerodynamic loads by pressure plotting.

A number of surface pressure transducers have been progressively attached to the left wing, the fin and major areas of the fuselage (Fig.11). These have been used to measure the pressure distribution over the aircraft in steady flight and during specific manoeuvres. Comparisons are then made with these and the measurements from wind tunnel results. With this the aerodynamic load prediction methods can be refined to a high degree of accuracy and confidence, enabling mass savings from more accurate structural design and reduced reserve factors. This technique is also .o be fully developed as a clearance technique to be use on aircraft e.g. EFA with highly complex structural load paths, where strain gauging would be extremely difficult and costly, if not impossible, enabling the loads to be validated and the aircraft cleared into service.

The trial commenced by introducing only a limited number of transducers plus the associated surface wiring and multiplexor units, to check their accuracy and repeatability. This highlighted a number of problems, the major being the overall reliability of the transducers when being used in a flight trials environment. The failure rate initially experienced was completely incompatible with the flying rate required, and of course attainable by EAP. Concerted efforts by our Instrumentation department and the manufacturers isolated the problem as predominately due to freezing moisture and improved transducers have been introduced. A further difficulty was keeping the surface wiring stuck to the aircraft. It was found that most tape adhesives failed at approx. -50°C ceasing to hold the wires to the aircraft surface. A comprehensive review of commercially available tapes and adhesives found a silicon based adhesive which was good down to at least -60°C and this is now being used.

As can be seen this stage of the next phase of the Demonstrator programme is only in its infancy, with around 30 flights planned to perform a complete loads survey over the full EAP flight envelope using nearly 400 transducers attached to the wing, fin and fuselage.

These flights will be integrated with other aforementioned tasks, making a total further programme of more than 100 flights, involving a major update of the FCS and a number of major avionic improvements. These will take EAP into the 1990's and probably overlap with the first flights of the EFA prototype.

REFERENCES

[1] Herr J. Kaul, Sig. F. Sella, Mr M.J. Walker.
"Flight Control System for the Experimental Aircraft Programme" (EAP) Demonstrator Aircraft. AGARD 65th FMP Symposium. Toronto 1984 "Active Control Systems".

[2] Mr T.D. Smith, Mr C.J. Yeo, Mr R.E.W. Marshal.
"Ground and Flight Testing on the Fly-By-Wire Jaguar, Equipped with a Full Time Digital Integrated Flight Control System." AGARD 35th Guidance & Control Panel Symposium. Portugal October 1982 "Advances in Guidance and Control."

[3] Mr J.R. Nelson, Mr T.D. Smith.
"Improved Combat Performance using Relaxed Static Stability and a Spin Prevention System (FBW Jaguar)." AGARD 68th FMP Symposium. Istrana AFB Treviso, April 1986 "Improvement in Combat Performance for Existing and Future Aircraft."

ACKNOWLEDGEMENT

Any mechanical vehicle and any test programme is only as good as the people who make it happen. The outstanding success of the Demonstrator aircraft and rapid achievement of the programmes objectives is therefore due to those who did a superb job in the initial design, engineering and planning and who have continued to give their support to the programme.

Principal Features of the EAP

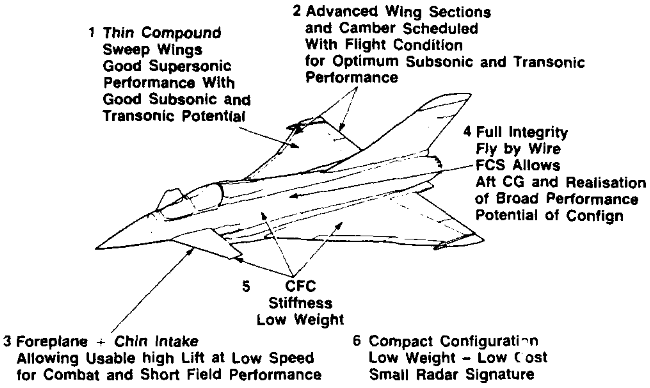


FIG.1

EAP – Overall Systems Architecture

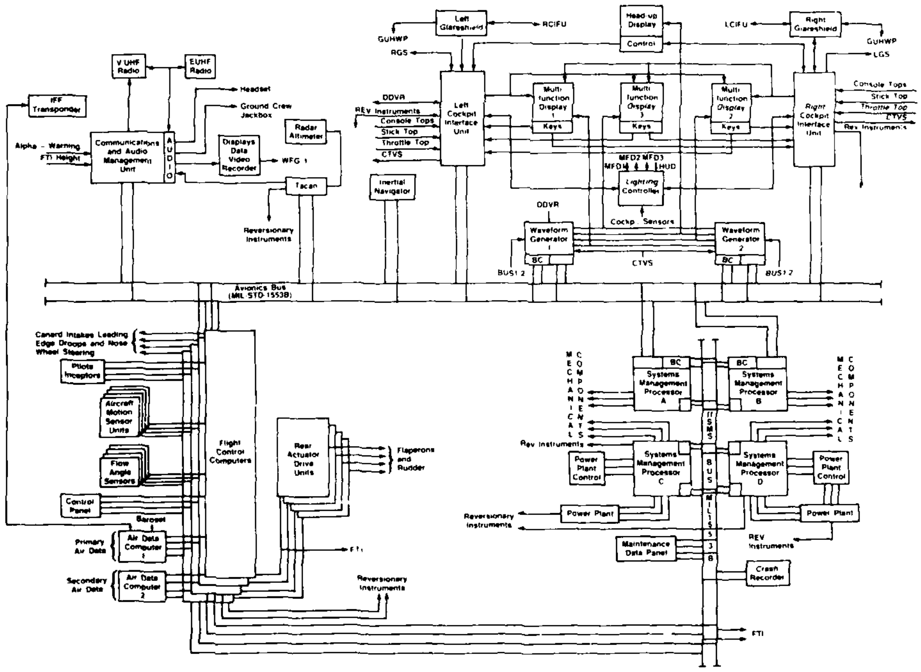
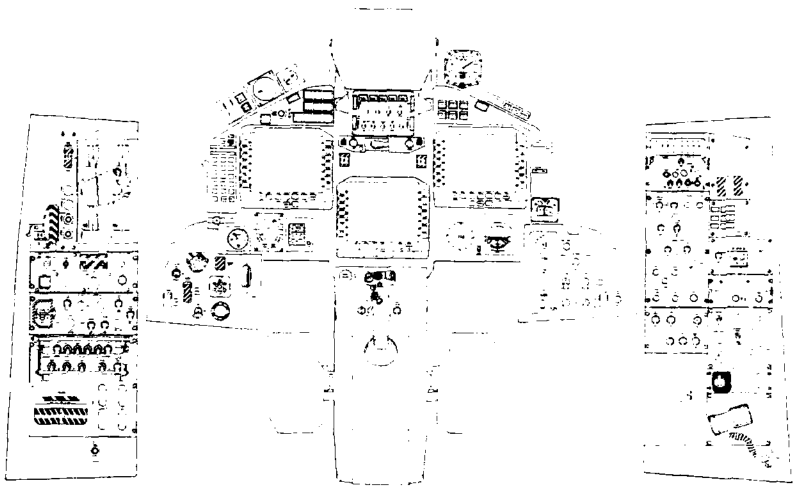
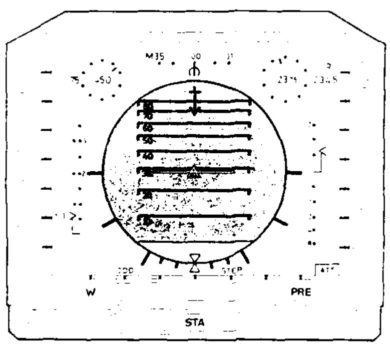


FIG.2



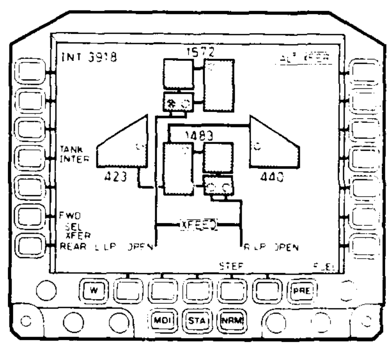
EAP Cockpit Layout

FIG.3



Multifunction Display Attitude Format

FIG.4



Multifunction Display Fuel Format

FIG.5

Experimental Aircraft Programme

First Flt		200 Flts
Aircraft Build and Ground Test	Initial Handling and Engineering Assessment Air Data Calibrations (FCS Reversionary)	High Incidence and Carefree Handling Envelope Expansion (Full Flight Control System)
EAP Phase 1	Phase 2	

Flight Test Plan

FIG.6

EAP – Flight Envelope After 12 Flights

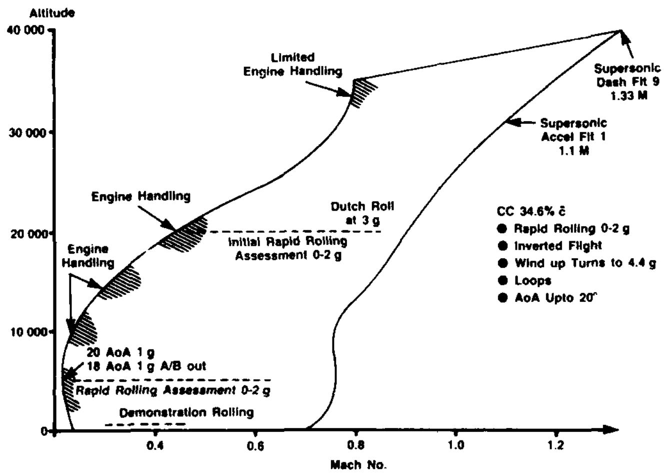


FIG.7

EAP – Flight Envelope After 50 Flights

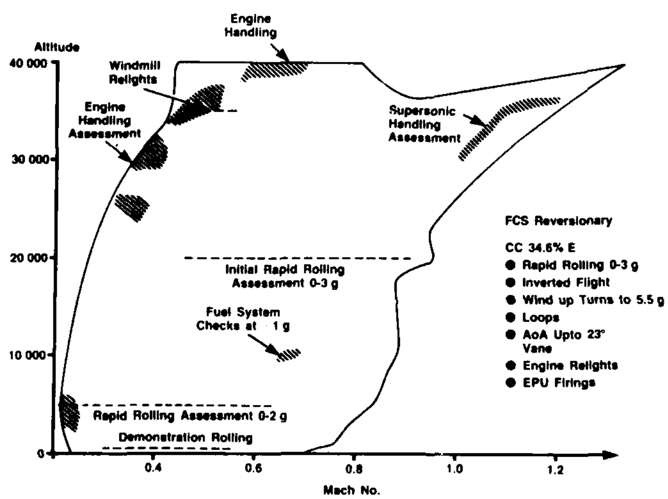


FIG.8

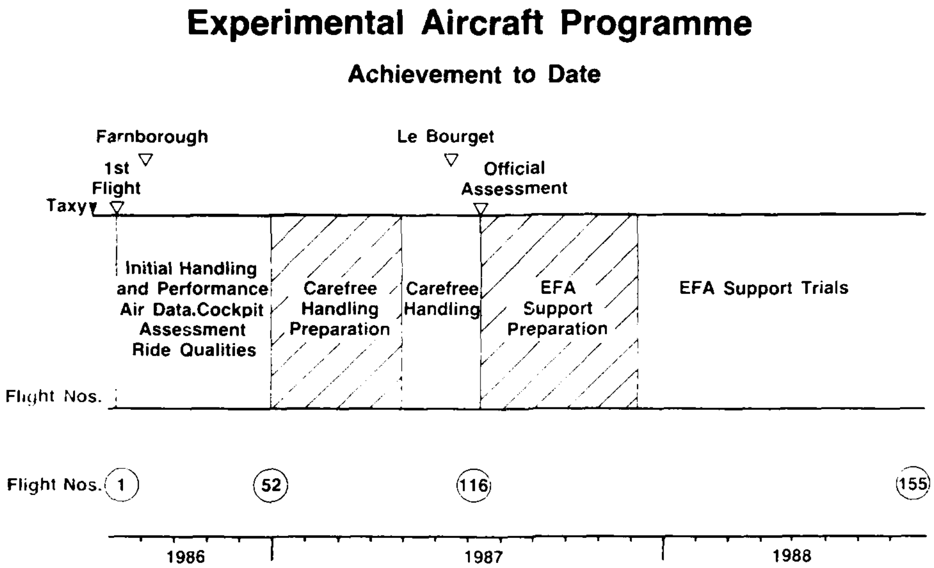

Experimental Aircraft Programme
Achievement to Date
Farnborough
1st Flight
Taxy
Le Bourget
Official Assessment
Initial Handling and Performance Air Data.Cockpit Assessment Ride Qualities
Carefree Handling Preparation
Carefree Handling
EFA Support Preparation
EFA Support Trials
Flight Nos.
Flight Nos.
1
52
116
155
1986
1987
1988

FIG.9

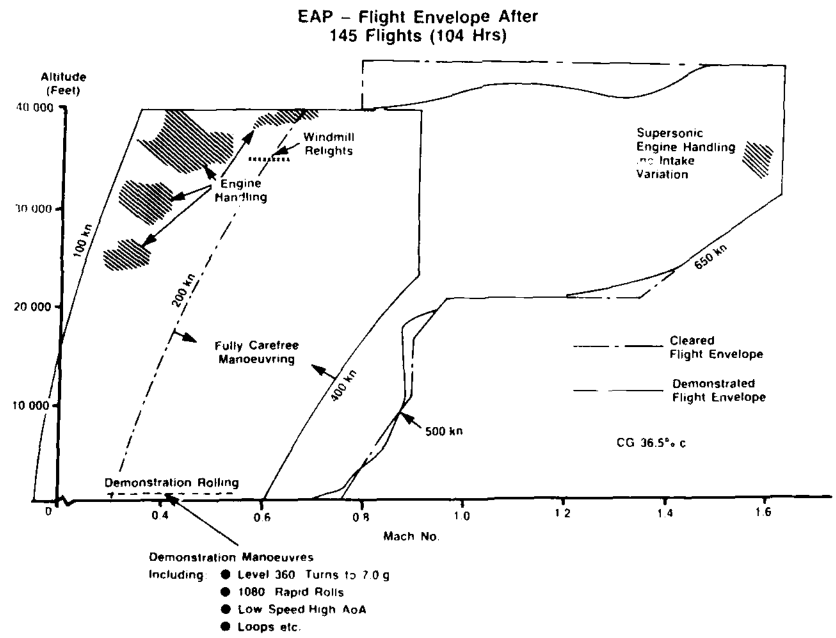

EAP – Flight Envelope After 145 Flights (104 Hrs)
Altitude (Feet)
40 000
30 000
20 000
10 000
0
Windmill Relights
Engine Handling
100 kn
200 kn
400 kn
500 kn
650 kn
Fully Carefree Manoeuvring
Supersonic Engine Handling Intake Variation
Cleared Flight Envelope
Demonstrated Flight Envelope
CG 36.5% c
Demonstration Rolling
0.4
0.6
0.8
1.0
1.2
1.4
1.6
Mach No.
Demonstration Manoeuvres Including
Level 360 Turns to 7.0 g
1080 Rapid Rolls
Low Speed High AoA
Loops etc.

FIG.10

EAP – Full Flight Loads Survey

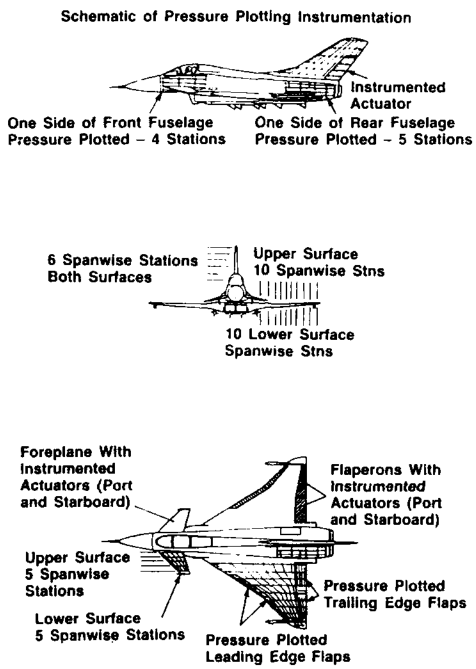


FIG.11

REAL-TIME FLIGHT TEST ANALYSIS AND DISPLAY TECHNIQUES FOR THE X-29A AIRCRAFT

John W. Hicks
Aerospace engineer
and
Kevin L. Petersen
Chief, Vehicle Technology Branch
NASA Ames Research Center
Dryden Flight Research Facility
P.O. Box 273
Edwards, California 93523-5000
U.S.A.

SUMMARY

The X-29A advanced technology demonstrator flight envelope expansion program and the subsequent flight research phase gave impetus to the development of several innovative real-time analysis and display techniques. These new techniques produced significant improvements in flight test productivity, flight research capabilities, and flight safety.

These techniques include real-time measurement and display of in-flight structural loads, dynamic structural mode frequency and damping, flight control system dynamic stability and control response, aeroperformance drag polars, and aircraft specific excess power. Several of these analysis techniques also provided for direct comparisons of flight-measured results with analytical predictions. The aeroperformance technique was made possible by the concurrent development of a new simplified in-flight net thrust computation method. To achieve these levels of on-line flight test analysis, integration of ground and airborne systems was required. The capability of NASA Ames Research Center, Dryden Flight Research Facility's Western Aeronautical Test Range was a key factor to enable implementation of these methods.

NOMENCLATURE

AR	analog reversion (flight control system mode)
BFF	body-freedom flutter
c.g.	center of gravity
C	loads calibration coefficient
C_D	coefficient of drag
C_L	coefficient of lift
D	aircraft drag, lb
DR	digital reversion (flight control system mode)
FCS	flight control system
F_{ex}	excess thrust, lb
F_g	gross thrust, lb
F_{np}	net propulsive force, lb
FM	frequency modulation
HPC	high-pressure compressor
L	aircraft lift, lb
LPT	low-pressure turbine
n_x	aircraft longitudinal acceleration, g
n_y	aircraft lateral acceleration, g
n_z	aircraft normal acceleration, g
ND	normal digital (flight control system mode)
p	roll rate, deg/sec
PCM	pulse-code modulation
P_s	specific excess power, ft/sec
p_{s0}	freestream static pressure, lb/in.^{2}

p_{s6}	afterburner inlet static pressure, lb/in.^{2}
p_{s7}	exhaust nozzle inlet static pressure, lb/in.^{2}
PSL	Propulsion System Laboratory (NASA Lewis)
p_{T5S8}	turbine discharge total pressure, lb/in.^{2}
q	pitch rate, deg/sec
$\bar{q}$	dynamic pressure, lb/ft^2
r	yaw rate, deg/sec
RIG	real-time interactive graphics system
S	reference wing area, ft^2
SGTM	simplified gross thrust method
SNTM	simplified net thrust method
V_t	true airspeed, ft/sec
WATR	Western Aeronautical Test Range
W_t	aircraft gross weight, lb
α	angle of attack, deg
β	angle of sideslip, deg
δ_a	differential flaperon deflection, deg
δ_c	canard deflection, deg
δ_f	symmetric flaperon deflection, deg
δ_{ps}	pitch stick deflection, in.
δ_r	rudder deflection, deg
δ_s	strake flap deflection, deg
δ_{ys}	lateral stick deflection, in.
ϕ	bank angle, deg
μ	loads strain gage measurement
θ	pitch attitude, deg

1. INTRODUCTION

Beginning with the X-29A maiden flight in December of 1984, a key program objective has been to evaluate several integrated advanced technologies for future military applications. The X-29A advanced technology demonstrator flight envelope expansion program and the subsequent flight research phase gave impetus to the development of several innovative real-time analysis and display techniques. Most of these developments resulted from the nature of the unique technologies to be evaluated and critical requirements for safety-of-flight assurance. The forward-swept wing design and the concern of its inherent tendency toward structural wing divergence created the need for constant in-flight structural loads monitoring. It was especially important to monitor the interaction between the vehicle bending and torsion loads against load limit envelopes for critical airframe members. In order to monitor critical structural modes, the dynamic characteristics of the structure were determined from real-time computation of the frequency and damping of five critical structural modes that were in turn graphically compared against predictions during the mission.

The large subsonic airframe negative static margin of 35 percent required high levels of augmentation to artificially stabilize the aircraft. The performance of the flight control system thus became a key factor in the flight envelope expansion. It was desirable to calculate and monitor system stability margins derived from open-loop frequency response characteristics in real time. In addition to monitoring stability in the frequency domain, actual aircraft time responses were compared with predicted responses from linear aircraft simulation models during flight. Although not safety-of-flight critical, real-time aeroperformance analysis in terms of in-flight net thrust and aircraft lift and drag polars would allow for immediate evaluation of aircraft maneuver technique and data quality to insure premium postflight data results and minimum flight repeats.

Such sophisticated real-time analyisis and display required careful integration of the aircraft telemetry data downlink system and the NASA Western Aeronautical Test Range (WATR) mission control facility.

This paper describes the primary features and analytical methods of each technique. It also summarizes how the techniques were used during flight to enhance flight safety and increase flight productivity. A description of the WATR facility is given along with a discussion of the flight data processing flow. Examples of data processed and the flight data displays are shown.

2. X-29A AIRCRAFT DESCRIPTION

The X-29A is an advanced technology demonstrator developed by the Grumman Aerospace Corporation (Bethpage, New York) in partnership with the Defense Advanced Research Projects Agency, NASA, and the Air Force. This small, single-seat fighter-type aircraft's (fig. 1) technologies include the 30° forward-swept wing that includes a thin, supercritical airfoil section with graphite-epoxy upper and lower wing skins configured to inhibit wing structural divergence. Other technologies are the close-coupled canard-wing configuration, a three-surface pitch control system, an automatic wing camber control mode, a large negative static margin, and a triplex digital fly-by-wire flight control system.

The aircraft is powered by a single General Electric (Lynn, Massachusetts) F404-GE-400 afterburning turbofan engine. The engine thrust rating is 16,000 lb of static thrust at sea level. Further details of the X-29A configuration and the technology benefits can be found in reference 1.

3. AIRCRAFT DATA ACQUISITION SYSTEM

All pulse-code modulation (PCM) data was encrypted and telemetered to the ground as a single uncalibrated serial PCM stream along with some high-response frequency modulation (FM) data. The instrumentation system schematic can be seen in figure 2. The 10-bit PCM system sampled data 25 to 400 samples/sec, depending on the data frequency content desired by the various engineering disciplines. Including the FM system, a total of 691 aircraft parameters were measured. The data parameter set included measurements for structural loads and dynamics, flight controls, aircraft subsystems, stability and control, propulsion, aeroperformance, aerodynamic buffet, wing deflections, and external pressure distributions.

A pitot-static noseboom system provided air data information and angles of attack and sideslip from boom-mounted vanes. A set of body-mounted accelerometers provided measurement of aircraft c.g. accelerations. The aircraft had an extensive array of control surface position sensors and flight control system (FCS) performance parameters. The airframe was also heavily instrumented with strain gages and high-response structural dynamic accelerometers.

The thrust-calibrated engine was fully instrumented for real-time thrust calculation as well as postflight analysis using the traditional gas generator thrust calculation method. The unique real-time thrust measurement instrumentation consisted simply of eight static pressure measurements in the afterburner section along with a 20-probe measurement rake of the turbine exhaust total pressure. A schematic of just the real-time engine measurements is shown in figure 3.

4. TEST RANGE AND REAL-TIME SYSTEM

The NASA Ames Research Center, Dryden Flight Research Facility's Western Aeronautical Test Range, or WATR, is a large, highly integrated facility that provides aircraft and telemetry tracking; communications systems; a real-time data acquisition, processing, and display system; and a mission control center.

Current capabilities of the WATR include reception of up to two simultaneous downlink data streams from each research aircraft at a maximum rate of 1 Mbit/sec/stream. The data stream is decrypted, time tagged, compressed, converted to engineering units, limit checked, and stored in real time at a maximum rate of 200,000 words/sec/data stream. This storage area can hold 4096 calibrated parameters plus 3200 computed parameters for recording, further processing, or display.

There are three dedicated real-time minicomputers for on-line data processing and control of display apparatus. Two of these computers are Gould 32/6780 (Gould Electronics, Inc., Cleveland, Ohio) machines and one is a Gould 32/9780 system.

Data display capabilities in each of two identical mission control rooms include eighteen 8-channel strip charts, numerous cathode ray tube digital data displays of either the fixed update or continuous scroll type, color graphics displays, and conventional analog meters and discrete lights. A terminal, located in the mission control center, controls the selection of several different engineering color graphics displays including aeroperformance, flight controls and stability and control, and structural dynamics and loads. A photograph of the mission control center is shown in figure 4. Details of the WATR configuration and operation can be found in references 2 and 3.

5. ANALYSIS AND DISPLAY TECHNIQUES

5.1 Structural Loads

The forward-swept wing's inherent tendency toward static structural divergence and the potential for high loads on the canard, wing flaperons, and strake flaps necessitated direct real-time monitoring of certain critical structural loads. The interaction and interrelationship between the component bending and torsional loads was of concern and required a cross plotting routine to display these loads relative to strength envelopes rather than displaying separate time histories (conventional stripcharts were also used). These data were plotted graphically and monitored continuously, particularly as the X-29A expanded its normal load factor envelope. Extensive loads analysis of these and other structural components was also conducted post-flight. Both symmetric and asymmetric loading were carefully explored through a series of flight clearance maneuvers (ref. 4). Point-to-point real-time loads clearance was critical for flight safety and productivity as the X-29A's flight and maneuver envelope was expanded.

Cross-plotting capabilities were used to display strength envelopes for various vehicle stations on the major aircraft components. These plots were generally in the form of bending/torsion interaction plots where structural limits are interdependent. Loads and flight parameters are computed on the Gould real-time computers and passed to the real-time interactive graphics (RIG) for display. Structural loads are computed from a conventional point load calibration where for shear, bending, or torque the general equation form is

$$\text{load} = C_1\mu_1 + C_2\mu_2 + \ldots + C_n\mu_n \tag{1}$$

where μ represents the individual strain gage measurements and C represents ground load calibration coefficients derived from regression techniques.

Figure 5 shows the color graphics display with sample data from a windup turn maneuver where the wing and canard plots are of primary interest. The upper left plot of the display shows the left canard root loads; the upper right plot, the fuselage lateral and vertical bending loads; the lower left plot, the left wing root loads; and the lower right plot, the vertical tail root loads. On the canard, wing, and vertical tail plots, the horizontal axis is the torsional load, and the vertical axis is the bending load. During a maneuver, flight data are plotted with 80 and 100 percent design limit "boxes" or polygons superimposed with the general intent being to stay within the inner 80 percent box. Alternate pages are available that replace the lower right plot with a different wing load station. Digital parameters displayed to the left of the plots contain flight conditions and aircraft state parameters in the upper block. The lower block contains discrete load channel outputs. All digital data are updated at one sample/sec. The cross-plotted loads are displayed along with their strength envelopes at a computed update rate of 5 to 10 samples/sec. Hardcopies of the display are available in near-real time. These real-time dynamic displays allowed for the efficient and safe structural loads envelope clearance for the X-29A.

5.2 Structural Dynamics

Tracking of the aircraft structural dynamics was a key factor in the safe expansion of the flight envelope of the X-29A. Flight monitoring of the aircraft structural modal stability included both the airframe elastic modes, or aeroelasticity, and the FCS-elastic mode interaction, or aeroservoelasticity. Some twelve structural dynamic modes could be identified on the X-29A, of which the five most critical were tracked in real time in flight for all three flight control system modes (normal digital, ND; digital reversion, DR; analog reversion, AR). The five modes tracked in real time included the first symmetric and antisymmetric wing bending modes, the first fuselage vertical and lateral bending modes, and the first vertical fin bending mode. Flight-derived modal frequency and damping were compared in real time against closed-loop aeroservoelastic predictions and provided stability trend data as a function of airspeed as the flight envelope was expanded. Of particular concern was monitoring for the onset of a potential dynamic interaction known as body-freedom flutter, or BFF. This was predicted to occur when the wing-first-bending-mode frequency decreased and coupled with the aircraft longitudinal short-period mode. Body-freedom flutter was predicted to act as a precursor to the static wing divergence. To date no BFF tendencies have been observed in flight. Flight data from the other seven structural modes were reduced and analyzed using postflight techniques.

Natural turbulence, pilot stick raps, and a wing flaperon eccentric rotary-mass excitation system were used to excite the aircraft structural modes. The flaperon rotary-mass excitation system used a frequency-sweep vibration input in an attempt to identify a predicted supersonic midflaperon torsion flutter mode. Typically the aircraft was stabilized in level flight for 1 to 2 min to perform the stick raps, the rotary-mass frequency sweeps, and for natural turbulence excitation.

The real-time data reduction technique consisted of a fast Fourier analysis method carried out on a Fourier analyzer using a flight data frame size of 1024 data samples at 100 samples/sec. The inverse Fourier transform was computed to obtain the autocorrelation function from which a data cutoff time could be manually selected. Smoothing the autocorrelation function yielded an autopower-spectrum display as a function of frequency (fig. 6) that was curve-fitted for each structural mode. The structural modal frequency occured at the maximum amplitude of the power spectrum density curve, and the structural damping was extracted using the half-power technique. More discussion of this technique can be found in reference 5. The flight-derived frequency and damping of the five primary structural modes were compared in real time with precomputed predictions as a function of aircraft equivalent airspeed (fig. 7).

Usually all three flight control modes were plotted and tracked at the same flight condition on the same data plot display to observe any aeroservoelastic effect. Adverse trends in frequency or damping of a particular structural mode would halt the flight envelope expansion until the phenomenon could be understood or further analyzed using postflight techniques.

5.3 Flight Control Systems

The safe and efficient flight testing of the X-29A required close monitoring of the dynamic stability levels because of the high degree of static instability and the minimal predicted stability margins at some flight conditions. A postflight data analysis method was used during initial envelope expansion flights for flight control systems clearance and dynamic stability checks. This process nominally required 1 to 3 days and allowed only one envelope expansion point be flown per flight to enable careful extrapolation of critical dynamic stability levels.

Efforts to improve flight productivity and safety resulted in the development of two new real-time dynamic stability techniques—one based on frequency response and the other based on time response. These methods improved flight test efficiency significantly by allowing multiple envelope expansion points on a single flight. The direct in-flight measurement of actual aircraft dynamic stability levels and online comparisons with preflight predictions also provided for enhanced safety.

Even though the X-29A longitudinal control system used multiple sensor feedbacks and a three-surface control effector mechanization (ref. 6), the control law did collapse into a single-loop configuration internally in the software. This allowed a classical open-loop frequency response technique to be used to assess longitudinal dynamic stability levels while maintaining all feedback loop closures. Pilot-generated frequency sweeps were used for excitation, and internal control system parameters were used for frequency response computations.

A diagram of the implemented technique is shown in figure 8. The internal control system parameters were telemetered to a ground computer, computations were performed using a fast Fourier transform algorithm, and the flight-determined frequency response was compared with a precomputed estimate based on simulation models. This technique provided a real-time comparison of predicted gain and phase margins with actual flight-determined values. This information was used to assess whether to proceed to the next flight test point immediately or to hold for further analysis.

A typical real-time graphical display is shown in figure 9. In general, the comparisons proved to be remarkably close, indicating that the mathematical models of the aircraft used in the predictive analyses were quite accurate. At one flight condition, the comparison was not too close, and a modification of the overall pitch loop gain was required to establish adequate stability margins. The successful modification (fig. 10) was made based solely on the frequency response results, attesting to the high quality data achievable using this technique.

A technique allowing real-time time response comparisons between flight and linear simulation data was also developed (ref. 7) to aid assessments of X-29A's flight control system performance. A technique similar to the one used for the frequency response tests was implemented. As shown in figure 11, aircraft sensor data was downlinked for use in ground computer algorithms, and computed results were displayed on graphical terminals. In the longitudinal case, pilot input signals were used as the input commands to a linear simulation of the aircraft. In the lateral-directional case, the aircraft's suface positions were used as a direct input into the simulation model equations. The output of the simulation was overplotted directly with the actual measured aircraft response parameters, thus allowing a real-time assessment of control system performance.

A typical comparison plot for a series of pulse maneuvers is shown in figure 12(a) for the pitch axis parameters and figure 12(b) for the lateral-directional axes parameters. The comparisons generally agreed closely and were sufficient to insure the aircraft motions were near those predicted and additional test points could be taken. Comparisons with linear simulation data rather than the full nonlinear simulation also allowed for easy detection of unexpected nonlinearities.

5.4 Aeroperformance

The X-29A aeroperformance real-time analysis technique development did not have a direct role in flight safety or flight envelope clearance. It was developed, rather, to increase flight efficiency and productivity through maneuver technique evaluation and data quality control to insure the best aeroperformance data possible. Direct real-time evaluation of the final data analysis product, as in the case of drag polar coefficients of lift and drag, minimized the number of flight repeats that often arise when postflight data reduction reveals poor data quality or poor flight maneuver technique such as unacceptably high maneuver dynamics. In addition to the value of immediate in-flight aircraft performance evaluation and immediate hard copy of flight results for postflight evaluation, the technique has the potential added bonus of utilization for real-time in-flight aerodynamic optimization of the aircraft.

The real-time aeroperformance data analysis method is based on the in-flight calculation of net thrust from static pressure measurements in the engine afterburner section. This algorithm was specially developed for the X-29A program by the Computing Devices Company (ComDev) of Ottawa, Canada and is known as the simplified net thrust method (SNTM). It is based on a complete thrust calibration over the power range of the flight test engine at the NASA Lewis Research Center Propulsion System Laboratory PSL-4 facility (ref. 8). The method is derived from the simplified gross thrust method (SGTM) developed 15 years earlier (ref. 9). The extension of the SGTM method to the SNTM method involved the real-time calculation of ram drag from true airspeed V_t and inlet mass flow rate. Net thrust was also corrected for estimated nozzle and spillage drag, yielding the net propulsive force F_{np}. A nominal accuracy of ± 3 percent was achieved from this algorithm for real-time net thrust calculation. Rapid engine throttle transients, performed to check the net thrust algorithm dynamic response, showed the algorithm could closely follow engine transient responses. Details of the method are found in reference 10.

Aircraft coefficients of lift C_L and drag C_D were calculated from the equations

$$C_D = \frac{D}{\bar{q}S} = \frac{F_{np} - F_{ex}}{\bar{q}S} \qquad (2)$$

where excess thrust is computed from

$$F_{ex} = n_x W_t \tag{3}$$

and

$$C_L = \frac{L}{\bar{q}S} = \frac{n_z W_t - F_g \sin\alpha}{\bar{q}S} \tag{4}$$

Aircraft specific excess power P_s is also computed and displayed as a function of Mach number from the equation

$$P_s = F_{ex} V_t / W_t \tag{5}$$

The maneuver techniques used were the dynamic pushover-pullup and the constant-thrust, constant-Mach windup turn to sweep out a wide range of angle of attack at a given Mach number in two short maneuvers. These maneuvers were flown back-to-back at a nominal 0.20 g/sec g-onset rate at fixed Mach number increments over the speed range of 0.40 to 1.30 Mach. The maneuver pair could be completed in less than a minute. The real-time data inputs were neither filtered nor thinned. Data were plotted on the color graphics display at up to 12.5 times/sec, while columnar engine and aircraft digital data were updated once per second on the same display screen.

Figures 13 and 14 show representative displays of the quality of drag polars achieved. These real-time results were compared with later postflight-reduced drag polar results and were found to be in good agreement. Because of the decision not to digitally filter the aircraft accelerometers in real time, aerodynamic buffet onset could also be seen as a function of angle of attack and coefficient of lift on the drag polar and lift curves.

6. CONCLUDING REMARKS

Consideration for flight safety and efficient flight envelope expansion of the X-29A led to the development of several new, innovative, real-time analysis and display techniques. Critical X-29A technologies requiring the continuous in-flight monitoring included the forward-swept wing with its inherent tendency toward structural wing divergence and the large degree of airframe subsonic static instability. The real-time analysis techniques developed included structural static loads, structural dynamics, flight control system stability characteristics and aircraft flight response, and aeroperformance. Several of these analysis techniques also provided direct comparisons of flight-measured results with analytical predictions. These techniques greatly improved flight productivity both during the flight envelope expansion phase and the subsequent flight research phase, reducing the need for repeat flights or unnecessary postflight data reduction. The NASA Western Aeronautical Test Range capabilities enabled telemetry acquisition, real-time data processing, and display of the flight data in the mission control center.

REFERENCES

1. Sefic, W.J., and W. Cutler, "X-29A Advanced Technology Demonstrator Program Overview," AIAA 86-9727, April 1986.
2. Moore, Archie L., *The Western Aeronautical Test Range of NASA Ames Research Center*, NASA TM-85924, 1985.
3. Moore, Archie L., and Constance D. Harney, *Development of an Integrated Set of Research Facilities for the Support of Research Flight Test*, NASA TM-100427, 1988.
4. Hicks, John W., James M. Cooper, Jr., and Walter J. Sefic, *Flight Test Techniques for the X-29A Aircraft*, NASA TM-88289, 1987.

5. Kehoe, Michael W., *AFTI/F-16 Aeroservoelastic and Flutter Flight Test Program — Phase I*, NASA TM-86027, 1985.

6. Gera, Joseph, and John T. Bosworth, *Dynamic Stability and Handling Qualities Tests on a Highly Augmented, Statically Unstable Airplane*, NASA TM-88297, 1987.

7. Bauer, J.E., D.B. Crawford, D. Andrisani, and J. Gera, "Real-Time Comparison of X-29A Flight Data and Simulation Data," AIAA 87-0344, Jan. 1987.

8. Burns, Maureen E., and Thomas A. Kirchgessner, *Airflow Calibration and Exhaust Pressure Temperature Survey of an F-404, S/N 215-209, Turbofan Engine*, NASA TM-100159, 1987.

9. Kurtenbach, Frank J., Evaluation of a Simplified Gross Thrust Calculation Technique Using Two Prototype F100 Turbofan Engines in an Altitude Facility, NASA TP-1482, 1979.

10. Ray, R.J., J.W. Hicks, and R.I. Alexander, *Development of a Real-Time Aeroperformance Analysis Technique for the X-29A Advanced Technology Demonstrator*, NASA TM-100432, 1988.

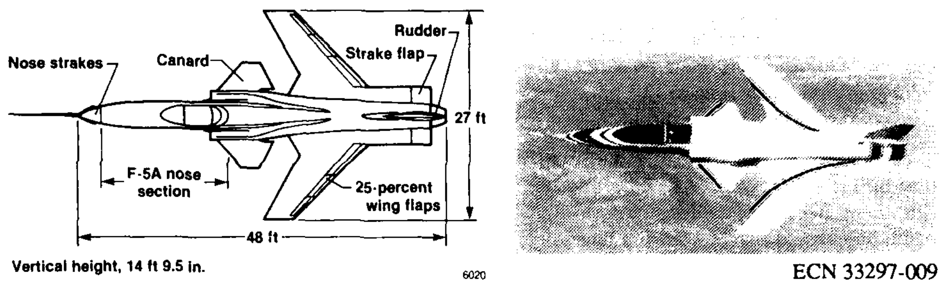


Figure 1. X-29A advanced technology demonstrator.

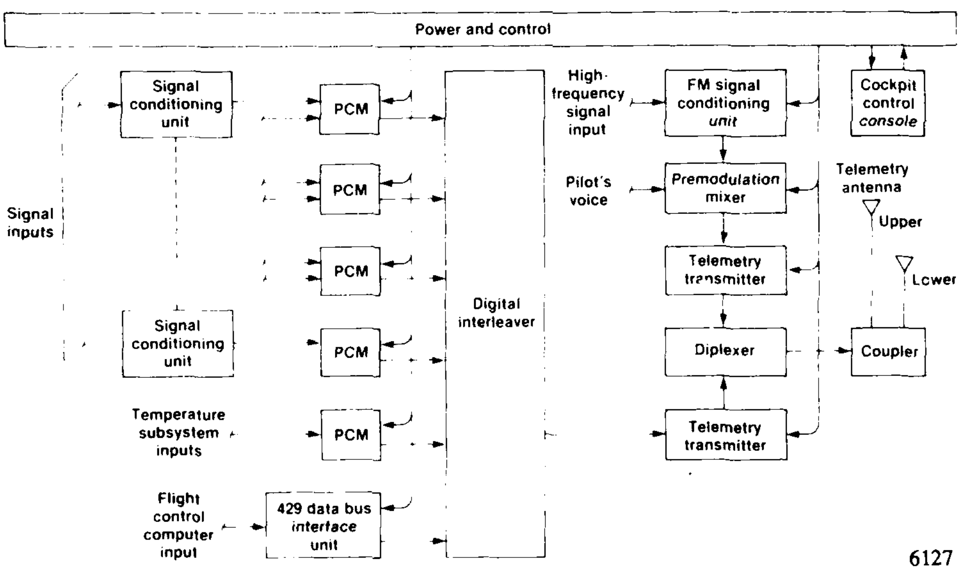


Figure 2. X-29A on-board data acquisition system.

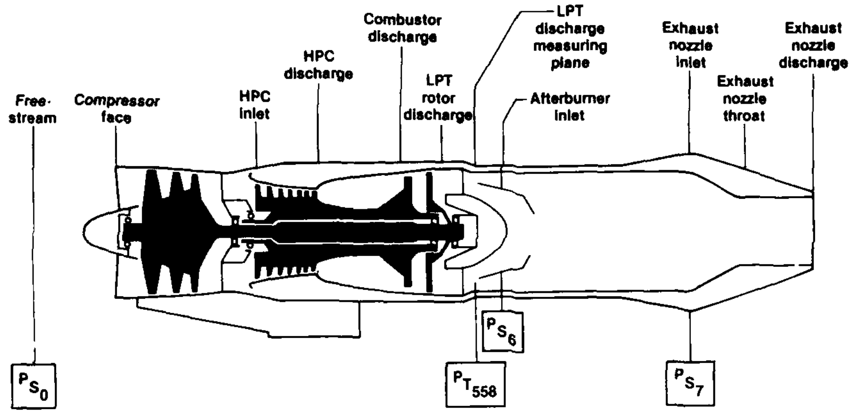
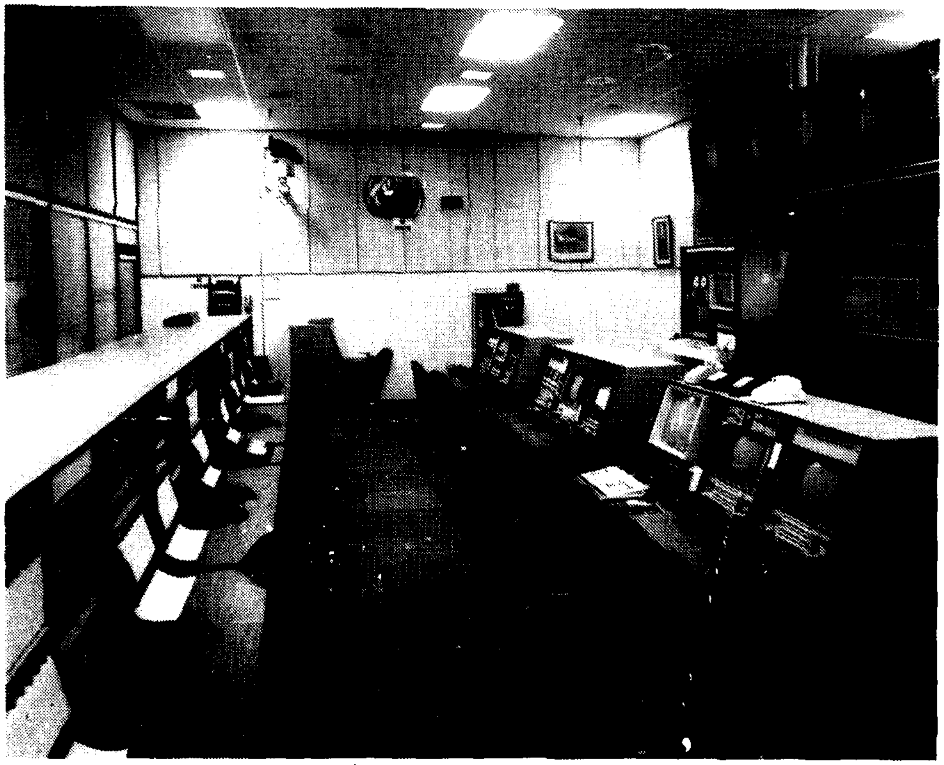


Figure 3. Engine instrumentation system.



EC 88-0019-001

Figure 4. Western Aeronautical Test Range mission control center.

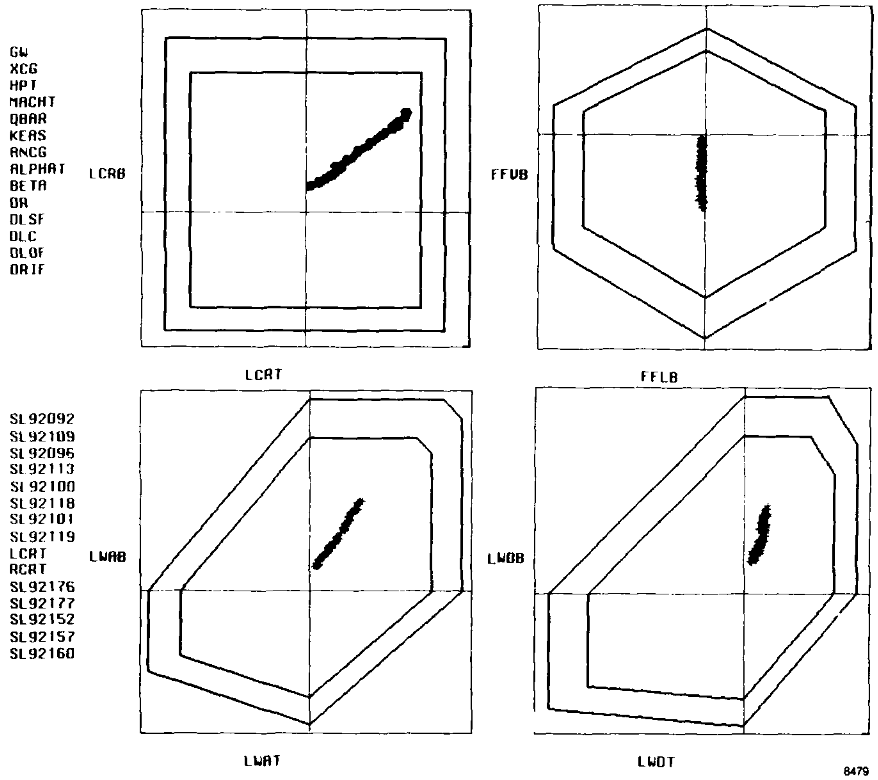


Figure 5. Hard copy version of real-time aircraft structural loads graphics display.

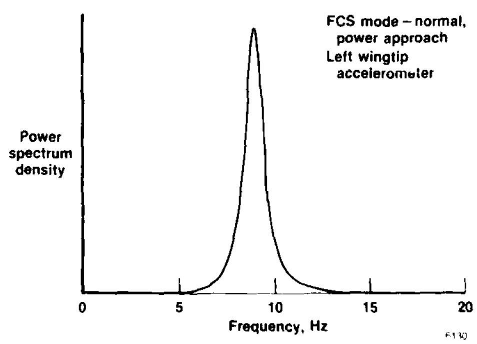


Figure 6. Parabolic curve fit to determine structural frequency and damping in real time.

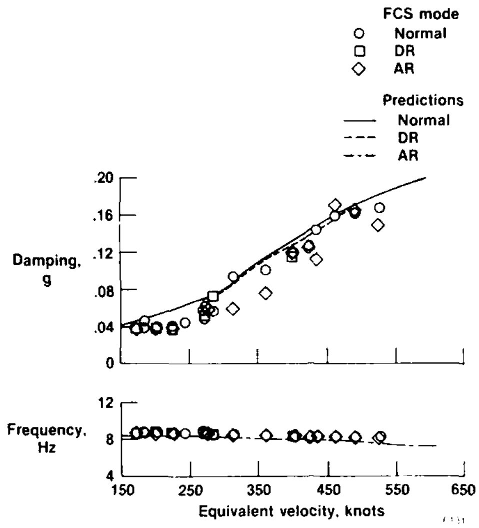


Figure 7. Structural mode frequency and damping characteristics.

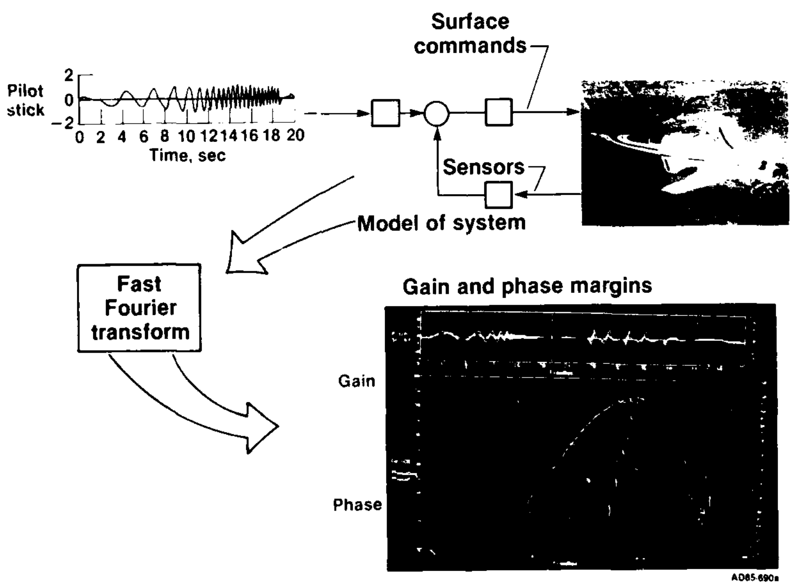
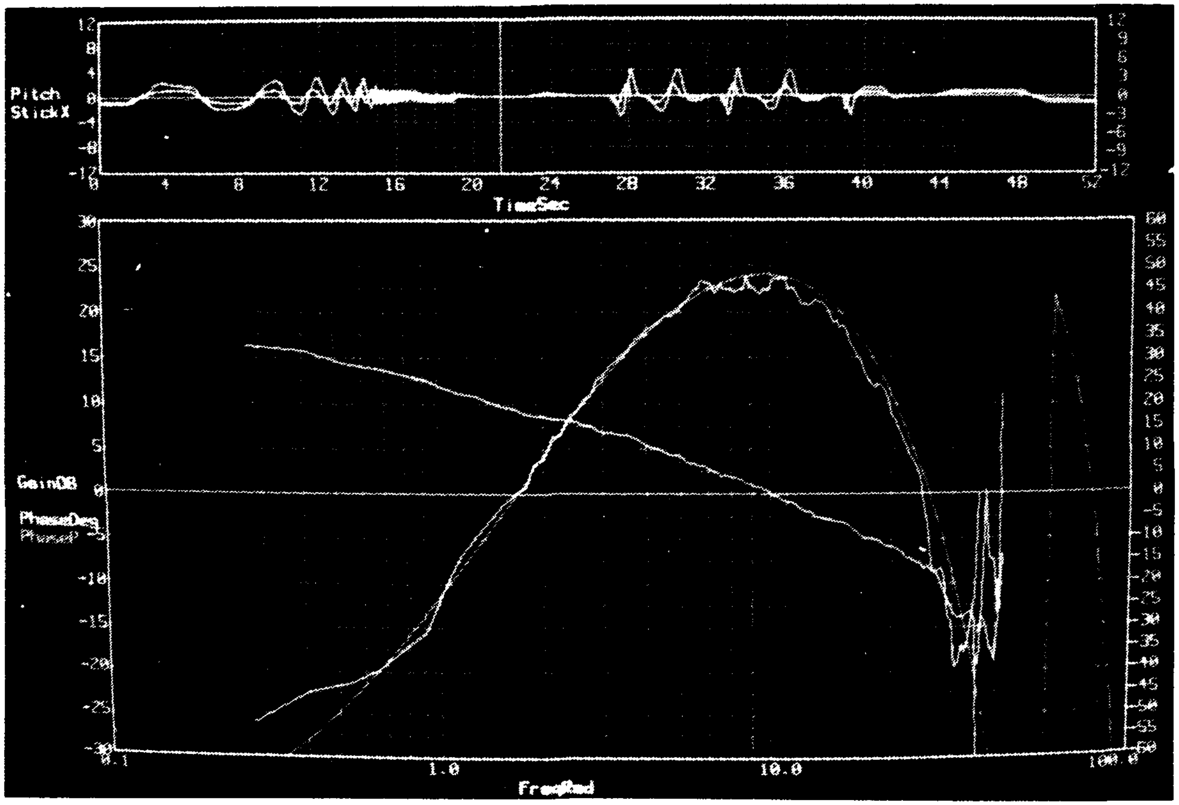


Figure 8. FCS real-time frequency response flow chart.



EC 86-33511-001

Figure 9. Photo of real-time FCS frequency response graphics display.

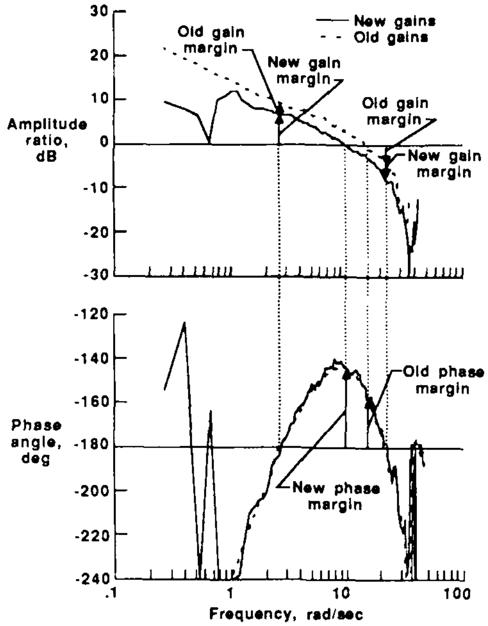


Figure 10. Effect of gain change on flight-measured Bode plot.

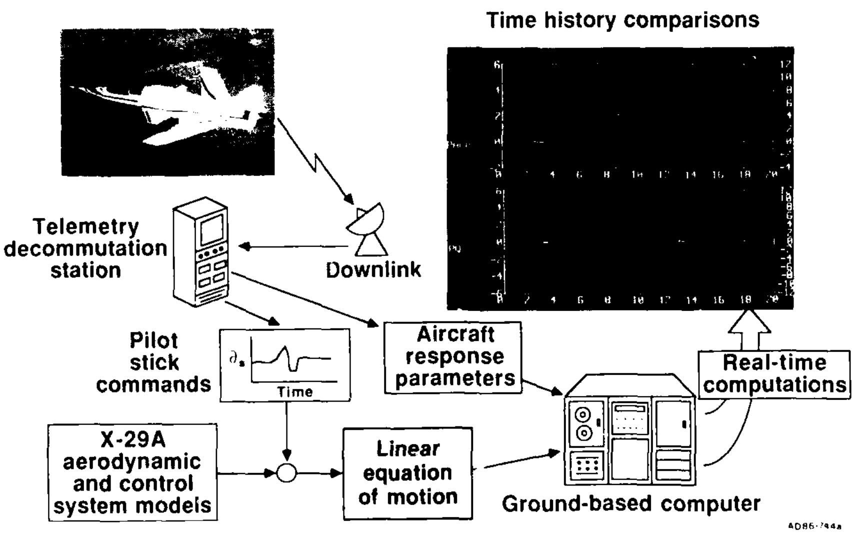
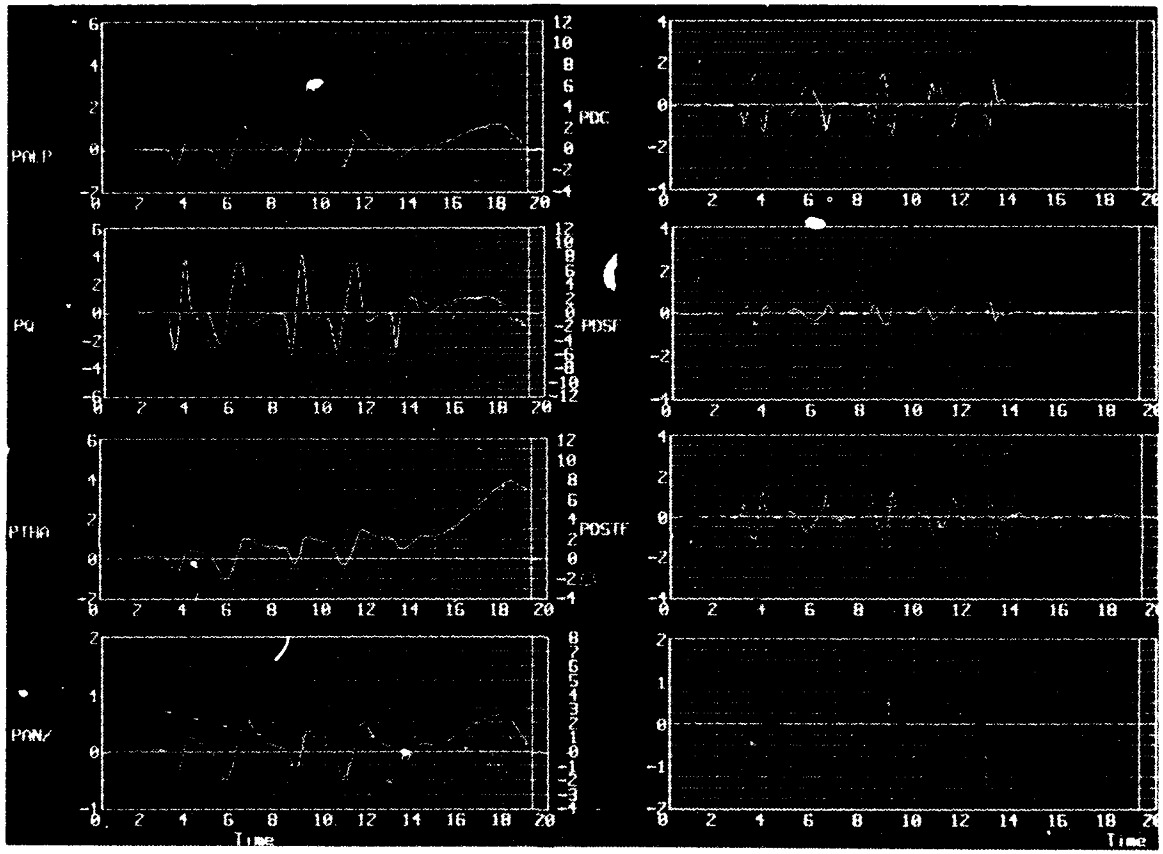


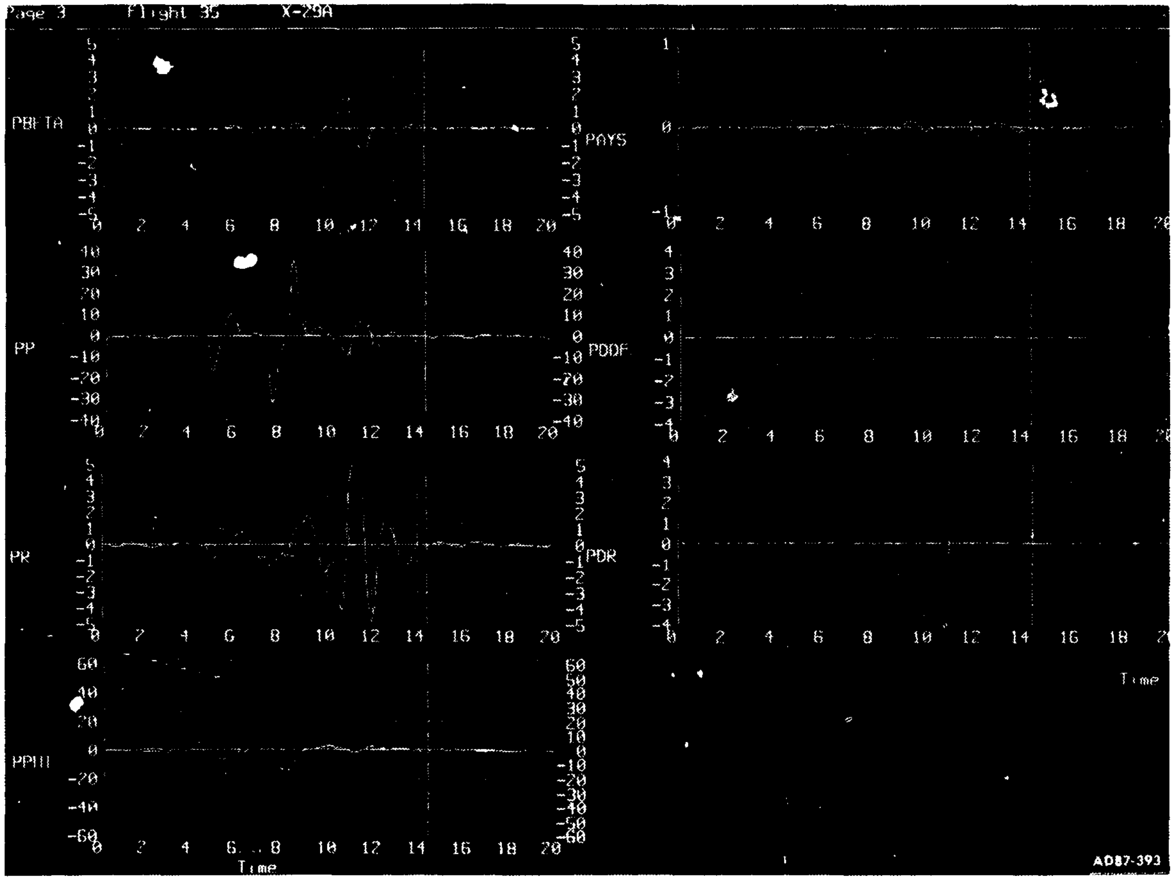
Figure 11. Aircraft flight response characteristics flow chart.



EC 88-0173-001

(a) Longitudinal.

Figure 12. Photos of graphics displays of time history comparison of real-time linear simulation data and flight data.



(b) Lateral-directional.

Figure 12. Concluded.

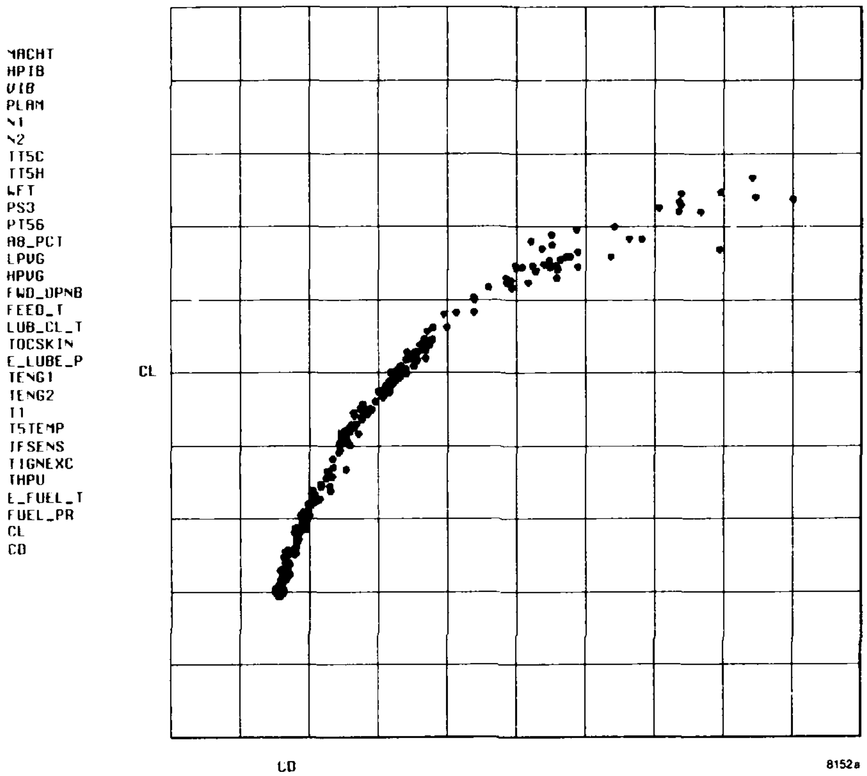


Figure 13. Hard copy version of real-time windup turn-generated drag polar.

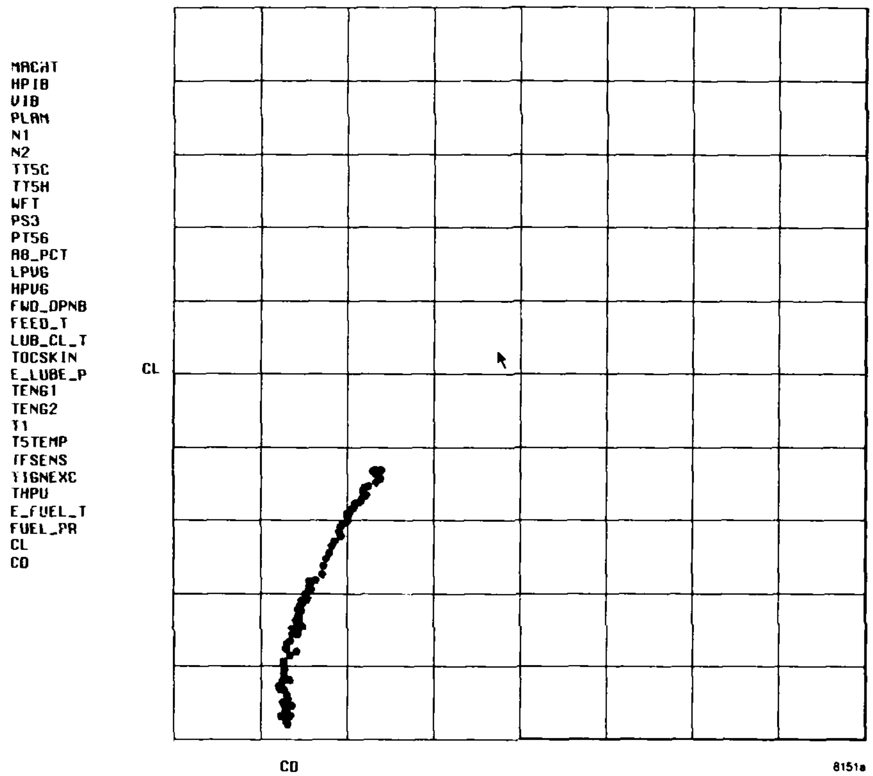


Figure 14. Hard copy version of real-time pushover-pullup-generated drag polar.

LES TECHNIQUES D'ESSAIS
AUX AVIONS MARCEL DASSAULT - BREGUET AVIATION

Comment augmenter l'efficacité des essais en vol

FLIGHT TEST TECHNIQUES ADOPTED BY
AVIONS MARCEL DASSAULT - BREGUET AVIATION

How to increase the efficiency of flight testing

By Jean COSTARD
Test Equipment Manager
AMD-BA BP 28 13801 ISTRES CEDEX (FRANCE)

Résumé :

L'augmentation de la complexité des avions et des systèmes a entraîné une augmentation parallèle très importante des essais de mise au point, donc du coût.

Les AMD-BA ont adopté des techniques qui permettent de diminuer le nombre d'essais en vol au profit d'essais au sol moins coûteux, les vols devant être des démonstrations en vue de la qualification.

Ces techniques ont permis de maîtriser les cadences d'acquisition de données ce qui permet de conserver des moyens de calcul raisonnables et faciliter les analyses.

Abstract :

The ever-increasing complexity of aircraft and systems entailed, in parallel, a very substantial increase in development tests, and consequently a rise of cost.

AMD-BA have adopted techniques enabling to reduce the number of flight tests for the benefit of less expensive ground tests : flights must be saved for demonstration with the aim of qualification.

These techniques allowed to keep in control of data acquisition rates while preserving reasonable means of computation and facilitating analysis.

1 - INTRODUCTION - PRESENTATION OF THE COMPANY

The Avions Marcel DASSAULT - BREGUET AVIATION Company was born in 1967 from the merger of two Companies with a very rich past in aeronautics'.

. La Société des Avions Marcel DASSAULT,
. La Société des Avions Louis BREGUET.

The Avions Marcel DASSAULT Company was founded by Mister Marcel DASSAULT who in a time span of 69 years devoted to aeronautical creation has produced 92 prototype aircrafts : 56 have been developed since World War II and 20 amongst them were produced in large series (Fig. 1.1).

The era from 1947 to 1960 was the "golden age" for pilots. Numerous prototypes came into being, leaning on rather vague specifications by Headquarters who had to re-create their Air Forces, and in the Design Offices of numerous Companies innovative "brainchildren" fostered, culminating in a large variety of projets encompassing light intercepters flying at high Mach number and high altitudes and Close Ground Support aircraft.

But very quickly designers had to fly in the face of the facts : finances were limited.

Mister DASSAULT took the lead with his polyvalent delta-winged aircraft MIRAGE III which was the forerunner of numerous versions : 1412 machines have been produced for 21 Air Forces all over the world.

This aircraft as well as several others : MIRAGE IV, ETENDARD, FALCON and interesting prototypes such as the vertical take-off MIRAGE III V and the swing-wing MIRAGE G have contributed in the *progressive refinement of airframe and flight* control systems development techniques. Experimentation of "fly-by-wire" controls started in 1959 on the MIRAGE IV, followed by the MIRAGE III V in 1962.

Another keydate of evolution is 1974 with the arrival of navigation and weapon systems which substantially increased the efficiency of aircraft - but also their price. Airframe development costs, in turn, started climbing with the introduction of new materials and technologies.

Design Engineering costs have been upsloping too, due to the use of powerful computers :

. aerodynamical computations,
. computerized design, that offers the advantage to enable the fast realization of mock-ups but calls for wind tunnel tests which, on account of their automation, follow the same slope.

On the other hand, these means, well run in, convey a solid confidence in the design and have succeeded in reducing the number of prototypes required for project development.

Eight years separate the penultimate military aircraft, MIRAGE 4000, and the latest DASSAULT creation, the RAFALE, and the results obtained with this experimental aircraft enable us to launch, directly, the pre-production aircraft.

In parallel to this evolution of our aircraft, the scope of our responsabilities has enlargened.

Being aircraft manufacturer in the first place, the various prototype aircraft presented are the backbone of our large experience in airframe development :

1 - INTRODUCTION - PRESENTATION DE LA SOCIETE

La Société des Avions Marcel DASSAULT - BREGUET AVIATION est née de la fusion en 1967 de deux Sociétés à très riche passé aéronautique :

. La Société des Avions Marcel DASSAULT,
. La Société des Avions Louis BREGUET.

La Société des Avions Marcel DASSAULT a été fondée par Monsieur Marcel DASSAULT, qui en 69 années de création aéronautique a produit 92 prototypes ; 56 l'ont été depuis la guerre dont 20 ont donné lieu à une production de série (planche 1.1).

La période 1947 à 1960 a été l'âge d'or des pilotes. De nombreux prototypes ont vu le jour sur des spécifications assez vagues des Etats-Majors qui devaient recréer leurs forces aériennes et des idées très florissantes des Bureaux d'Etude de nombreuses sociétés : projets d'intercepteurs légers à grand Mach haute altitude et d'avions d'appui tactique.

Il a fallu se rendre rapidement à l'évidence que les financements étaient limités.

Monsieur DASSAULT s'est alors imposé avec son avion Delta MIRAGE III, polyvalent, qui a donné le jour à de nombreuses versions ; 1412 exemplaires ont été produits pour 21 Armées de l'Air.

Cet avion ainsi que plusieurs autres : MIRAGE IV, ETENDARD et FALCON, de même que des prototypes intéressants comme le MIRAGE III V à décollage vertical et le MIRAGE G à géométrie variable, ont permis de développer des techniques de mise au point de cellules et de *commandes de vol*. Le début des commandes de vol électriques date de 1959 avec le MIRAGE IV puis le MIRAGE III V en 1962.

Une autre évolution date de 1974 avec l'arrivée de systèmes de navigation et d'armement qui ont augmenté de façon importante l'efficacité des avions, mais également leurs coûts. Le coût de la cellule augmente quant à lui par l'introduction de nouveaux matériaux et technologies.

Le prix des études augmente également par l'utilisation d'heures d'ordinateurs puissants :

. *calculs aérodynamiques*,

. *conception sur ordinateur qui permet en* particulier de réaliser rapidement des maquettes mais avec des coûts d'essais soufflerie qui, automatisés, suivent la même pente.

Par contre ces moyens bien rodés donnent une grande confiance dans la conception et ont permis de diminuer le nombre de prototypes.

Huit années ont séparé le dernier avion militaire MIRAGE 4000 et le RAFALE et les résultats de cet avion expérimental permettent de lancer directement des avions de présérie.

Parallèlement à cette évolution de nos avions, nos responsabilités dans la mise au point se sont étendues.

En premier lieu étant constructeur d'avions, les différents prototypes présentés nous ont donné une très large expérience dans les mises au point de cellule :

. Flight envelope extensions with :

  . Monitoring of structural vibrations to prevent flutter
  . Monitoring of stress and strains

. Study of handling qualities

. Spin and high AoA testing

. Tuning of the flight controls comprising conventional systems, electrical controls for hovering, fully electrical systems with analog computers (MIRAGE 2000 and 4000) and digital processors (RAFALE).

. Performance measurements

. Engine integration with air intake adaptation

. Engine tune-out, for instance the GARRETT ATF 3 on FALCON 20 GUARDIAN

. Development of the integral aircraft systems and sub-systems.

. Development engineering of landing gears, in particular test of deck-landings on aircraft--carriers for naval aircraft.

. Fine-tuning of arrester hooks : aircraft--carrier operation or emergency arrestments.

. Noteworthy too, extensive participation in pilot ejection testing.

. Civil aircraft certification :

  . Business aircraft line : FALCON 10, 20, 50, 900,
  . Short and medium range transport aircraft : MERCURE.

. Our experience in armament testing has been maturing throughout the same time span :

  . Armement integration
  . Gunnery and missile firing

and our developments of certain weapons have also made their proofs :

  . gun pods,
  . rocket-launcher,
  . special tanks,
  . photographic recce pods.

Development of reconnaissance aircraft can be included in this chapter.

In 1972, we understood that to export our aircraft, we had to take over the responsibility of weapon systems integration.

The means deployed to ensure this integration and the experience we have been gaining in this process now enable us to take over this task on other than Company-built aircraft, and thus to assist countries in the renovation of their air fleets.

In 1978, we moreover included countermeasures integration and the development of tools, optimizing the man-machine interface, in our spectrum of activities.

. Extensions de domaine de vol avec

  . Contrôle des vibrations de structure pour prévenir le flutter.
  . Contrôle des efforts et contraintes

. Etude des qualités de vol

. Essais de vrilles et grandes incidences

. Mise au point des commandes de vol, classiques, électriques en vol stationnaire, entièrement électriques avec calculateurs analogiques (MIRAGE 2000 et 4000) et numériques (RAFALE).

. Mesure de performances

. Intégration des moteurs avec adaptation des entrées d'air.

. Mise au point de moteurs, cas de l'ATF 3 GARRET sur FALCON 20 GUARDIAN.

. Mise au point des circuits et sous systèmes intégrés avion.

. Mise au point de trains d'atterrissage, en particulier avec essais d'appontage d'avions Marine.

. Mise au point de crosse d'arrêt : porte-avions ou détresse.

. A signaler également une grande participation aux essais d'éjection pilote.

. Certification d'avions civils

  . série des avions d'affaires FALCON 10, 20, 50, 900,
  . Avion transport court-moyen courrier MERCURE.

. Notre expérience dans les essais d'armements est toute aussi ancienne.

  . Intégration d'armements,
  . Tirs canon et missiles

mais également développements de certains armements.

  . Chassis canons
  . Lance roquettes
  . Bidons spéciaux
  . Bidons de reconnaissance photo.

On peut placer dans ce chapitre des développements d'avions de reconnaissance.

En 1972, nous avons compris que pour exporter nos avions nous devions prendre la responsabilité de l'intégration des systèmes d'armes.

Les moyens mis en oeuvre pour cette intégration et l'expérience que nous en avons retirée nous permettent maintenant d'assurer cette tâche sur d'autres avions que ceux de notre société, permettant ainsi à des pays de rénover leurs flottes.

Depuis 1978, nous y avons ajouté l'intégration des contre-mesures et le développement d'outils de mise au point de l'interface homme/machine.

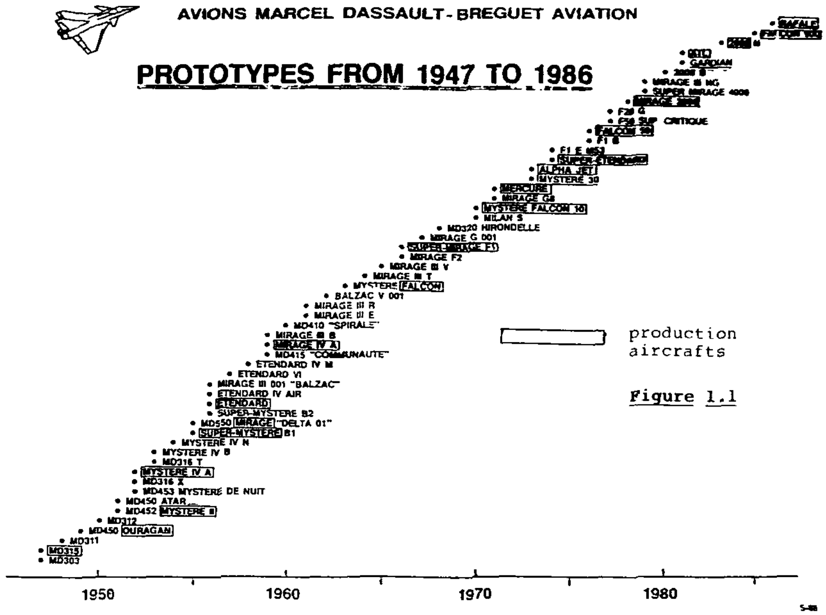


Figure 1.1

2 - ORGANIZATION OF DEVELOPMENT ENGINEERING

To be able to understand the test techniques and means used in other countries and by the various manufacturers, it is indispensable to have an inside knowledge of the structural organization of development.

To develop modern aircraft very powerful computation means are a must. Systems integration problems also necessitate electronic processing tools without large computation capacity but offering numerous and fast entry possibilities. When adding all the peripherals and infrastructures required, the cost of investment is considerable.

To avoid the duplication of these means, several orientations have been adopted.

One approach consists in incorporating all development, integration and evaluation test activities within the framework of an Official Test Center that is also the center for operational evaluation. This is, for example, the solution adopted by the USA. It allows to reduce investments - although the latter can be retraced, dispersed on serveral test centers in lieu of various manufacturers - and ensures a complete integration of development teams and users. However, this solution has its drawbacks:

. Test activities have moved apart from Design Offices, psychologically even more than geographically.

. As manufacturers' test teams are more disseminated, thus less in charge, it may become more difficult to keep qualified staffers.

2 - ORGANISATION DE LA MISE AU POINT

Pour pouvoir comprendre les techniques d'essais et les moyens utilisés dans les différents pays et chez les différents constructeurs il est nécessaire de connaître la structure de l'organisation dans la mise au point.

La mise au point d'un avion moderne demande des moyens de calcul très puissants. Les problèmes d'intégration de systèmes demandent également des moyens informatiques sans grosse puissance de calcul mais avec des possibilités d'entrées sorties importantes et rapides. Si l'on y ajoute tous les périphériques et infrastructures nécessaires, le coût des investissements est considérable.

Pour éviter la duplication de ces moyens plusieurs orientations ont été adoptées.

L'une consiste à intégrer tous les essais de mise au point, d'intégration et d'évaluation au sein d'un centre d'essais officiel étant centre d'évaluation opérationnelle. C'est par exemple la solution adoptée aux USA. Elle permet de diminuer les investissements - bien qu'on les retrouve dispersés dans plusieurs centres d'essais au lieu de plusieurs constructeurs - et assure une intégration complète des équipes de mise au point et utilisateurs. Par contre, elle apporte des inconvénients :

. Les essais sont éloignés des Bureaux d'Etude et surtout plus psychologiquement que géographiquement.

. Les équipes d'essais constructeurs plus dispersées et étant moins responsabilisées, il peut devenir plus difficile de conserver des personnels qualifiés.

. A well equipped Test Center may be tempted to process data more extensively before forwarding them to the Design Office, this entailing a delay in modification proposals.

. Inversely, the Design Office may "forget" the existence of a test team and may fail to consult it or to inform it on modifications that ought to be applied on aircraft under test, and on test requirements for validation.

In EUROPE, manufacturers have upkept their own test activities.

Avions Marcel DASSAULT - BREGUET AVIATION succeeded in reducing development costs by integrating the Flight Test Division into the comprehensive development cycle, encompassing :

. Design Office,
. Partial tests,
. Simultations,
. Global ground tests,
. Flight tests.

Thanks to this integration, it is possible to optimize the adaptation of means to the actual requirements, and to reduce the number of flight tests for the benefit of ground tests that are less costly and often more exhaustive.

On figure 2.1 are outlined the various stages of the development cycle as well as the responsibilities conferred. The same imbrication of tasks has been adopted to handle the different types of development : airframe, weapon system, armaments.

The user, Air Force or Navy Headquarters, express their needs which are defined technically by the French Official Authorities , the "Service Technique de l'Aeronautique (STAé) who, in cooperation with the manufacturer's Design Office, establish the global specifications. At this stage is finalized the choice of the general form of the new aircraft, amongst several computed solutions that were tested in the wind tunnel.

. The following phase consists in reviewing in greater detail the specifications of the various equipment and sub-systems. For the electric flight control system for instance (Fig. 2.2) this optimization also involves the means for simulation and analytic study.

As for weapon systems, the specifications for pilots' displays and man-machine interfaces are elaborated with the help of the OASIS Center (Avionics Specifications Design Tool) where the users join in the manufacturer's team "on the job". This Center, although placed under the responsibility of the Design Office for conceptional functions, is accommodated within the Flight Test Division, to be close to the pilots.

Equipment developments fall under the responsibility of their manufacturers. The various materials are then integrated into test benches :

. Flight Controls Bench in the Design Office (Fig. 2.3). For these tests, the means for computerized processing are the same as during the preceding phase, complemented by other specific tools to meet simu˜tor requirements.

. Integration Benches for Weapon Systems.

. Un centre d'essais bien équipé peut être tenté de traiter de façon plus approfondie les données avant de les transmettre au Bureau d'Etude, d'où un retard dans les modifications à proposer.

. Inversement le Bureau d'Etude peut oublier qu'il a une équipe d'essais, ne pas la consulter, ni l'informer des modifications qu'il serait nécessaire d'appliquer sur les avions en essais et des essais de validation nécessaires.

En EUROPE, les constructeurs ont gardé leurs essais propres.

Aux Avions Marcel DASSAULT BREGUET AVIATION la diminution des coûts de mise au point a été réalisée en intégrant les Essais en ˜ 1 dans tout le cycle de mise au point.

. Bureau d'Etude,
. Essais partiels,
. Simulations,
. Essais au sol globaux,
. Essais en vol.

Cette intégration permet d'adapter au mieux les moyens aux besoins et de réduire le nombre d'essais en vol au profit d'essais au sol moins coûteux et souvent plus complets.

Sur la planche 2.1, on trouvera les différentes étapes dans ce cycle de mise au point avec les différentes responsabilités. L'imbrication des tâches se retrouvera dans les différents types de mises au point : cellule, système d'armes, armement.

L'utilisateur, Etat Major de l'Armée de l'Air ou de la Marine, exprime des besoins qui sont mis en forme techniquement par le Service Technique de l'Aéronautique (STAé) qui, travaillant en collaboration avec le Bureau d'Etude constructeur, élabore les spécifications globales. On aboutit à ce stade à un choix de la forme générale avion parmi plusieurs solutions calculées et essayées en soufflerie.

. La phase suivante permettra d'entrer plus en détail dans les spécifications des différents équipements et sous systèmes.

Dans le cas des commandes de vol par exemple (planche 2.2) interviennent, pour cette optimisation, des moyens de simulation et d'étude analytique.

Dans le cas des systèmes d'armes, les spécifications de visualisations pilote et d'interface homme/machine sont élaborées grâce au centre OASIS où se retrouveront les utilisateurs. Ce centre, bien que dépendant du Bureau d'Etude pour ses fonctions de conception, est situé aux Essais en Vol pour être proche des pilotes.

. La mise au point des équipements est de la responsabilité ces fabricants. Ils seront ensuite intégrés dans des bancs :

. Banc commandes de vol au Bureau d'Etude (planche 2.3). On utilise pour ces essais les mêmes moyens informatiques que pour la phase précédente a:ec d'autres, adaptés au travail de simulateurs.

. Bancs d'intégration pour les systèmes d'armes.

These tests are completed by ground test series in order to update the different computation bases.

On completion of this phase, the equipment is thoroughly debugged, operating directives and provisional limitations are defined, and the performance of the system on the whole and the aircraft are "canned". After having investigated the results concurrently with the governmental Authorities, the aircraft is declared "airworthy".

The first tests are dedicated to identification to introduce parameters that cannot be covered in ground testing, as for instance the evolution of "soft" modes and the man-machine couple. Means and procedures used for test processing are identical to those described earlier, both in the Design Office and on the test rigs (Fig. 2.4).

After having up-dated the bases, it is thus possible to readjust the various directives and instructions for the validation flights, prior to general evaluation and certification.

For test activities, the integration principle was adopted again to associate the teams of the military users and their technical delegations, and the Official French Flight Test Center (C.E.V). Evaluation flights are performed very early and throughout development to validate progress status advances (see Fig. 3.2.5).

Homogeneity of procedures is obtained through the convergence of all operational test teams of the Air and Naval Forces in a single technical Test Center.

This Test Center also places at the manufacturers disposal :

. Facilities for Trajectory measurements,
. A Telemetry Reception Center,
. Chase aircraft.

It also takes over development flight tests for equipment manufacturers who don't have their own test center.

In the following, the users carry out operational qualification flights in their own Flight Test Centers. New needs that may have been revealed in the course of these qualification flights or scheduled status evolutions will give rise to new specifications, undergoing the same development circuit.

The powerful computer backing the management of this organization is that of the Design Office (Fig. 2.5) : after having been the support for the elaboration of the computional bases it also comes in, in association with the flight controls simulator, to assist Design Office in defining the operating directives : flight envelope, flight manoeuvre limits...

The Flight Test Team assumes the part of a very critical controller who compares the real aircraft with design assumptions. This comparison has to be made very quickly, in order to be able to interrupt a test series or a flight, if significant discrepancies are highlighted.

The role of the Flight Test Team, and especially that of the pilot is very weighty, from the definition phase. Their practical authority they convince Management into adjustments to obtain a machine perfectly tailored to the requirements of domestic users and export potentials.

Ces essais seront complétés par des essais au sol qui recaleront les différentes bases de calcul.

On aboutit, à l'issue de cette phase, à des ensembles bien déverminés, des consignes d'emploi et limitations provisoires, *ainsi qu'à* des performances d'ensemble du système et de l'avion.

L'avion alors est déclaré bon de vol après examen des différents résultats avec les Services Officiels.

Les premiers essais seront des essais d'identification pour faire *intervenir les* paramètres qui ne peuvent être obtenus au sol, comme par exemple l'évolution des modes souples et le couplage homme/machine.

Ces essais sont exploités par les mêmes moyens et procédures que précédemment, au Bureau d'Etude ou aux bancs (planche 2.4).

Ceci permet après le recalage *des bases, de* mettre à jour les différentes consignes pour les vols de validation avant l'évaluation générale et certification.

Du point de vue *essais il y a* également une intégration avec les équipes des utilisateurs militaires et de leur délégation *technique, le* Centre d'Essais en Vol officiel (CEV). Des vols d'évaluations sont effectués très tôt et tout le long de la mise au point pour valider des étapes (voir planche 3.2.5).

Une homogénéité dans les procédures est obtenue par la présence d'un centre d'essais technique unique qui reçoit les équipes d'essais opérationnelles des deux armes, Armée de l'Air *et Marine.*

Le centre d'essais met également à la disposition des constructeurs des facilités :

. Trajectographie,
. Centre de réception télémesure,
. Avions d'accompagnement,

Il assure également les essais en vol de mise au point d'équipementiers qui n'ont pas de centres d'essais propres.

Les utilisateurs effectuent ensuite les vols de qualification opérationnelle dans leurs propres centres. De nouveaux besoins mis *en évidence par* ces essais de qualification ou des évolutions prévues feront l'objet de nouvelles *spécifica*tions avec le même circuit de développement.

Avec cette organisation l'ordinateur, à grande puissance de calcul, est celui du Bureau d'Etude (planche 2.5). Il a servi à établir *les bases de* calcul et avec l'aide du simulateur commande de vol, le Bureau d'Etude établit *les consignes :* domaine de vol, limites d'évolution. L'équipe d'essais en vol a un rôle de contrôleur très critique qui compare la machine réelle avec les prévisions. Cette comparaison doit être effectuée très rapidement *de manière à* interrompre une série d'essais ou un vol s'il est constaté des écarts notables.

Le rôle de l'équipe d'essais et particulièrement du pilote est très important dès le stade de définition. Ils peuvent infléchir la Direction Société pour obtenir la machine la mieux adaptée aux besoins des utilisateurs nationaux et potentiels "Export".

To optimize contacts with the users, the AMD-BA pilots are chosen among the various operational disciplines : Air Defense, Spot Bombing, Reconnaissance, Navy.

The Flight Test Team is responsible for :

- The pre-flight preparation of the aircraft and its test installation,
- The organization of ground test programs,
- The organization of flight programs : long-term schedules with the consent of the Design Office and Governmental Authorities, flight-to-flight calendars under their own responsibility.

It is in charge of all the development tasks that do not necessitate extensive processing programs. When anomalies found on a weapon system cannot be explained by simple traces, trouble shooting is entrusted to the integration bench specialists who use play-backs of an incriminated flight phase, extracted from flight recordings, to reach their diagnostic.

For the more complex tests dedicated to performances, handling qualities, load measurements, etc ..., the test data, after check of their validity, are forwarded to the Design Office which will then be in a position, in association with additional information obtained by computation or wind tunnel results, to update the initial bases and, in the following, to extend the limitations.

The Flight Test Team is also in charge of transmitting to equipment manufacturers the data they need to develop their products.

To ensure the correct and expeditious operation of this organization as a whole, it is necessary to attach a particular importance to the standardization of data exchange (Fig.2.6). This chart outlines the standards used by AMD-BA (also confer to figures 2.7 and 2.8) that were also adopted by other institutions in FRANCE.

The data of flights made at ISTRES during the day are checked, and can be at hand the next morning in the Design Office, located 750 kilometers away, at Saint-Cloud (outskirts of PARIS). The transport is handled by lorries at night.

Pour assurer les meilleurs contacts avec les utilisateurs, les pilotes AMD-BA sont issus des différentes spécialités : Défense Aérienne, Bombardement, Reconnaissance, Marine.

L'équipe d'essais a la responsabilité de :

- La préparation de l'avion et de son installation de mesure,
- L'organisation des essais au sol,
- L'élaboration des programmes de vol, les programmes à long terme avec l'accord du Bureau d'Etude et des Services de l'Etat, les programmes vol à vol sous leur propre responsabilité.

Elle est chargée d'assurer les mises au point à partir d'exploitations qui ne nécessitent pas de gros programmes de traitement. Lorsque des anomalies sur le système d'armes ne peuvent être expliquées par des tracés simples, le diagnostique s'effectue sur les bancs d'intégration par des play-back de la phase correspondante à partir des enregistrements de vol.

Pour les exploitations plus complexes (performances, qualités de vol, commandes de vol, efforts) les données, après contrôle de validité, sont transmises au Bureau d'Etude qui pourra alors, à l'aide d'autres informations de soufflerie ou de calcul, recaler les bases initiales puis étendre les limitations.

Les essais en vol assurent également la transmission des données qui sont nécessaires aux équipementiers pour leurs mises au point.

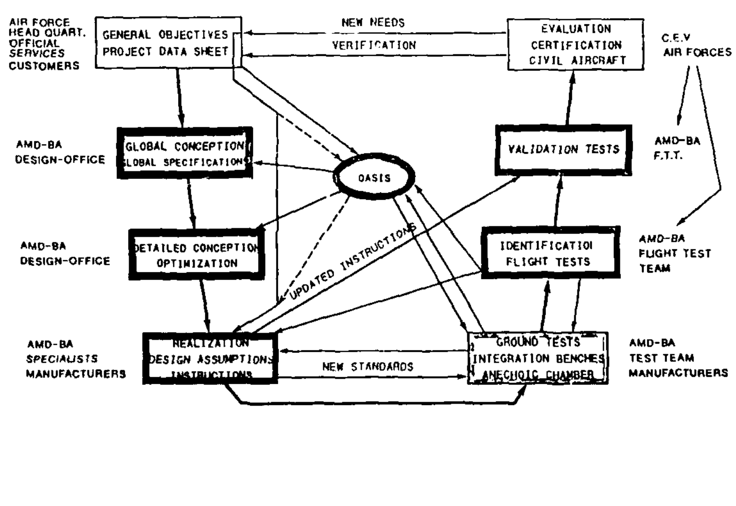
Pour que cet ensemble fonctionne correctement et rapidement il est nécessaire d'apporter un soin particulier à la standardisation des échanges de données (planche 2.6). Cette planche fait apparaître des standards (BX et VAR) (planches 2.7 et 2.8) qui ont été adoptés par d'autres organismes en FRANCE.

Les données contrôlées d'un vol effectué à ISTRES peuvent être disponibles le lendemain matin au Bureau d'Etude à SAINT-CLOUD, centres éloignés de 750 Kms. Ce transport de données s'effectue de nuit par camion.

AVIONS MARCEL DASSAULT - BREGUET AVIATION

Figure 2.1

DEVELOPMENT CYCLE OF AN AIRCRAFT



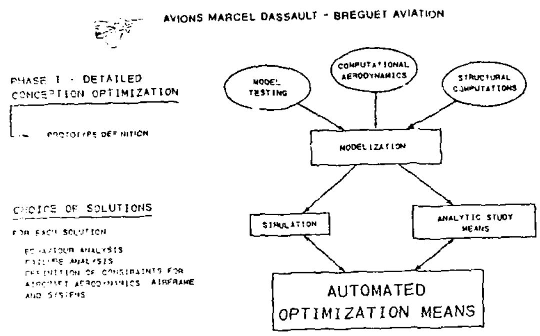


Fig. 2.2

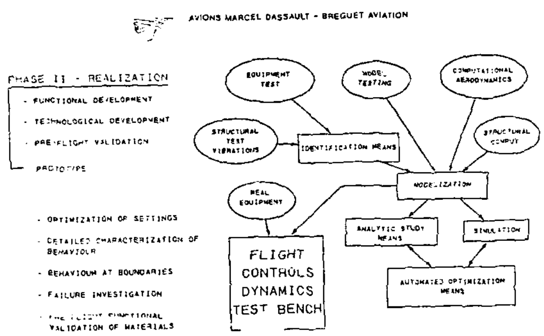


Fig. 2.3

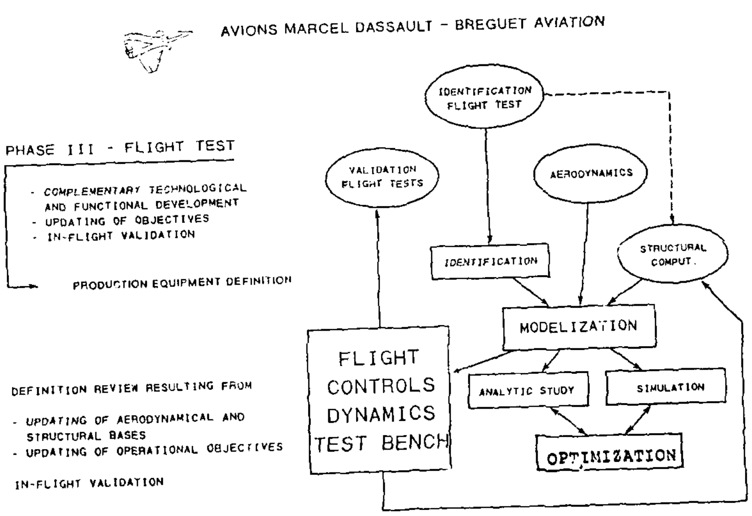


Fig. 2.4

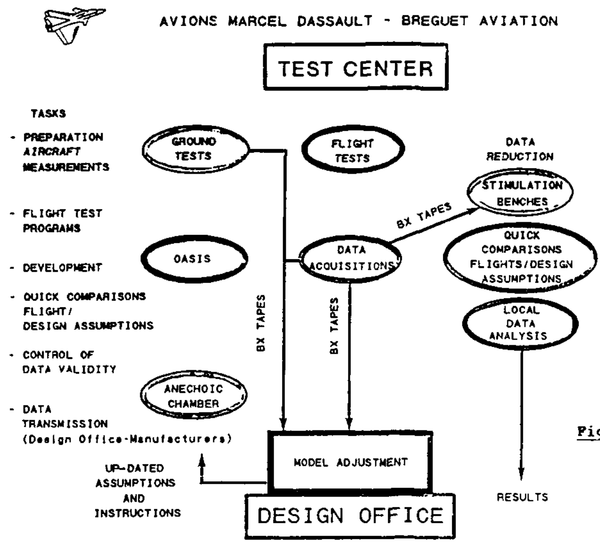


Fig. 2.5

DATA REDUCTION ORGANIZATION

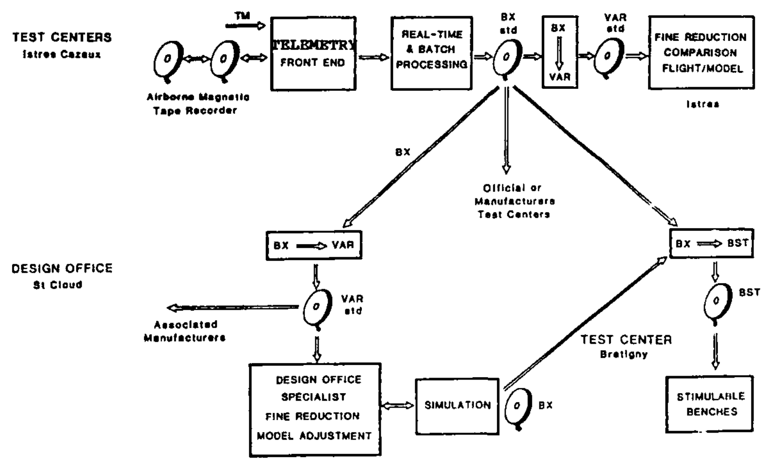


Fig. 2.6

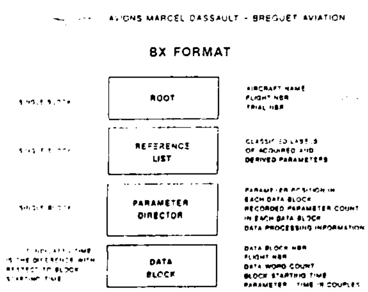


Fig. 2.7

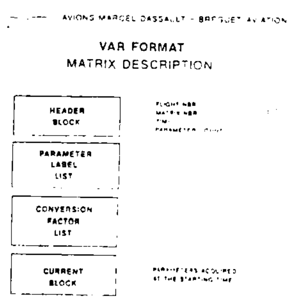


Fig. 2.8

3 - TEST TECHNIQUES

Through inventive test techniques, AMD-BA have succeeded :

. to increase the safety of flights,
. to reduce the number of flights,
. to shorten development times,
. to decrease the number of measured parameters and the acquisition rates.

We subdivised these techniques into several categories :

. Data acquisition and processing in real time,
. Model technique for data processing,
. Integration tests of weapon systems and armament,
. Countermeasures testing.

A time history of our experience gained on these tasks is epitomized on figure 3.0.1. The keydates are :

1960 : Model technique applied to performance analysis,

1965 : Modelized processing of handling qualities. Acquisition of magnetic tapes of airborne analog recorders,

1968 : Flutter analysis technique,

1968 : Real-time acquisition of telemet-y,

1974 : Integration of the weapon system on stimulable test bench,

1978 : Countermeasures testing in anechoic chamber,

1979 : Man-machine interface on simplified simulators.

3 - TECHNIQUES D'ESSAIS

Des techniques d'essais originales ont permis :

. d'accroître la sécurité des vols,
. diminuer le nombre de vols,
. diminuer les délais de mise au point,
. diminuer le nombre d'avions en essais,
. diminuer le nombre de paramètres mesurés et les cadences d'acquisition.

Nous les avons classées en plusieurs rubriques :

. l'acquisition et traitement de données en temps réel,
. l'exploitation par la méthode du modèle,
. les essais d'intégration de système de navigation et armement,
. les essais d'intégration des armements,
. les essais de contre-mesures.

Nous donnons sur la planche 3.0.1 l'historique de notre expérience sur ces sujets. Les dates importantes sont :

1960 : exploitation des performances par la technique du modèle,

1965 : exploitation des qualités de vol par la technique du modèle, acquisition de données sur bandes d'enregistreurs magnétiques embarqués,

1968 : méthode d'exploitation du flutter,

1968 : acquisition temps réel de la télémesure,

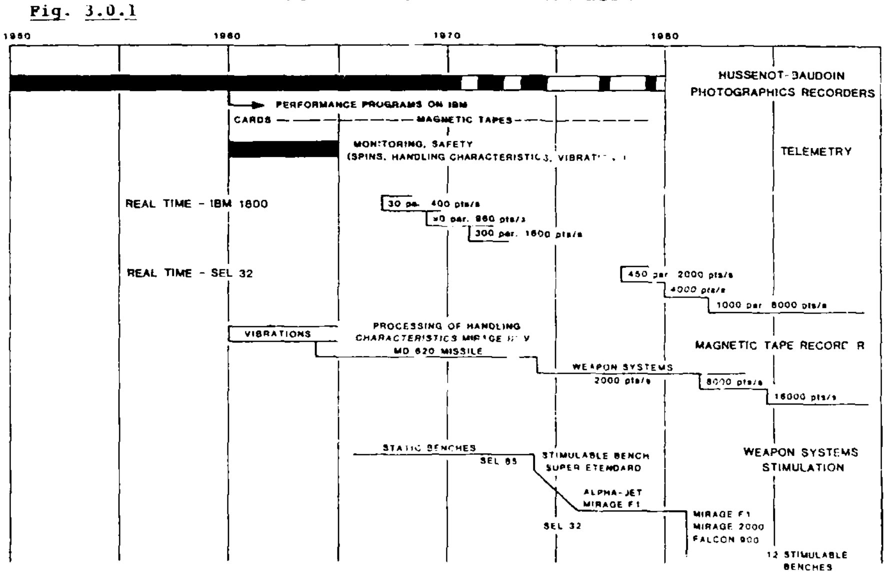
1974 : intégration du système d'armes sur banc stimulable,

1978 : essais de contre-mesures en chambre anéchoïde,

1979 : interface homme/machine sur simulateurs simplifiés.

EXPERIENCE ON TEST TECHNIQUES

Fig. 3.0.1



3.1 - Data Acquisition and Processing in Real-Time

3.1.1 - Goal

The primary concern of Flight Test must be SAFETY. To reach this aim, a certain number of precautions are to be taken :

. Constitution of a closely cooperating Test Team,headed by a Flight Test Engineer who must have a solid knowledge of his machine to assist the pilot in test conduct.

. A perfect knowledge of the limitations and the possible hazards.

Test teams have paid a heavy tribute to the technical progress of aircrafts, despite all the necessary forethought and precautions. One safety rule consists in installing on board only the people indispensable for the conduct of the mission, bearing in mind that accidents frequently happen during tests that are not considered critical.

. A test program must be established with a maximum of care, and the pilot, well briefed, signs it, hereby assuming the responsibility of the flight, aided by the Flight Test Engineer who monitores the various safety parameters. To increase the efficiency of this supervision, AMD-BA have developed, since 1968, a real-time acquisition technique through telemetry.

In the beginning focused on conventional monitoring of risky flights :

. Flight envelope, flutter, structural loads, handling qualities,

. High AoA, spin tests,

. Engine testing

this technique has been, progressively, extended to all flights of our aircrafts and to the monitoring of all circuits and systems.

3.1.2 - Principles (Figure 3.1.1)

The data collected aboard are multiplexed. A first data frame is composed through measurements in small passband, less than 20 Hz. These measurements are not only conventional analog information but, to an ever--increasing extent, digital data that are circulating on asynchronous buses and consequently at their proper rate. Multiplexing is performed through a digital PCM serial message (Pulse Coded Modulation).

A second data frame is constituted through measurements in intermediate pass-band, below 200 Hz. In particular, these are measurements of vibrations, accelerometers structural loads sensors ... Analog multiplexing is used to this end.

3.1 - Acquisition et traitement de données en temps réel

3.1.1 - But

Le premier souci des essais en vol doit être la sécurité. Elle est d'abord obtenue par un certain nombre de précautions :

. La constitution d'une équipe d'essais bien soudée avec un ingénieur qui doit bien connaître son avion pour aider le pilote dans la conduite de l'essai.

. La parfaite connaissance des limites et des risques pouvant être encourus.

Les équipages d'essais ont payé un lourd tribut aux progrès techniques des avions, même en prenant toutes les précautions nécessaires. Une sécurité consiste à ne mettre à bord que le personnel nécessaire à la mission en se souvenant que les accidents arrivent souvent dans des essais qui n'étaient pas jugés critiques au départ.

. Les programmes d'essais doivent être préparés soigneusement. Le pilote bien informé signe son ordre d'essais et prend alors la responsabilité du vol, l'ingénieur apportant son aide en surveillant les différents paramètres de sécurité.

. Pour augmenter l'efficacité de ce contrôle les AMD-BA ont développé depuis 1968 une technique d'acquisition temps réel par télémesure. Au début, centrée sur des surveillances classiques pour des vols à risques :

. Domaines de vol, flutter, efforts, qualités de vol,

. Essais de grandes incidences, vrilles,

. Essais moteurs.

elle a été progressivement étendue à tous les vols de nos avions et à la surveillance de tous les circuits.

3.1.2 - Principes (planche 3.1.1)

Les données acquises à bord sont multiplexées. Un premier message est constitué par des mesures à faible bande passante, inférieure à 20 Hz. Ces mesures sont non seulement des informations analogiques classiques mais de plus en plus des informations numériques qui circulent sur des Bus asynchrones et qui ont donc leur rythme propre. Le multiplexage est effectué par un message numérique série (PCM).

Un deuxième message est constitué par des mesures à bande passante intermédiaire, inférieure à 200 Hz. On trouve en particulier les mesures vibratoires, accéléromètres, efforts... Le multiplexage est effectué en analogique.

A third message covers pilot's voice transmission.

These three data frames are multiplexed again to be telemetered to the ground. For safety reasons, to provide for range losses, they are also recorded aboard on the analog magnetic tape recorder, which is moreover used for additional digital or analog recordings, especially in large pass-bands, in general smaller than 2,500 Hz. A ground Telemetry Reception Center (implemented by the French Official Flight Test Center) collects the telemetered data and transmits the complete multiplex to AMD-BA. Back-up antennas are also installed in our Ground Station.

The three messages are separated :

. During the test phases dedicated to"vibrations", the different signals of the "vibrations" multiplex are separated :

1) they are transmitted to the Flight Control Room and to the specific "flutter" processing room for quick display versus time on analog plotters and in form of spectra for two signals via an analog real-time analysor,

2) they are filtred and converted into digital units for further processing by a computer which carries out the spectral analyses of all signals (15), with results upon completion of excitation, and also performs a modal analysis, providing the frequencies and dampings of all strutural modes the results being available within only 10 to 15 seconds after the test.

. The different words of the PCM message are dissociated by synchronizers :

1) they are transmitted to the Flight Control Room via a decommutator enabling simple analog displays in analog output format and digital words (presentation of event signals) and in the form of bar-graphs for all parameters ; 32 can be converted into engineering units using a first-order calibration :

$$ax + b \text{ or } \frac{a}{x} + b$$

2) they are sent to the computer on line. This computer takes into account <u>all</u> the data transmitted, i.e. currently 8,000 words/second (but with a capability of 16,000).They are converted into engineering units through polynominal calibrations or in segmented form.

Various treatments can be applied :

. Systematic computations throughout the flight of parameters of general interest. Examples : Corrected altitude, speed and Mach number, weight and center of gravity location, lift coefficient, true AoA,

Un troisième message est constitué par la phonie pilote.

Ces trois messages sont de nouveau multiplexés pour être transmis au sol par télémesure.

Par sécurité en cas de pertes de portée, ils sont également enregistrés à bord sur enregistreur magnétique analogique qui est utilisé pour d'autres enregistrements numériques ou analogiques à grande bande passante en général inférieure à 2.500 Hz.

La réception est effectuée au sol dans un centre de réception (mis en oeuvre par le CEV) qui nous transmet le multiplex complet. Des antennes de secours sont installées sur notre centre d'exploitation.

Les trois messages sont séparés :

. Dans les phases d'essais de vibrations les différents signaux du multiplex vibrations sont séparés :

1) transmis en salle d'écoute et en salle dédicacée aux essais flutter pour des visualisations rapides fonction du temps sur traceurs analogiques et sous forme de spectres pour deux signaux par un analyseur analogique temps réel.

2) filtrés et convertis en digital pour être traités par un ordinateur qui effectue les analyses spectrales de tous les signaux (15), résultats obtenus juste à la fin de l'excitation et une analyse modale fournissant les fréquences et amortissements de tous les modes structuraux en 10 à 15 secondes après l'essai.

. Les différents mots du message PCM sont dissociés par des synchronisateurs :

1) transmis en salle d'écoute par un décommutateur qui assure des visualisations simples sous forme de sorties analogiques et mots digitaux (présentation de tops) et sous forme de bargraphs pour tous les paramètres ; 32 peuvent être convertis en grandeur physique à l'aide d'un étalonnage du 1er ordre $ax + b$ ou $\frac{a}{x} + b$

2) transmis à l'ordinateur temps réel. Cet ordinateur prend en compte <u>toutes</u> les informations transmises soit actuellement 8.000 mots/s (avec une possibilité de 16.000). Elles sont traduites en grandeur physique par des étalonnages polynomiaux ou sous forme segmentée.

Différents traitements peuvent être effectués :

. Calculs systématiques pendant tout le vol de grandeurs d'intérêt général, exemples :
Altitude, vitesse et Mach corrigés
Masse et centrage, CZ, incidence vraie ...

. Computations adapted to the test under progress. :

All the collected and calculated data are stored on magnetic tape throughout the flight. They are also accessible in the Flight Control Room on graphics and alpha-numerical displays. The aim of these visualizations is to enable the *Flight Test Team to react* "on the spur of the moment" : great conviviality of data display arrangements tailored to test progress, all controls being housed in the Flight Control Room.

To improve the response times of the FTT, the computer monitors the limits and transmits three event-signal words of 16 bits. Each of these bits may assemble several items of information *of the same* type. Artificial intelligence modules are also capable of failure diagnostics.

The team has at its disposal three graphics display terminals with hard-copy capability, two A/N terminals and one graphics console enabling to reconstitute the operation of the aircraft circuits or its navigation.

Listings on printers and traces are available at the end of a flight for the debriefing.

To carry out all these functions, it is necessary to resort to a top-class computer with "real time" capability, the CPU used is the GOULD SEL 32.97.50.

The pilot's voice link is also transmitted to the Flight Control Room. A precious help, in particular for the weapon systems tests, is the transmission of video signals through a separate telemetry transmitter, enabling to visualize the pilot's Head-Up Display or the Head-Level and Head-Down CRT's.

3.1.3 - Gains yielded in Test Efficiency

Thanks to our facilities, we were in a position to prevent at least two mishaps which is another proof - should this still be necessary - for the cogency of the safety aspect, and the rentability of the investment consented.

This technique still holds other advantges :

Pilot's Assistance

Flight preparation has to be very meticulous and must be optimized. Real-time display of the results makes it possible to check the quality of the tests, to help the pilot in conducting them, thus enabling to increase test density. Consequently, in-flight time is decreased.

. Calculs adaptés à l'essai en cours.

Toutes les données acquises et calculées sont stockées sur bande magnétique pendant tout le vol. Elles sont également accessibles en salle d'écoute sur des écrans graphiques et alphanumériques. Le logiciel d'accès à ces visualisations est réalisé pour permettre une réaction rapide de l'équipe d'essais :

grande convivialité pour les modifications de présentation suivant le déroulement des essais, toutes les commandes étant en salle d'écoute.

Pour améliorer le temps de réponse de l'équipe, l'ordinateur effectue des surveillances de limites et transmet trois mots tops de 16 bits. Chacun de ces bits pouvant regrouper plusieurs informations de même type. Des modules d'intelligence artificielle peuvent également effectuer des diagnostics de pannes.

L'équipe dispose de trois consoles graphiques avec copiage, deux consoles alphanumériques et une console graphique permettant de reconstituer le fonctionnement des circuits de l'avion ou sa navigation.

Des listages sur imprimantes et tracés sont disponibles à la fin du vol pour le debriefing.

Pour réaliser toutes ces fonctions, il est nécessaire de disposer d'un ordinateur "temps réel" haut de gamme, le type utilisé est le GOULD SEL 32.97.50.

3) La phonie pilote est également transmise en salle d'écoute. Une aide importante, en particulier pour les essais de système d'armes, est réalisée par la transmission de signaux vidéo par un émetteur de télémesure séparé, ce qui permet de visualiser la tête haute pilote ou les autres visualisations têtes moyennes ou basses.

3.1.3 - Gains apportés sur l'efficacité des essais

Nos installations nous ont permis d'éviter au moins deux accidents ce qui - si cela était nécessaire - permettrait de démontrer l'aspect sécurité et la rentabilité de l'investissement consenti

Cette technique a également d'autres avantages :

Assistance au pilote

La préparation du vol doit être très soignée et optimisée. La présentation des résultats en temps réel permet de contrôler la qualité des essais, aider le pilote dans sa conduite des essais qui peuvent être plus denses. On diminue les heures de vol.

Reduction of Processing Time

The fact to "table" the important results at the end of a flight substantially increases the effectiveness of the debriefing with the pilote and enables to prepare the next flight immediately. Typical examples are spin and flutter tests. Thanks to on-line monitoring, it has become possible to increase the number of spin tests per flight and to concatenate flights. It is also feasible to carry out two envelope extension flights daily, since the results can be analyzed upon touch-down.

Permanent Monitoring of the Aircraft and its Test Installation throughout the Flight

The possibility to access all data in the Flight Control Room generally allows to explain all failures. Under these conditions, the ground engineering team can start work effectively on arrival of the aircraft in the hangar. The same applies for the test installation.

Active downtime between flights is reduced and, as a consequence, the rate of flight is increased and development on the whole is accelerated.

Integration of the Flight Test Team and Equipment Manufacturers in the Flight Follow-Up

The team following on line the flight in progress has a better understanding of the targeted goals, and the critical points which will be analyzed in depth off line. The pilot's debriefing is facilitated and complementary processing jobs can be undertaken at once.

3.1.4 - Realization

Figure 3.1.2 gives the block diagram of real-time acquisition in AMD-BA.

Two Air Bases are equipped with on line facilities :

- ISTRES : Main development Center,
- CAZAUX : Armament and external stores delivery testing.

From 1978 to 1988, more than 200 different aircraft have been developed. They belong to about 15 different families :

- MIRAGE III & 5
- MIRAGE IV
- JAGUAR
- ALPHA-JET
- SUPER-ETENDARD
- MIRAGE 2000
- MIRAGE 4000
- ATLANTIQUE ATL 2 and refurbished
- ATLANTIC
- RAFALE
- The large dynasty of MYSTERE FALCON comprising : FALCON 20, 10, 100, 200, 50, 900, GUARDIAN and GARDIAN.

In addition to these different types of aircraft, the number of aircraft under test is attributable to airframe renovations, and particularly to the large variety of distinctive weapon systems.

Diminution des temps d'exploitation

Le fait d'avoir tous les résultats importants à la fin du vol rend le debriefing avec le pilote plus efficace et permet de préparer immédiatement le vol suivant. Les exemples types sont les vrilles et les essais de flutter. Avec le temps réel on a pu augmenter le nombre d'essais de vrilles par vol et enchaîner les vols.
On peut également effectuer facilement deux vols de domaine par jour puisque les résultats peuvent être analysés dès la fin du vol.

Contrôle permanent de l'avion et de son installation de mesure pendant le vol

L'accès possible à toutes les données dans la salle d'écoute permet en général d'expliquer toutes les pannes. Dans ces conditions, le travail de l'équipe de piste peut être entrepris efficacement dès l'arrivée de l'avion dans le hangar. Il en est de même pour l'installation de mesure elle-même.

L'indisponibilité entre les vols est diminuée ce qui augmente la cadence et accélère la mise au point générale.

Intégration de l'équipe d'essais et des équipementiers au suivi du vol

L'équipe qui suit le vol en comprend mieux les buts, les points critiques qui feront l'objet des exploitations en temps différé. Le debriefing pilote est facilité et les exploitations complémentaires peuvent être entreprises immédiatement.

3.1.4 - Réalisation

La planche 3.1.2 donne le synoptique de l'acquisition temps réel.

Deux bases sont équipées :

ISTRES
centre principal pour la mise au point,

CAZAU[illegible]
pour [illegible] e[illegible]is armement et séparation de cha[illegible]

De 1978 à 1988, plus de 200 avions différents ont été mis au point. Ils appartiennent à environ 15 familles différentes :

- MIRAGE III et 5,
- MIRAGE IV,
- JAGUAR,
- ALPHA-JET,
- SUPER-ETENDARD,
- MIRAGE 2000,
- MIRAGE 4000,
- ATLANTIQUE ATL2 et ATLANTIC rénovés,
- RAFALE
- Toute la famille des MYSTERE-FALCON 20, 10, 100, 200, 50, 90, US GUARDIAN et GARDIAN.

En dehors de ces types d'avions différents le nombre d'avions en essais se justifie par des rénovations cellules et surtout par des systèmes d'armes différents.

The multi-role MIRAGE 2000 aircraft family comprises 14 different versions, distinguishing by :

- . the engine (2) SNECMA M 53-5 and M 53 P2
- . the radar (4) THOMSON-CSF RDM, RDI, RDY and ANTILOPE manufactured by ELECTRONIQUE Serge DASSAULT.
- . the weapon systems : French versions : 2000 DA and 2000 N Export versions.

For these variants, 600 external stores configurations have been validated. To perform these tests at a rate that may reach 15 flights daily, the following means are available :

- . 3 control Rooms for real-time follow-up at ISTRES,
- . 1 real-time Control Room at CAZAUX,
- . 2 mobile ground stations, one fully equipped for on-line operation, with computer, for tests abroad.

Figures 3.1.3 roughly outlines the computer network in the main AMD-BA Center at ISTRES.

In addition to the GOULD 32-97 real-time acquisition CPU, ISTRES accommodates :

- . 1 GOULD 32-87 for off-line processing and develpment of software programs,
- . 1 IBM 4341 computer for local processing tasks, with simplified Design Office programs,
- . 1 IBM 4381 computer for general management and creation of definition drawings of measurement installations, using CATIA CAD - CAM programs.

Pour la famille MIRAGE 2000, avion polyvalent, il y a 14 versions différentes compte tenu :

- . du moteur (2) SNECMA M 53-5 et M 53 P2
- . du radar (4) THOMSON-CSF RDM, RDI, RDY et ESD ANTILOPE.

des systèmes d'armes : français 2000 DA et 2000 N et versions "export".

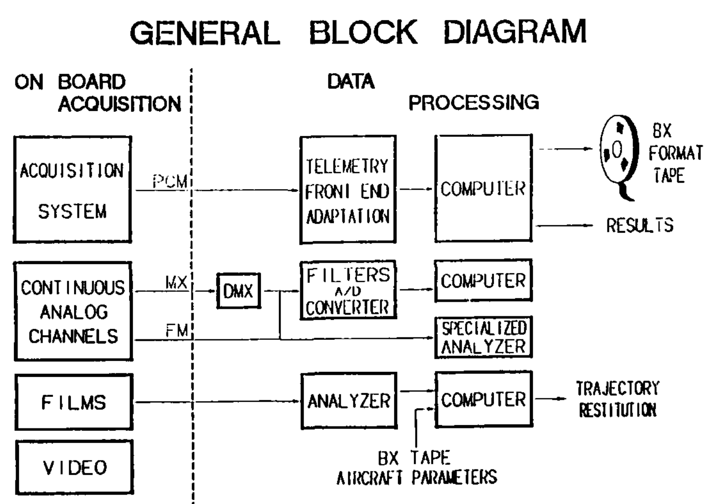
Pour ces différentes versions 600 configurations de charges extérieures ont été validées. Pour effectuer ces essais à une cadence pouvant atteindre 15 vols par jour les moyens sont les suivants :

- . 3 salles d'écoute temps réel à ISTRES,
- . 1 salle d'écoute temps réel à CAZAUX,
- . 2 stations mobiles dont une temps réel avec ordinateur pour des essais à l'étranger. La planche 3.1.3 donne la configuration informatique du centre principal d'ISTRES.

En plus des ordinateurs GOULD 32-97 d'acquisition temps réel on trouve :

- . 1 ordinateur GOULD 32-97 pour des exploitations en temps différé et les mises au point de programmes.
- . 1 ordinateur IBM 4341 permet d'effectuer des exploitations locales avec des programmes simplifiés du Bureau d'Etude.
- . 1 ordinateur IBM 4381 pour la gestion générale et les dessins définition des installations de mesure sur programme CATIA.

Fig. 3.1.1



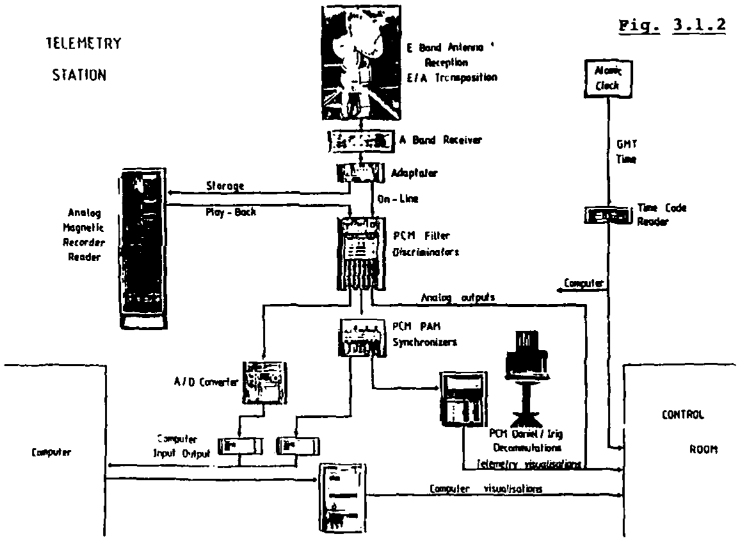


Fig. 3.1.2

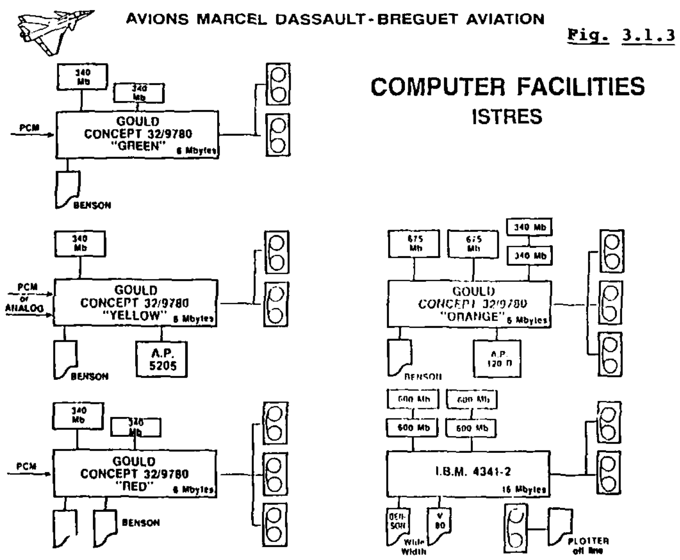


Fig. 3.1.3

3.2 - Airframe Tests, Model Technique

Conventional processing methods of performance, handling qualities, structural stresses and strains consist in collecting in-flight data and in processing them :

. retranscription into standard conditions at given weight and C.G. location,

. computation of aerodynamic coefficients,

. computation of aerodynamical loads.

To obtain sufficiently accurate results, it is mandatory to account for numerous flight parameters, for instance the Reynold's number, local air-inlet and profile drag variations as a function of engine conditions...

Interesting for performance measurements are flight manoeuvres at high incidence where the flight parameters may evolve very quickly.

Aircraft can no longer be considered as rigid structures, notions of aero-elasticity must be taken into consideration as well as the position of numerous control surfaces, which also depend on the flight conditions and the center of gravity location.

To determine the different coefficients it would be necessary to solve a great number of equations. Since 1960, we have been developing a methode called "model technique", the use of which is presently generalized for a large variety of applications.

3.2.1 - Principle

The principle is outlined on Fig. 3.2.1. The various parameters are collected as usual, but they are kept unchanged, except for corrections of calibrations and elementary computations, such as the mean efficiency of air intakes. These data are transmitted to the Design Office computer where is stored a mathematical aircraft model for performances, handling qualities, structural loads, ... Flight conditions and aircraft inputs (control surfaces deflections) logged in flight are entered into the computer, and model response is then compared to the actual aircraft response.

If they diverge, the model coefficients are modified automatically in order to minimize these differences and, through deduction, a new set of coefficients is established. Test points are then selected to account for important external influences :

. Altitude,
. Speed,
. Mach number,
. Dynamic pressure,
. Angle of attack,

3.2 - Essais cellule technique du modèle

Les méthodes classiques d'exploitation des performances, qualités de vol, efforts et contraintes de structure consistent à faire l'acquisition des données de vol et de les traiter :

. retranscription en conditions standard à masse et centrage donnés,

. calcul de coefficients aérodynamiques,

. calcul des efforts aérodynamiques,

Pour avoir des résultats suffisamment fins, il est nécessaire de tenir compte de nombreux paramètres de vol comme par exemple le nombre de Reynold, des variations de traînées locales d'entrée d'air et culot en fonction des conditions moteur ...

Les performances intéressantes sont les évolutions à grande incidence ou les paramètres de vol peuvent évoluer rapidement.

Les avions ne peuvent plus être considérés comme des structures rigides, il faut faire intervenir l'aéro-élasticité et des positions de nombreuses gouvernes qui dépendent également des conditions de vol et du centrage.

L'obtention des différents coefficients demanderait la résolution d'un grand nombre d'équations. Depuis 1960, nous avons développé une technique dite du modèle, cette technique est maintenant généralisée à beaucoup de sujets.

3.2.1 - Principe

Le principe est donné en planche 3.2.1.

On effectue également l'acquisition des différents paramètres mais qui sont conservés en leur état mis à part des corrections d'étalonnages et des calculs élémentaires comme un rendement moyen d'entrée d'air.

Ces informations sont transmises à l'ordinateur du Bureau d'Etude où se trouve un modèle mathématique de l'avion.

On entre les conditions de vol et les ordres avion (positions de gouvernes) enregistrés en vol et on compare la réponse du modèle avec la réponse avion.

En cas d'écart, on modifie automatiquement les coefficients du modèle de façon à minimiser ces écarts et on en déduit un jeu de nouveaux coefficients. Si l'on sélectionne des points d'essais pour tenir compte des influences extérieures importantes :

. Altitude,
. Vitesse,
. Mach,
. Pression dynamique,
. Incidence,

In this way it is possible to update the model that now allows to output the new performances and limitations within the whole flight envelope and, moreover, to check all possible pilot's inputs, as well as the C.G. configurations.

When the aircraft is fitted with an electric flight control system, it is possible to "inject" minutely chosen and calibrated orders directly into the servo-actuators, so-called "stimili" - without any intervention of the pilot. This signal is optimized by the computer, whose processing is adjusted to the coefficients to be determined.

3.2.2 - Advantages

This method is a reliable means to obtain more refined and more comprehensive results since the complete pass band of interest is scanned. It enables to carry out very quick flight manoeuvres of the type performed in air combat. An example of processing is given on figures 3.2.2 and 3.2.3 for handling qualities. Figure 3.2.2 compares the flight and the original model (roll and sideslip) and figure 3.2.3. shows the final result, after updating of the model.

This technique permits to reduce the number of test points and, consequently, the number of flights.

Moreover, it is possible to process the same "stimuli" in several different ways.

A first signal in the maximum band width possible through the servo-actuators enables to check the frequences and structural dampings for flutter prevention. It can also be used for fine tuning of the electric flight control system :

- determination of aircraft response for each control surface deflection,
- computation of the transfer function of aircraft response to servo-actuator inputs and servo-actuator response,
- it is thus possible to obtain gain margins and critical frequences.

A low-frequency type signal gives access to the aerodynamical coefficients (figures 3.2.2 and 3.2.3) and also to the structural loads measurements.

Figure 3.2.4 exemplifies 2 wing load measurements. The flight is represented by + and the model is plotted as a continuous line. The model shown has already been updated with regards to the aerodynamical coefficients. This figure also highlights that our method allows to reduce the acquisition rate. It is possible to validate a model even if the measurement is slightly scattered, judged by the deviations. But, if it were necessary to apply derivational processing on the measurement, implying smoothing, the acquisition rate would have to be increased.

on obtient un modèle recalé qui permettra de donner de nouvelles performances et limitations dans tout le domaine de vol et avec un contrôle de toutes les entrées possibles du pilote et les configurations de centrage.

Lorsque l'avion est équipé de commandes de vol électriques, il est possible d'injecter des ordres bien sélectionnés et calibrés, directement en ordres servocommandes sans l'intervention du pilote, appelés stimuli. Ce signal est optimisé par l'ordinateur d'exploitation suivant les coefficients recherchés.

3.2.2 - Avantages

Cette méthode permet d'avoir des résultats plus affinés et plus complets en balayant toute la bande passante intéressante. Elle permet d'effectuer des évolutions rapides du type combat.

Nous donnons un exemple d'exploitation des qualités de vol en planches 3.2.2 et 3.2.3. Sur la planche 3.2.2 on compare le vol et le modèle origine (*roulis et dérapage*) et sur la planche 3.2.3, le résultat final après recalage du modèle.

Cette méthode permet de réduire le nombre de points d'essais donc de vols.

Il est également possible, avec le même stimuli, de conduire plusieurs exploitations.

Un premier signal de bande passante maximum possible par les servocommandes permet de contrôler les fréquences et amortissements structuraux pour la prévention du flutter. Il permet également d'être utilisé pour la mise au point des commandes de vol électrique :

- détermination de la réponse avion au braquage de chaque gouverne,
- calcul de la fonction de transfert réponse avion sur les ordres servocommandes et réponse servocommandes,
- on obtient ainsi les marges de gain et les fréquences critiques.

Un type de signal à basse fréquence permet d'accéder aux coefficients aérodynamiques (planches 3.2.2 et 3.2.3) mais également aux mesures de structure.

La planche 3.2.4 représente deux efforts voilure, le vol est représenté par les "+" et le modèle, en tracé continu. Le modèle correspondant est déjà recalé du point de vue coefficients aérodynamiques

Cette planche met également en évidence que cette méthode permet de diminuer la cadence d'acquisition. Il est possible de valider un modèle même si la mesure est un peu dispersée en jugeant les écarts. Par contre, s'il fallait faire des traitements sur la mesure du type dérivation, nécessitant un lissage, il faudrait augmenter la cadence.

The model technique allows to decrease the number of measurement points. Let us take as example the structural loads : the model was created through computation on finite elements. With the help of this model, significant measurement points are selected that permit to close in upon in-flight loads.

On the RAFALE demonstrator, to make up for the lack of a static test cell, 300 gauges were bonded onto the airframe for a ground "proof test" with a 0.8 charge, to update the model. 200 among these gauges had been wired to be "airworthy", but the flight envelope was covered with only 90 test points. The others were used for 3 flights dedicated to general structural studies. With the experience gained through the development of the MIRAGE 2000, MIRAGE 4000, FALCON 900 and RAFALE A aircraft, we are in a position to demonstrate that it is possible to success fully master a development task with a data flow of 128 kbits/second, to which are to be added vibration measurements.

RAFALE A exhibited the efficiency of the definition, prediction and test techniques adopted by Avions Marcel DASSAULT-BREGUET AVIATION by completing 153 flights in a single year (figure 3.2.5).

La technique du modèle permet de diminuer le nombre de points de mesure. Par exemple dans le cas des efforts : le modèle a été créé par des calculs par éléments finis. Avec ce modèle on sélectionne les points de mesure significatifs permettant de remonter aux charges en vol.

Sur RAFALE A - qui ne disposait pas de cellule d'essais statiques - il a été collé 300 jauges pour un essai d'épreuve au sol à charge 0,8 recalant ce modèle ; 200 de ces jauges ont été câblées bonnes de vol mais le domaine de vol a été couvert avec 90 mesures. Les autres ont été utilisées au cours de 3 vols spécialisés pour les études générales structure.

Avec l'expérience acquise sur la mise au point, des avions MIRAGE 2000, MIRAGE 4000, FALCON 900 et RAFALE A, nous pouvons démontrer qu'il est possible d'effectuer une mise au point avec un débit d'information de 128 Kbits/s auquel il faut ajouter les mesures vibratoires.

RAFALE A a démontré l'efficacité des méthodes de définition, prévisions et essais en effectuant en une année 153 vols (planche 3.2.5).

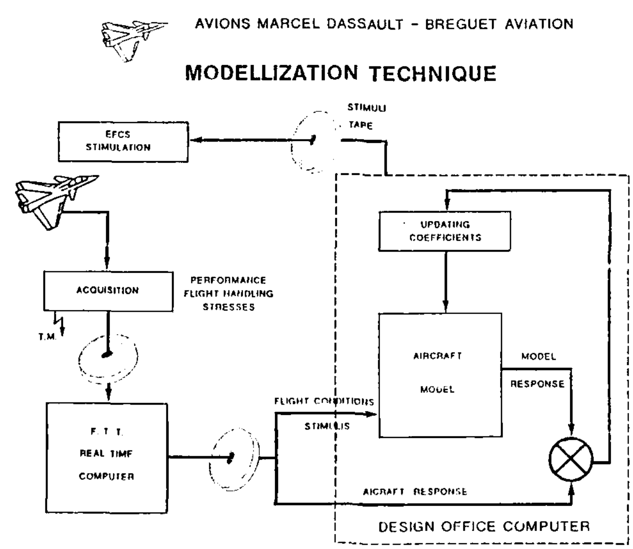


Fig. 3.2.1

RAFALE : EXAMPLE OF AIRCRAFT MODEL AUTOMATIC CORRECTION
ORIGINAL MODEL RESPONSE COMPARED
TO THE FLIGHT RESULT

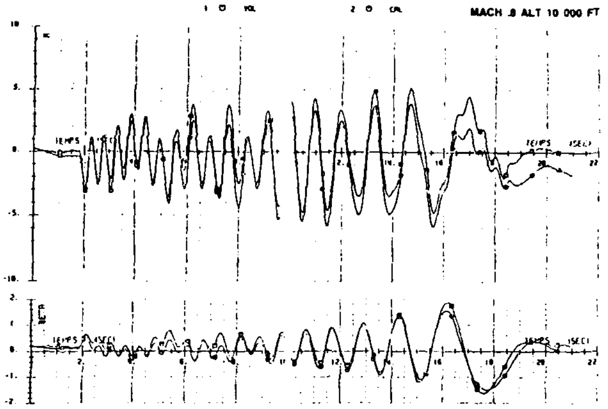


Fig. 3.2.2

RAFALE : EXAMPLE OF AIRCRAFT MODEL AUTOMATIC CORRECTION
CORRECTED MODEL RESPONSE COMPARED
TO THE FLIGHT RESULT

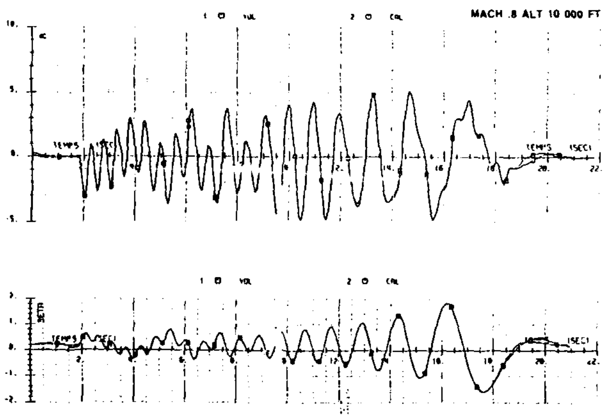


Fig. 3.2.3

RAFALE STRENGTH MEASUREMENTS

Comparison Flight /Updated model

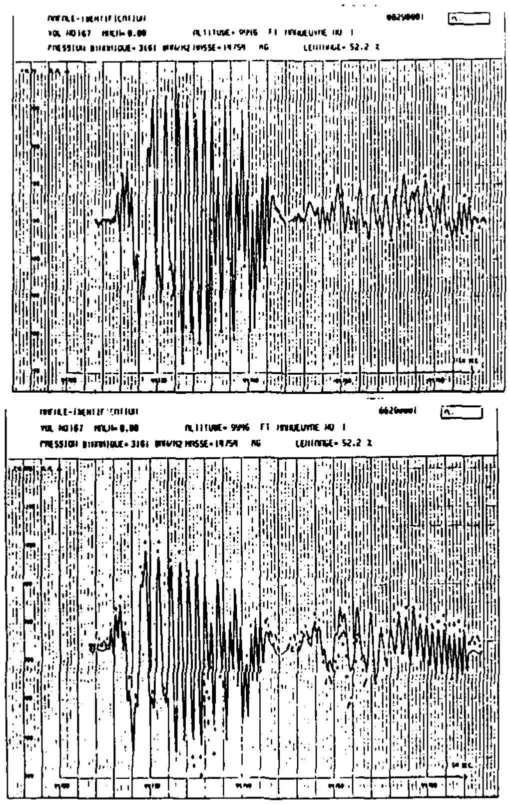
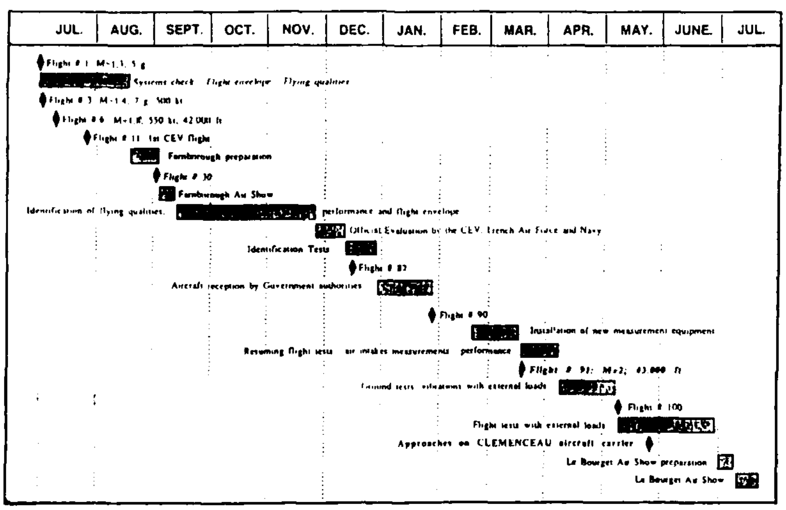


Fig. 3.2.4

Fig. 3.2.5

RAFALE : ONE YEAR OF FLIGHT TESTING



3.3 - Systems integration - stimulation technique

The development cycle of a system, schematized on Fig. 3.3.1, is similar to that of the aricraft it-self.

The advent of equipment with digital outputs, integrated into complex systems with computerized management, makes the *explosure of operational malfunctions* and design anomalies more difficult and, as a result, the adoption of strict development and integration rules has become an almost inevitable must.

The equipment must have undergone beforehand stand-alone tests in the manufacturer's premises, complemented, if judged necessary, by increasingly generalized tests on service aircraft.
The work of AMD-BA begins with the arrival of the equipment.

The newcomers are mounted on a test bench that reproduces as accurately as feasible, the aircraft wirings. The test bench allows to check :

. the electrical connections,
. the equipment performance, its sensitivity to electrical power generation,
. input/output sensitivity of parameters visualized on pilot's displays,
. the failure logics.

These conventional tests are no longer sufficient for the new integral systems intralinked through bus. Only a few problems can be apprehended in this way, because of equipment interfaces and logics inconsis*tencies and the limited number of sampling* possibilities. More particularly, the various equipment must be operated with correlated digital inputs.

The problems linked to the dynamic variable only appear during in-flight operation : filtering, extrapolation, noise, dynamic accuracy.

In this case, the analysis and the solution of problems found are difficult :

. Difficulty to record the right parameters (or the key parameter). The means on board of the aircraft are limited.

. The pilot sometimes fails to elucidate certain malfunctions.

. It is necessary to carry out new flights for each modification and it is difficult to repeat exactly the same flight configuration.

The above considerations incited us to imagine a new integration technique.

3.3.1 - Principles

The block diagram of a standard-type system is plotted on Figure 3.3.2.

The aircraft evolves in an external environment : flight conditions, flight manoeuvres, target, threats, terrain condition.

3.3 - Intégration de système - La stimulation

Le cycle de développement d'un système présenté sur le planche 3.3.1 est semblable à celui du développement de l'avion lui-même.

L'arrivée d'équipements à sorties numériques intégrés dans des systèmes complexes avec gestion par calculateurs rend plus difficile la détection d'anomalies de fonctionnement ou de conception et rend plus impératif l'adoption de règles de mise au point et d'intégration.

Les équipements doivent avoir effectué au préalable des essais individuels chez le fabricant *avec* suivant le *cas*, et de façon de plus en plus générale, des essais sur avion de servitude.

Le travail des AMD-BA *commence à* leur réception.

Ils sont montés sur un banc qui reproduit au mieux les liaisons avion. Ce banc permet de contrôler :

. les liaisons électriques,
. les performances de l'équipement,
. sa sensibilité à la génération électrique,
. la sensibilité des sorties de paramètres vers les visualisations pilote,
. les logiques de pannes.

Ces essais classiques sont devenus insuffisants avec les nouveaux systèmes intégrés avec liaisons par Bus.

Seulement quelques problèmes peuvent être ainsi appréhendés du fait de problèmes d'interfaces, d'équipements, de logiques, échantillonnages limités en nombre. Il faut en particulier faire fonctionner les équipements avec des entrées digitales corrélées.

Les problèmes liés à la dynamique *n'apparaissent qu'en vol* : filtrage, extrapolation, bruit, précision dynamique.

Dans ce cas l'analyse et la solution des problèmes *rencontrés sont difficiles* : difficulté d'enregistrement des bons paramètres (ou du paramètre clef). Les moyens à bord sont limités.

Le pilote a parfois des difficultés d'interprétation de certaines anomalies. Il est nécessaire de refaire de nouveaux vols pour toutes modifications et il est difficile de répéter la même configuration de vol.

Ce sont ces considérations qui nous ont conduit à imaginer une nouvelle technique d'intégration.

3.3.1 - Principes

Le schéma d'un système type est donné en planche 3.3.2.

L'avion se trouve dans un environnement extérieur : conditions de vol, évolutions, cible, menaces, terrain.

The system is composed of sensors restoring this environment and the status of the internal sub-systems.
The whole data stream transits towards computers on buses.

The results of the computations that are adapted to the functions selected by the pilot, are sent to the pilot's displays in the cockpit and, as orders, to the weapons.

The method developed by AMD-BA is epitomized on Figure 3.3.3.

The actual system's components are mounted on the wiring rig - it is consequently possible to carry out the customary first tests specified before.

This test bench can be stimulated through a computer on the basis of correlated information rendering account of the external environment "seen" by the sensors. These information are injected into the test bench as stand-in for the sensors by means of the stimulation interface.

The data may proceed from different sources :

Since these tests must be carried out prior to the system's aircraft integration capability tests, the data may originate from recordings made on another aircraft type, from a simulation tape or may be a short simulation created on the same computer.

In the following, it is possible to play back flight phases recorded on the fighter aircraft the system is intended for. In this way it is possible to create "authentic" play-backs that are reproducible.

A hurdle to clear was the acquisition of necessary measurements on board of the aircraft. In 1974, the magnetic tape recorders were capable of only 2000 words per second. The task to tackle was to merge the analog data and the digital information collected on asynchronous buses.

To limit the acquisition rate, it was mandatory to sort the data on the bus and to limit their rate to that of the pass-band of the parameter and not to that of the bus. To reconstitute the proper rate on the test bench, acquisition had to meet a time accuracy requirement of 1 ms.

The solution to be found was to simplify acquisition and the means of processing leading to the "stimuli" tape.

These reflections converged in the definition of the "DANIEL" acquisition system, whose PCM output message, in IRIG standard, has become a standard in FRANCE known as PCM DANIEL standard.

Le système est constitué de capteurs rendant compte de cet environnement et de sous systèmes internes.

L'ensemble des données transite vers des calculateurs par des Bus. Les résultats de calcul, qui sont adaptés à des fonctions sélectionnées par le pilote, sont envoyés en visualisation pilote et en ordres vers les armements.

La méthode développée par AMD-BA est donnée en planche 3.3.3.

Les équipements réels du système sont montés sur le banc de câblage qui permet d'effectuer les premiers essais classiques mentionnés, ci-dessus.

Ce banc peut être stimulé par un ordinateur par des informations corrélées rendant compte de l'environnement extérieur, vu par les capteurs. Ces informations sont injectées sur le banc à la place des capteurs avec la même forme grâce à l'interface de stimulation.

Ces informations peuvent provenir de différentes sources :

. ces essais devant être effectués avant l'avionnage sur l'avion d'armes, elles peuvent provenir d'enregistrements sur un autre type d'avion, de bandes de simulation ou d'une petite simulation réalisée sur le même ordinateur.

Il est possible ensuite de rejouer des phases de vol enregistrées sur l'avion d'armes. On effectue ainsi de réels play-back qui peuvent être reproduits.

Un problème a été l'acquisition des mesures nécessaires à bord. En 1974, les enregistreurs magnétiques étaient limites à 2000 mots par seconde. Il s'agissait de mélanger des informations analogiques et des informations numériques acquises sur des Bus asynchrones.

Pour limiter la cadence d'acquisition, il a été nécessaire de faire des tris sur les informations Bus et de limiter leurs cadences à celles correspondant à la bande passante du paramètre et non celles du Bus.

Pour reconstituer la bonne cadence au banc, il fallait que cette acquisition soit effectuées avec une précision en temps de 1 ms.

Il fallait enfin simplifier l'acquisition et les traitements qui conduisent à la bande de stimuli.

Ce sont ces différentes considération qui nous ont conduits à la définition du système d'acquisition DANIEL dont le format du message PCM de sortie, au standard IRIG, est devenu un standard en FRANCE sous la dénomination de PCM DANIEL.

The exigencies of stimulation were the primary incentives of our choice, in 1974 already, to opt for SYSTEM ENGINEERING SEL 85 computers which, in the following, were replaced by GOULD SEL 32/75, 32/77 computers, and presently our CPU's are SEL 32/97.

In 1974, stimulation was based on recordings of 2000 words of information per second. Presently, 8000 words per second (128 Kbits/second) are recorded, including roughly thousand useful parameters. It has never been necessary to log the totality of the buses in flight.

3.3.2 - Advantages of Stimulation

The assets of this technique can be recapitulated as follows :

The stimulable test bench makes it possible to study dynamic phenomena. Stimulation is a tool that is independent of the aircraft and that can intervene effectively ahead of the flight phases. It thus enables to pinpoint anomalies of equipment or specifications and in links prior to aircraft integration. Consequently, it contributes to economize flights.

It also helps (in association with the OASIS Center) to define the specifications of systems, from the outset of the flowchart elaboration stage.

It is a processing tool, not only *allowing a better comprehension* of problems found in flight and fault diagnostics (aircraft or system origin), but also assisting in solving these problems and to validate modifications with the same flight profile.

For trouble hunting, since a given flight phase is reproducible at will, stimulation makes it easier to access parameters of interest and enables to use ground means, that are less costly and more powerful than airborne installations. Consequently : economy of flights, equipment expenditures and data storage space.

The same flight profile used for normal modes also serves for the evaluation of back-up and emergency modes.

We also take advantage of the stimulation benches to instruct and train the test teams before flights.

Cost-saving through stimulation is very substantial but hard to quantify - in quintessence it is enough to know that without this tool it would not have been possible for us to develop the great number of systems we put on board of our aircraft and to offer our clients a panoply of armements adapted to their needs, even if only small series were ordered.

Ce sont les exigences de la stimulation qui nous ont fait choisir dès 1974 les ordinateurs System Engineering SEL 85 qui ont par la suite été remplacés par des GOULD SEL 32/75, 32/77 et maintenant 32/97.

La stimulation a pu être réalisée au départ en 1974 avec l'enregistrement de 2000 informations par seconde. Actuellement l'enregistrement a été porté à 8.000 mots/s (128 Kbits/s) comprenant une centaine de paramètres utiles. Il n'a jamais été nécessaire d'enregistrer la totalité des Bus en vol.

3.3.2 - Avantages de la stimulation

En résumé les avantages apportés par cette technique sont les suivants :

Le banc stimulable permet d'étudier des phénomènes dynamiques. C'est un outil indépendant de l'avion et qui peut travailler avant les phases de vol. Il permet ainsi de détecter des anomalies d'équipements ou de spécification dans les liaisons, avant l'avionnage. Ceci se traduit par une économie de vols.

Il est également d'une aide (avec le *centre OASIS) pour définir les* spécifications de systèmes dès la première phase de l'organigramme.

C'est un outil d'exploitation. Il permet de mieux comprendre les problèmes rencontrés en vol et d'en déterminer leur origine (problème avion ou problème système, de résoudre ces problèmes et de valider la modification avec le même profil de vol.

Pour cette recherche, la phase de vol étant réproductible à volonté, il est plus aisé qu'en vol d'avoir accès aux paramètres intéressants et d'utiliser des moyens sol moins coûteux et plus puissants. Economie de vol, de coût d'équipements et de stockage de données.

Les modes secours sont évalués avec le même vol que les modes normaux.

Les bancs sont également utilisés pour la formation des équipes d'essais avant les vols.

Les économies très importantes sont difficilement chiffrables, il suffit de savoir que sans cet outil il nous aurait été impossible de développer le nombre de systèmes que nous possédons sur nos avions et d'offrir à nos clients une panoplie d'armements adaptés à leurs besoins même avec une faible série commandée.

3.3.3 - Material Realization

Each aircraft system is installed on a test bench in the AMD-BA center at *BRETIGNY*.
Currently, 14 systems are under development on 18 comprehensive and 7 partial test rigs. They are stimulated through 4 GOULD SEL 32/77 and 1 GOULD SEL 32/97 computers.

Stimulation has been increasingly extended to the entire system by stimulating certain sensors (radar, inertial navigation unit). Radars and missiles coupled to the test bench enable to make real target acquisitions and to control missiles lock-on.

Figure 3.3.4 is the schematic diagram of a test bench. A more detailed study reveals that some laboratories are reserved to equipment manufacturers who thus preserve their development pregoratives on their own test equipment.

3.3.4 - OASIS (Specifications Design Tool)

A delicate problem to bring to a successful issue concerns the development of the symbology to present to the pilot. With conventional indicators, it was possible to await the cockpit mock-up phase to optimize their layout. The arrival of head-up displays complicated this adaptation. Each modification implies a software adjustment and "reaction" lead times are very long ...

We defined a very simple tool, christened DAISY (Synthetic Imagery Animation System) enabling us to display on screens (on displays assigned to the Ground Station in the beginning) the proposed symbology and to modify it easily. This helped to define the General Specifications for sighting heads more efficiently.

The application spectrum of this tool was optimized in the OASIS Center, where are implemented all feasible output varieties (head-up or head-down displays (cathode ray tubes). On account of its role in aircraft definition, the responsibility of this Center was given to the Design Office, but its accommodation in the Flight Test Division facilitates the contacts with the pilots and test teams and cuts down their times of unavailableness.

The computers used are of GOULD brand : 1 GOULD 32/97 and 2 GOULD 32/67.

3.3.3 - Réalisation

Chaque système avion est installé sur un banc à BRETIGNY. 14 systèmes sont actuellement en développement avec 18 bancs complets et 7 bancs partiels. Ils sont stimulés par 4 ordinateurs GOULD SEL 32/77 et 1 GOULD SEL 32/97.

La stimulation a été de plus en plus étendue au système en stimulant certains capteurs (radar, centrale à inertie). Des radars et missiles couplés sur le banc permettent de faire de réelles acquisitions de cibles et contrôler les accrochages missiles.

La planche 3.3.4 donne le synoptique d'un banc. On remarquera que des laboratoires sont réservés aux équipementiers qui conservent leur responsabilité de mise au point avec leurs équipements d'essais propres.

3.3.4 - OASIS

Un problème difficile à mettre au point concerne les figurations à donner au pilote. Avec les indicateurs classiques on pouvait attendre la phase maquettage cabine pour les répartir au mieux. L'arrivée des viseurs tête haute rendait plus difficile cette adaptation. Chaque modification se fait par logiciel et les délais de réaction sont très longs.

Nous avons défini un outil simple appelé DAISY qui permet de présenter sur écrans (d'abord ceux affectés au vol en salle d'écoute) les visualisations proposées et les modifier facilement. On peut ainsi mieux définir le cahier des charges viseur.

Cet outil a été généralisé dans le centre OASIS (Outil d'Aide à la Spécification) avec toutes les variétés de sortie, tête haute ou tête basse (écrans cathodiques). Compte tenu de sa fonction dans la définition avion, la responsabilité en a été donnée au Bureau d'Etude mais sa localisation aux Essais en Vol facilite les contacts avec les pilotes et équipes d'essais et diminue leur temps d'indisponibilité.

Les ordinateurs utilisés sont du type GOULD : 1 GOULD 32/97 et 2 GOULD 32/67.

DEVELOPMENT CYCLE OF A NAVIGATION ATTACK SYSTEM

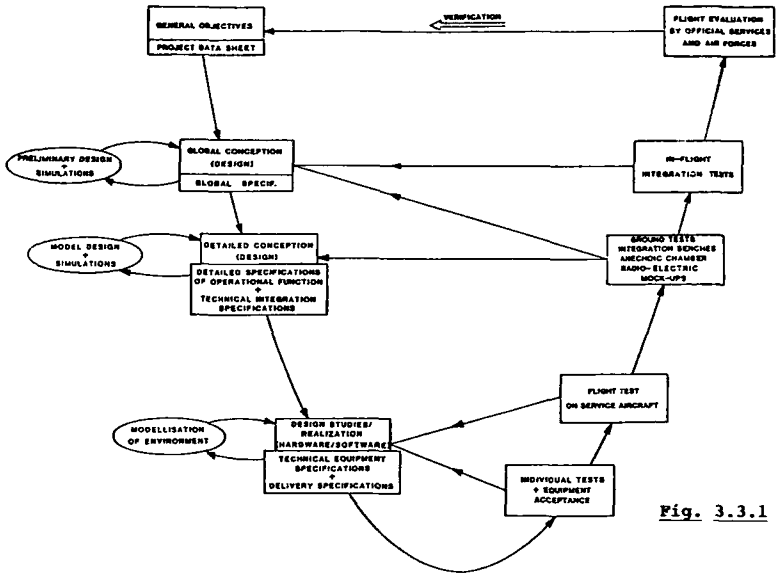


Fig. 3.3.1

INTEGRATION OF NAVIGATION AND ATTACK SYSTEMS

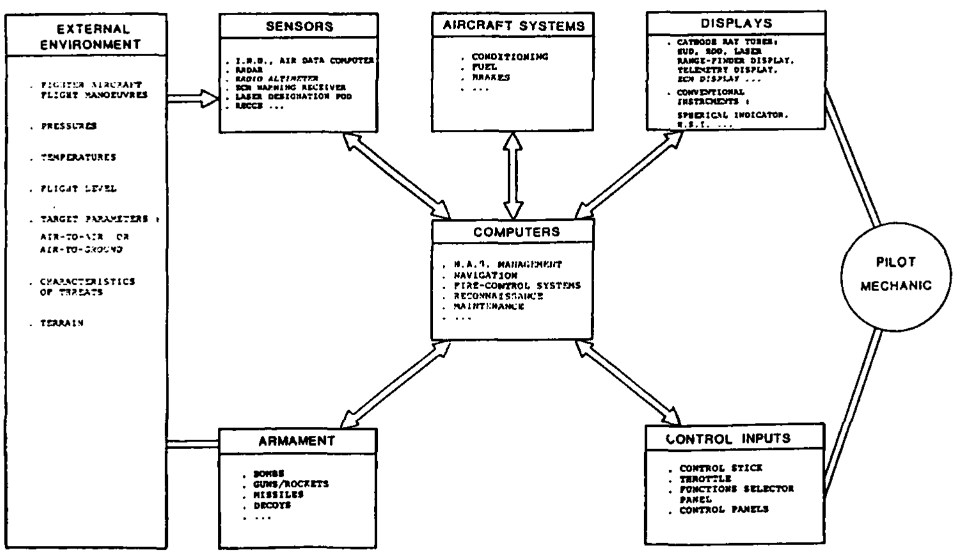


Fig. 3.3.2

Fig. 3.3.3.

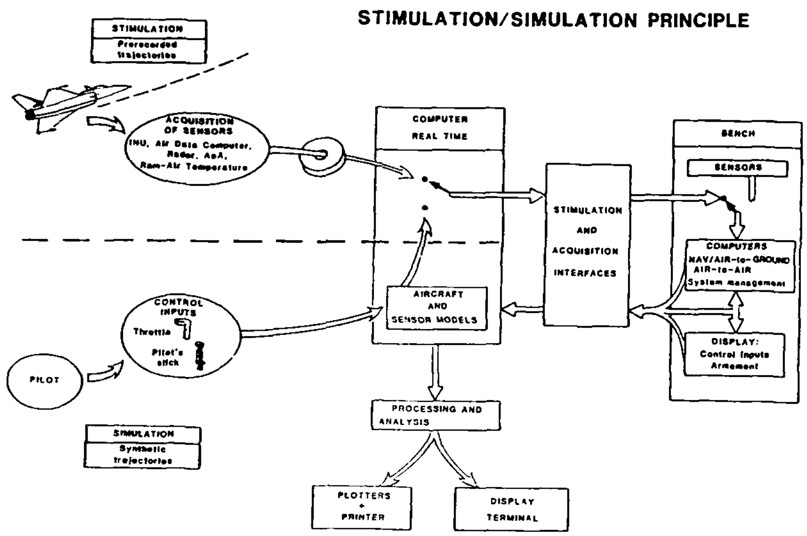
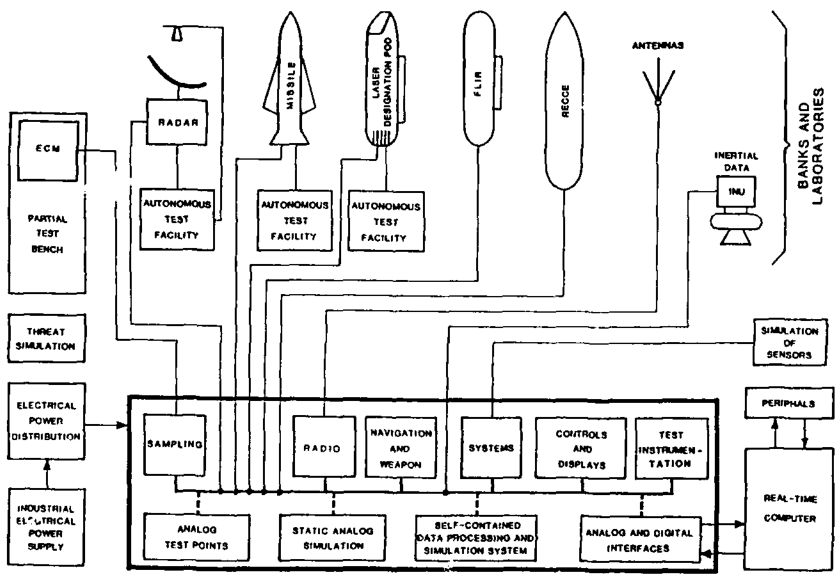


Fig. 3.3.4.



3.4 - Armament Tests

The general methodology for qualification of a new weapon on an aircraft is presented on Figure 3.4.1 The chronological order to respect is the following.

3.4.1 - Opening of the Flight Envelope

The methods used are identical to those specified for airframe testing :

- Monitoring of structural vibrations to prevent flutter and checking of gain margins of the flight control system channels,
- Handling qualities,
- Measurement of stress,
- Determination of armament drag.

3.4.2 - Separation and Firing Envelope (Figure 3.4.2)

The usual test procedure is applied :

- Ground release tests from a gantry or directly from the aircraft.
  These tests allow to select the ejectors that provide a satisfactory vertical speed and a correct pitch rate.

- The test results are checked in the wind tunnel within the desired envelope for release of a given external load.

- Flight tests.

The main purpose is to ensure safety by checking the trajectory of the external store with regard to the aircraft. This is achieved by means of a set of cameras filming the attitude of the released store in the vicinity of the aircraft (Figure 3.4.3).

To check the correct run of the firing sequence, it is necessary to make sure that the store leaves the area perturbed by the aircraft with attitudes that ensure its satisfactory flight from the point of view of accuracy, and that allow to conserve the automatic target lock-on, for guided weapons.

The films taken by the cameras mounted at different locations must enable to restitute the attitudes up to an aircraft-store distance of 30 to 50 meters. To determine the inertial trajectory of the store, it is mandatory to know precisely the setting of the cameras with respect to the aircraft and the aircraft movements after separation, that are to be synchronized with the photos.

In case of salvo firing, it is moreover necessary to check the programmed time intervals between each bomb release to ensure a correct grouping of hits.

3.4 - Essais armement

La méthodologie générale de qualification d'un nouvel armement sur un avion est présentée sur la planche 3.4.1. La chronologie qui est à respecter est la suivante :

3.4.1 - Ouverture du domaine de vol

Les méthodes employées sont celles mentionnées pour les essais cellule :

- Surveillance des vibrations de structure pour prévenir le flutter et pour contrôler les marges de gain des chaînes de commandes de vol,
- Qualités de vol,
- Mesures de contraintes,
- Détermination de la traînée des charges.

3.4.2 - Domaine de séparation et tir (planche 3.4.2)

On retrouve le processus d'essais habituel :

- Essais au sol sous un portique ou directement avec l'avion.
  Ces essais permettent de sélectionner des éjecteurs qui donnent une vitesse verticale satisfaisante et une vitesse de tangage correcte.

- Ces essais sont contrôlés en soufflerie dans le domaine de largage désiré.

- Essais en vol

Le but principal est d'assurer la sécurité en contrôlant la trajectoire de la charge par rapport à l'avion ceci est obtenu grâce à un jeu de caméras filmant l'attitude de la charge larguée au voisinage de l'avion (Planche 3.4.3).

Pour contrôler le bon fonctionnement de la séquence de tir, il faut s'assurer que la charge quitte le champ perturbé par l'avion avec des attitudes qui lui assurent un vol satisfaisant pour la précision et qui permettent de conserver l'accrochage automatique sur la cible pour les charges guidées.

Les films, pris par les différentes caméras, doivent permettre de restituer les attitudes jusqu'à une distance de 30 à 50 mètres. Pour accéder à la trajectoire inertielle de la charge, il faut connaître de façon précise le calage des caméras par rapport à l'avion et les mouvements avion après séparation, synchronisés avec les photos.

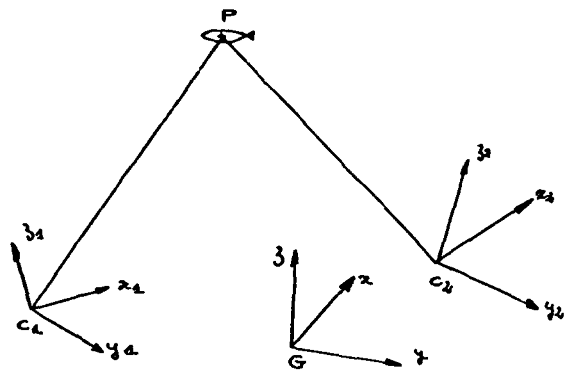
Dans le cas de tir en salve, il faut également contrôler les écarts de temps programmés entre les différentes bombes pour assurer un bon groupement.

3.4.3 - Processing Techniques for Stores Trajectory Plotting

3.4.3.1 - Présent Method

16-mm film cameras are accommodated on various aircraft stations or housed in modified pods.

The method used is the triangulation of two lines of sight.



<u>With</u> :

Gxyz	trihedral of reference linked to the aircraft
C1x1y1z1	trihedral of camera C1
C2x2y2z2	trihedral of camera C2

<u>It is possible to write</u> :

$$\vec{GP} = \vec{GC1} + \vec{C1P} = \vec{GC2} + \vec{C2P}$$

To determine the $\vec{GP}$ vector in the aircraft reference trihedral, it is necessary :

- to measure the position of each camera, C, with respect to this trihedral,
- to identify the rotation matrix of each camera to be able to define the direction of each vector GP in the reference trihedral,
- to know the intrinsic characteristics of each camera (focal distance, distortion of the objective, etc...)

All the abovementioned actions have to be completed prior to the flight to be able to solve the above-cited vectorial equation.

3.4.3 - Méthodes d'exploitation de la trajectographie des charges

3.4.3.1 - Méthode actuelle

On implante dans différentes parties de l'avion ou dans des réservoirs modifiés des caméras film 16 mm.

La méthode utilisée est la triangulation de 2 lignes de visée.

<u>Soit</u> :

Gxyz	le trièdre de référence lié à l'avion et
C1x1y1z1	le trièdre de la caméra C1
C2x2y2z2	le trièdre de la caméra C2

<u>On peut écrire</u> :

$$\vec{GP} = \vec{GC1} + \vec{C1P} = \vec{GC2} + \vec{C2P}$$

Pour déterminer le vecteur $\vec{GP}$ dans le trièdre de référence avion, il faut :

- Mesurer la position de chaque caméra C par rapport à ce trièdre,
- Identifier la matrice de rotation de chaque caméra pour permettre de définir la direction de chaque vecteur $\vec{GP}$ dans le trièdre de référence,
- Connaître les caractéristiques intrinsèques de chaque caméra (focale, distorsion de l'objectif, etc...).

Il est nécessaire que tous ces points soient traités avant le vol afin de pouvoir résoudre l'équation vectorielle citée.

As explained beforehand, the point P can be defined with respect to the reference trihedral on condition to be "seen" from at least two different viewpoints.

In order to determine the relative *trajectory of the* store put to test until about 30 meters from the original position (limit that is sufficient today), it is necessary to install about 5 pairs of cameras.

Determination of Attitudes

For a given image, it is possible to identify "n" points on the studied object ; if only three points are determined, the relative attitudes (phi, teta, psi) of the store with respect to the aircraft can be computed.

Computation is performed by means of a NAC analyser, coupled to our real-time acquisition computers.

3.4.3.2 - Evolution, Diminution *of Cost*

The first step to cut costs consists in optimizing the preparation of the general release program in order to reduce the number of flights to the minimum.

The second action consists in changing the methodology with the aim to decrease the number of cameras required.

For about *10* cameras :

. Boresighting, even automated, immobilizes the aircraft during half a day, with two technicians.

. The complete processing of films requires 8 hours of work and analysis between 2 flights.

AMD-BA developed a technique enabling to automate processing and to reduce the number of cameras.

Presently, this process is applied for photos from 16-mm films that are digitized.

The external load under test is extracted from the digitized images.

This store image is converted into a skeleton that extends to the extremities (angular points). (Confer to Figure 3.4.4). This linearized skeletonization enables to identify the characteristic axes (Figure 3.4.5) which, in turn, permit to determine the rotations (phi, teta, psi).

Nous avons vu que le point "P" pourra être défini par rapport au trièdre de référence que s'il est vu par au moins 2 postes d'observation différents.

Si l'on veut déterminer la trajectoire relative de la charge étudiée jusqu'à environ 30m par rapport à la position d'origine (limite aujourd'hui suffisante), il est nécessaire d'implanter sur l'avion environ 5 couples de caméras.

Détermination des attitudes

Pour une image donnée, il est possible d'identifier "n" points sur l'objet étudié, si 3 points seulement sont déterminés, on peut calculer les attitudes relatives (Phi, Téta, Psi) de la charge par rapport à l'avion.

L'exploitation s'effectue par un analyseur NAC couplé à nos ordinateurs d'acquisition temps réel.

3.4.3.2 - Evolutions, diminution des coûts

Une première action de diminution des coûts consiste à bien préparer le programme général de largage en vue de *faire* le *nombre* de vols minimum.

Une deuxième action consiste à changer la méthodologie en vue de diminuer le nombre de caméras nécessaires.

Avec une dizaine de caméras :

. L'harmonisation même automatisée immobilise l'avion pendant une demi-journée avec 2 opérateurs.

. L'exploitation complète des films demande environ 8 heures de travail et d'analyse entre 2 vols.

Nous avons développé une méthode permettant d'automatiser l'exploitation et de diminuer le nombre de caméras.

Elle est actuellement développée à partir des photos film 16 mm qui sont numérisées.

Des images numérisées, on extrait la charge elle-même. Cette image charge est transformée en un squelette qui joint les points terminaux (points anguleux) (planche 3.4.4).
Ce squelette linéarisé permet d'identifier les axes caractéristiques (planche 3.4.5) à partir desquels on détermine les rotations (Phi, Téta, PSI).

Using our tridimensional, interactive CAD - CAM design tool CATIA, one image of the store, in a given attitude, is synthetized and by means of the skeletonization technique, the analyst tries to restore this attitude. The results obtained are good : 0.4° in roll, 1° in pitch, 0.2° in yaw.

The aircraft-store distance is also determined on the basis of synthetic images representing the store at two different distances but in the same attitude (See Figure 3.4.6).

. Currently, this technique allows :

A gain of 50% on the number of observation points, consequently cost-saving in terms of equipment.

Based on an average rate of 300 flights per year, the gain in personnel is estimated at 600 manhours of preparation and 1500 manhours of processing by graduate engineers.

The time spent between two flights can be shortened from 8 to 2 hours, thus enabling to carry out 2 flights per day.

A further evolution towards real-time video transmission could lead to an additional reduction of the number of flights, enabling to carry out a second release after having performed a safety check on the first test.

3.4.4 - Weapon-System Integration

This is a complement to the general integration tests we described earlier.

Firing accuracy checks are performed, consisting in a verification of the various control orders with updating, based on bench tests.

Accuracy tests are also made with small, well calibrated practice bombs whose trajectory calculation was entered into the fire control computer.

3.5 - Countermeasures Tests

The AMD-BA Flight Test Center at ISTRES accommodates two vast anechoic chambers, without equal in EUROPE :

. Depth : 28 meters
. Width : 20 "
. Height : 13 "
. Volume : 7280 m³

Complete aircraft in flight-readiness condition, except for the engine which is simulated, are suspended in the center of the chamber. It is thus possible :

A une attitude donnée, l'on synthétise une image de la charge (à partir de programmes de dessin 3D CATIA) et l'on essaie de retrouver cette attitude par la méthode de squelettisation. On obtient ainsi de bons résultats : 0,4° en roulis, 1° en tangage, 0,2° en lacet.

La distance de la charge est également obtenue par des images de synthèse à deux distances différentes mais dans la même attitude (planche 3.4.6).

. Actuellement cette méthode permet : Un gain de 50% sur le nombre de points d'observation d'où gain en matériel.

Compte tenu d'une cadence moyenne de 300 vols par an, le gain en personnel est estimé à 600 heures de préparation et 1500 heures d'ingénieur à l'exploitation.

Le délai de 8 heures entre 2 vols peut être ramené à 2 heures ce qui permettra d'effectuer 2 vols par jour.

Une évolution vers la transmission vidéo temps réel pourrait permettre de diminuer le nombre de vols en faisant un deuxième largage après contrôle de sécurité sur le premier essai.

3.4.4 - Intégration système d'armes

Il s'agit d'un complément aux essais généraux d'intégration que nous avons décrits.

Des contrôles de précision de tir sont effectués en contrôlant les différents ordres de commande avec recalage sur les essais effectués au banc.

Des essais de précision sont également effectués avec des petites bombes d'exercice bien calibrées dont la trajectographie a été entrée dans le calculateur de tir.

3.5 - Essais contre-mesures

Le centre d'essais AMD-BA d'ISTRES est équipé de deux chambres anéchoïdes les plus grandes d'EUROPE :

. profondeur : 28 mètres
. largeur : 20 "
. hauteur : 13 "
. volume : 7280 m³

Des avions complets en état de vol, sauf le moteur qui est simulé, sont pendus au centre de la chambre. On peut ainsi :

Fig. 3.4.1.

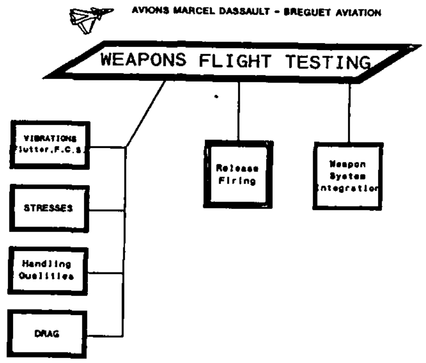


Fig. 3.4.2.

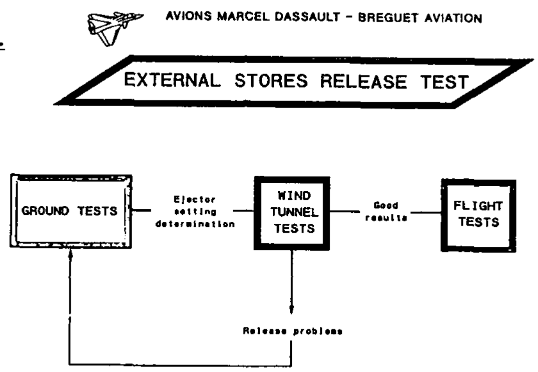


Fig. 3.4.3.

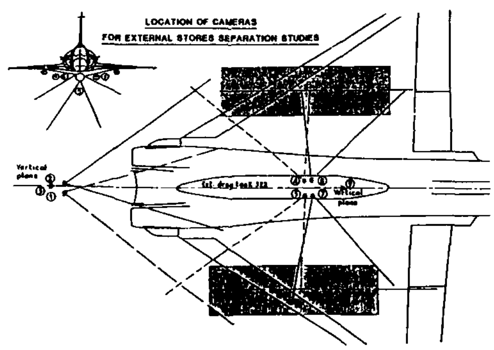
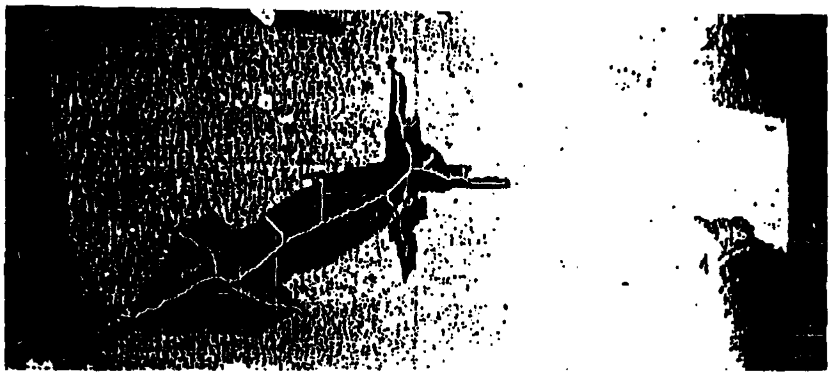
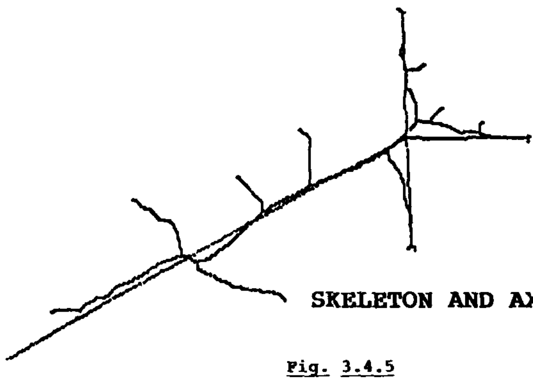


Fig. 3.4.4.





SKELETON AND AXES DETERMINATION

Fig. 3.4.5

DISTANCE DETERMINATION
WITH TWO SYNTHETIC IMAGES
AT TWO DISTANCES -
SAME ATTITUDE

Fig. 3.4.6.

. to check without external interference (anechoic chamber and Faraday cage) the susceptibility of equipment to electro-magnetic radiations. These are safety tests in dense or protected electro-magnetical environment,

. to check discoupling of antennas,

. to test the efficiency of the countermeasures equipment.

The purpose of these tests is to optimize the relative operation of the ECM equipment and that of the radar and the missiles. They also enable a judicious selection of limit test points that will be checked in flight. One month of testing in an anechoic chamber allows to optimize the profile of 3 or 4 validation flights.

. contrôler sans parasitages extérieurs (chambre anéchoïde et cage de Faraday) la susceptibilité des équipements aux rayonnements électromagnétiques. Ce sont des essais de sécurité en ambiance électromagnétique dense ou pas.

. contrôler le découplage des antennes,

. vérifier l'efficacité des équipements de contre-mesures.

Le but est d'optimiser le fonctionnement rélatif de ces équipements et ceux du radar et des missiles. Ces essais permettent également de bien sélectionner des points limites qui seront à contrôler en vol. Un mois d'essais en chambre anéchoïde permet de sélectionner 3 ou 4 vols de validation.

4 - CONCLUSIONS

We are in a position to confide a non exhaustive assortment of recipes to increase the efficiency of flight testing with the purpose to reduce the cost of development of an aircraft or airborne systems.

Certain advices seemingly stand to reason, but they imply an inside experience that cannot be learned in a single program. It is also necessary to convince the various participants and, in particular, the equipment manufacturers.

First of all, to reduce the number of test flights, the product put to test must have been defined with care and must have completed all feasible tests on the ground.

Flights should only be qualification tests as for missiles ; the difference lies in the man-machine match and in the operational use that is much more diversified than that of a mono-operation missile.

The means available are :

. theoretical aerodynamic and structural calculations on powerful computers,

. wind-tunnel tests whose increase was possible thanks to :

  . computer-assisted design and manufacturing, thus cutting down prices and mock-up production lead times,

  . automation of wind-tunnel acquisitions, thereby decreasing the time needed to yield results,

. tools assisting in the specification of airborne systems (Avions MARCEL DASSAULT - BREGUET AVIATION developed OASIS).

. systems test benches coupled to computers :

  . Flight controls rig with simulator,

  . Stimulable benches for navigation and armament systems. These test benches can take part in the qualifications and certifications of systems, more particularly for failure investigations.

4 - CONCLUSION

Nous pouvons donner quelques recettes non exhaustives permettant d'augmenter l'efficacité des essais en vol, le but étant de réduire le coût de la mise au point d'un avion ou des systèmes embarqués.

Certains points peuvent paraître des évidences mais ils demandent une expérience qui ne peut être acquise sur un programme. Il faut également convaincre les différents participants et en particulier les équipementiers.

En premier lieu, pour diminuer les vols d'essais, il faut avoir un produit qui a été bien défini et qui a subi tous les essais nécessaires au sol.

Les vols ne devraient être que des essais de qualification comme pour des engins ; la différence portant sur le couplage homme-machine et sur l'utilisation opérationnelle qui est plus diverse que dans le cas d'un engin mono-opération.

Les moyens à utiliser sont :

. des calculs théoriques aérodynamique et structure sur des ordinateurs puissants,

. des essais en soufflerie qui ont pu être augmentés grâce à :

  . la conception et fabrication assistées par ordinateur qui diminuent le prix et le délai de fabrication des maquettes.

  . l'automatisation des acquisitions des souffleries qui diminue les délais d'obtention des résultats.

. des moyens d'aide à la spécification des systèmes embarqués (OASIS aux Avions Marcel DASSAULT BREGUET- AVIATION).

. des bancs systèmes couplés à des ordinateurs.

  . bancs commandes de vol avec simulateur,
  . bancs stimulables de système de navigation et d'armement.

Ces bancs peuvent participer à des qualifications et certifications des systèmes en particulier pour l'étude des pannes.

- The Test Programs must be prepared very conscientiously. It is at this stage when development requirements are analyzed and when thorough reflection must be given to the part of testing that can be made on the ground and in laboratories (less costly) and to what remains to be covered in flight.
- This analysis, associated to the definition of the processing means, enables to deduce the required measurement installations. This approach must always be made in this order. Measurement installation have to be well conceived to reduce subsequent processing work. Software costs may be considerable.

The problem of data storage must be given careful thought.

A priori, it is better to refuse recordings, they may not be indispensable ... Experience proves that not validated and not checked recordings are very costly in terms of invested equipment, and they always fail to work when they are likely to be needed.

Precedence is to be given to flight safety problems by providing the ground test team with the means to cope quickly with malfunctions.

For complex analyses, it may become necessary to resort to artificial intelligence.

Developers must constantly be on the watch out for possible problems of standardization of data acquisitions and transmissions.

AMD-BA have succeeded in containing the data-flow requests. For an experimental aircraft such as RAFALE, the development tasks were tackled with 1300 wired measurements, 1000 of them recorded at a data rate of

- 8000 measurements per second, processed in real time throughout the flight,
- 45 accelerometers processed on the ground with an output of 200 points/second for scrupulously selected test points and the possibility to treat, simultaneously, flight envelope, handling qualities and stress problems.

Weapon systems can be stimulated on the ground, using play-backs offering the same information stream of 8000 measurements per second for several hundred parameters.

- Les programmes d'essais doivent être très bien préparés. C'est à ce stade que seront analysés les moyens nécessaires à cette mise au point et en particulier ce qui peut être fait au sol et en laboratoire, moins coûteux et ce qui reste à contrôler en vol.
- Cette analyse permet, en définissant les moyens d'exploitation d'en déduire les installations de mesure nécessaires. Cette approche doit toujours être réalisée dans cet ordre. L'installation de mesure doit être conçue pour diminuer les travaux ultérieurs d'exploitation. Les coûts de logiciels peuvent être très importants.

Il faut songer aux problèmes de stockage des données.

Refuser des enregistrements a-priori, des fois que ... L'expérience prouve que ces enregistrements non validés et non contrôlés sont très lourds en matériel et ne marchent jamais lorsqu'on pourrait en avoir besoin.

Il faut toujours s'intéresser en priorité aux problèmes de sécurité du vol en fournissant à l'équipe d'essais au sol des moyens permettant de répondre rapidement à ces anomalies.

Pour des analyses compliquées, il peut être nécessaire d'avoir recours à de l'intelligence artificielle.

Faire attention aux problèmes de standardisation des acquisitions et de transmission de données.

Aux AMD-BA, nous avons réussi à contenir les demandes de débit de données. Pour un avion expérimental du type RAFALE, la mise au point a été réalisée avec 1300 mesures câblées dont 1.000 enregistrées avec un débit de 8.000 mesures par seconde traitées en temps réel pendant tout le vol. 45 accéléromètres traités au sol au débit de 200 points par seconde pendant des points d'essais bien sélectionnés et permettant de traiter simultanément les problèmes de domaine, qualité de vol et efforts.

Les systèmes d'armes peuvent être stimulés au sol en réalisant des play-back avec le même débit d'information de 8.000 mesures par seconde sur quelques centaines de paramètres.

GRATE - A NEW FLIGHT TEST TOOL FOR FLYING QUALITIES EVALUATIONS

by

R. Koehler
Institut für Flugmechanik
Deutsche Forschungs- und Versuchsanstalt
für Luft- und Raumfahrt e.V. (DFVLR)
D 3300 Braunschweig-Flughafen, West Germany

E. Buchacker
Federal Armed Forces Engineering Center for Aircraft
D 8072 Manching, West Germany

D.J. Biezad
School of Engineering
Air Force Institute of Technology
Wright-Patterson Air Force Base
Dayton, Ohio, USA

1. INTRODUCTION

A flight test tool has been worked out by DFVLR for flying quality evaluations of ground attack tracking phases (Ref. 1). This Ground Attack Technique (GRATE) has been proven in test flights of the German Federal Armed Forces Engineering Center for Aircraft and has been integrated in the Large Amplitude Multimode Aerospace Research Simulator (LAMARS) of Air Force Wright Aeronautical Laboratories (AFWAL) in Dayton. The German/US cooperation has been supported by the Memorandum of Understanding (MoU) "Flight Control Concepts".

In this paper a description of the flight test method and of the pilot's role and ratings is given. Head-up-display films have been evaluated to determine a so called align-time and a circular error probability (CEP). The influence of different test conditions on the mission parameters has been investigated. The results of the numerical analysis and the pilot ratings have been compared. The determined gradients show the sensitivity of a pilot rating to the mission parameters. In this context a configuration with slight PIO-tendencies is discussed.

Simulator tests have shown that the technique is an effective tool for unmasking aircraft handling problems. The effects caused by different turbulence levels on pilot ratings were found to be small in comparison to conventional methods.

2. BASIC IDEA

Dynamic handling qualities of an aircraft can be tested by a pilot with application of a three-step technique as follows:

- The pilot determines the response characteristics of the aircraft by moving the controls and watching the aircraft response.
- He compares the dynamic behaviour of the aircraft with other airplanes.
- He assesses the differences with regard to the importance of the characteristics for a mission or flight phase.

The GRATE was developed and tested in consideration of the three steps mentioned. In this technique light targets are placed at different positions on the ground (Fig. 1). During a prolonged dive the lights are switched by an input signal to generate aiming errors in the sights of the head-up display (HUD). Thus the pilot is forced to react continuously, and the closed-loop pilot-aircraft system is excited over a wide frequency range of interest. The pilot gets more information about mission-oriented flying qualities, and suitable data to calculate mission parameters can be generated.

3. TECHNICAL ARRANGEMENTS OF FLIGHT AND SIMULATION TESTS

A flight test program was performed utilizing an experimental aircraft based on the alpha-jet configuration with some modes of changeable flying qualities.

The setup of the test equipment consists of an airborne and a ground system (Fig. 2). The main part of the ground system is the collection of light targets.

The overall arrangement of the ground system (all lamps switched on) is shown in Fig. 3 from the point of view of a pilot during a simulated attack.

Each target is a lamp cross with eight halogen lamps switched on and off by a microprocessor according to the signal received via cables connected to the telemetry ground station (Fig. 4).

The LAMARS-simulator consists of a five-degree-of-freedom beam-type motion system which carries a single-seat cockpit and a display screen on the end of a beam (Fig. 5).

The cockpit was representative of a single-seat, high-performance aircraft with a centre stick controller and a HUD.

On the spherical screen is displayed a three-dimensional terrain model (Fig. 6). The pictures are generated using a gantry-supported optical-probe-equipped television camera. A hybrid computer system is connected to the motion base, the cockpit controls, the HUD and the terrain board servo system.

There are two ways to represent the target configuration in the simulator. Lights can be displayed in the HUD of the cockpit, or small lamps can be mounted on the terrain board. Lights in the HUD can easily be modified by the computer program whereas the lamps on the terrain board yield a more realistic image.

4 TEST PROGRAM PREPARATIONS

The target illumination sequence on the ground produces a visual input signal to the pilot-aircraft system.

The criteria to select suitable target arrangements and light switching signals were derived from experience in system identification (chapter 4.1 and 4.2).

In order to investigate system responses to small perturbations, the targets were placed within a narrow area on the ground. The signals were selected with respect to a maximum target leap of 1.0 degree in the sight of the cockpit.

The calculations were performed for a reference flight which had the following characteristics:

- a straight-line glide path to the centre of the target area
- flight path angle of -10 degrees
- initial airspeed of 400 kn
- constant acceleration due to the gravity component in the flight direction
- distance range from 3300 m to 800 m to the centre of the target area.

4.1 TARGET CONFIGURATIONS

The distribution of the targets on the ground is designed in the MIL-plane which is perpendicular to the line of sight from the pilot to the centre of the target area (Fig. 7). Different design concepts result in different arrangements; see target configurations A and B in Figure 7.

Configuration A was designed to yield a symmetric picture in both the vertical and lateral directions. It has the same structure and spread with respect to both axes. Thus a random signal would create similar aiming errors in the vertical and lateral direction.

The targets in configuration A are also arranged favourably with respect to the following aspect. The input signal z in Fig. 8 which is used to switch the lamps contains the components z_x and z_y. The extremes in one component of the signal appear when the other signal component vanishes. Thus the numerator of the correlation coefficient becomes small compared to the denominator and the signals are nearly uncorrelated. The pilot will excite the longitudinal and lateral motion independently and he gets more information about the aircraft characteristics (Ref. 2).

The target configuration B shown in Fig. 7 was selected from several arrangements to satisfy the following criterion:

To perform many input signal steps a long dive is desirable. The maximum and minimum acceptable tracking distances were determined in flight. They have a ratio of 4 to 1. Hence the visual angles between targets are of the same ratio. The targets should be positioned to obtain the desired step size at any location within the test range. Therefore the ratio of the largest to smallest distance between the targets should be 4 to 1 and the other distances should have a uniform distribution within these limits (see Fig. 9). Such characteristics were achieved using only eight targets.

To implement configurations A and B during a test, the arrangement of targets must be transferred from the MIL-plane to the ground plane. The final plans of the configurations are shown in Fig. 10 and 11.

For some flights the targets were arranged along a line in both the longitudinal and cross directions. These arrangements emphasized the motions with respect to tl selected axes.

4.2 INPUT SIGNAL DESIGN

In order to evaluate or identify aircraft handling qualities from closed-loop tests, sufficient excitation or disturbance must exist over a reasonably wide bandwidth (Ref. 3). This is especially important during precision flight control phases, where flying qualities "cliffs" may exist which seriously degrade performance at critical times during the flight (Ref. 4). The input signals of GRATE are generated by the motions of the line of sight between the pilot and the targets when the lights are switched. They are designed in the frequency domain (Ref. 5).

The line of sight angle is split into the vertical (ϵ_x) and lateral (ϵ_y) components. The dependence of the angles on the distance between the aircraft and the centre of the target area (x_F) is shown in Fig. 12. The curves for the lamps which are not illuminated are also shown in the diagrams.

The time histories of the line of sight angles (ϵ_x and ϵ_y) are input signals to the pilot/aircraft system. They can approximately be represented by a multi-step function r(t) (Fig. 13).

The equation describing the power spectrum $|R(\omega)|^2/T$ of a signal with constant time intervals Δt has two factors.

The first factor $2\ \Delta t\ (1 - \cos \Omega)/\Omega^2$, where $\Omega = \omega\Delta t$, is a function of the interval duration Δt and the frequency ω, and is not affected by the switching amplitudes v_i. The amplitude is obtained from the power spectrum by taking the square root and results in a clearer picture for the first factor (Fig. 13). This term is shown for a range of Δt intervals which were found acceptable by pilots in flight tests.

The peaks in the functions shown steeply decrease with increasing frequency Ω. Since the second factor is periodic with $\Omega = 2\pi$, the decrease in the amplitude spectrum at higher frequencies cannot be prevented even by a special selection of the amplitudes v_i. These characteristics of the spectrum prevent the generation of signals with an approximately constant spectrum. Nevertheless, the best possibilities within the limits discussed were utilized.

A computer program was used to generate sets of input signals with uniform power spectra for the line of sight angles ϵ_x and ϵ_y.

5. PILOTS' ROLE

The basic idea and the test layout were mainly based on theory. Flight and simulation tests were necessary to adapt the method to the demands of practical operations. Hence the test pilots fulfilled various tasks:

- they tracked the targets for data collection;
- they provided information about different aspects of the test method; and
- they assessed the flying qualities of the aircraft using the well-known Cooper-Harper rating scale.

The pilots rated the technique to be well suited for evaluating air-to-ground handling qualities. Results showed that GRATE is effective and easy to both learn and use. It is as effective as turbulence in unmasking poor flying qualities. As turbulence is not available on call, poor flying qualities may remain masked even in an otherwise rigorous conventional test program.

6. EVALUATION OF FLIGHT TEST DATA

The evaluation of the flight test data has been concentrated on the investigation of tracking performance parameters.

Flight test data were measured from HUD camera film including position of the pipper and the illuminated lamp. In the time histories of the example shown in Fig. 14, the steps in pitch and azimuth of the target light and the changes initiated by the pilot in order to track the target are clearly visible.

The star-like pattern in the cross plot of the aiming error indicates four loops which correspond to the four steps of the light signal. The time histories of these four sequences can be treated as four isolated characteristic motions with different initial conditions of the pilot-aircraft system. When the light jumped in the negative direction of pitch or yaw, the time histories were multiplied by -1 to maintain a characteristic motion with a positive initial condition.

The mean values calculated from the four characteristic motions and curves of limits of confidence are shown in Fig. 15. Thus the influence of noise on the time histories can be reduced.

6.1 DEFINITION OF MISSION PARAMETERS

A mean radial deviation ρ_r was calculated eliminating estimated disturbance effects. Details are given in Appendix A; time histories are shown in Fig. 16.

The time up to the moment when the mean radial deviation passed the value of 3 mrads was determined and increased by 10 %. This result was defined to be the align-time.

After the align-time, tracking is approximately a stationary random process. The test data of the stationary tracking can be evaluated by determining two circles which surround tracking data from 50 % of the time after the align-time (Fig. 17). Their radii are called circular error probability around the mean aiming point, CEP_{MAP}, and circular error probability around the target, CEP_{TGT}.

The radius CEP_{MAP} is smaller and more sensitive to disturbances during the aiming process and is therefore more suitable for handling quality evaluations. Note the CEP_{MAP} in this report is not affected by the recoil of a gun and is averaged utilizing the data of all stationary tracking time.

6.2 SEPARATION OF DEPENDENCIES

The mission parameters depend on different test conditions, e.g. turbulence, target configuration, and feedback mode. The evaluated parameters which are valid for the same test conditions with the exception of one variable are drawn in a diagram. Thus the dependencies can be ascertained.

The align-time and CEP_{MAP} are shown in Fig. 18 for different turbulence levels which were determined from pilot comments. Each point in the diagram is an evaluation result of an attack dive.

The align-time may vary if manoeuvrability characteristics are changed, but obviously it does not depend on the turbulence.

The CEP_{MAP} remains constant in low turbulence levels but increases when the turbulence becomes moderate to heavy.

In a similar way the influence of the target step direction on the align-time was investigated. Averages of align-times for all targets placed in the longitudinal direction was slightly shorter than for targets placed crossways or for the two-dimensional target configuration (A).

For comparison of different feedback modes, align-times were taken with target configuration A and targets placed crossways. Quantities of CEP_{MAP} were taken with turbulence levels of none to light to moderate (chapter 6.4).

6.3 PIO TENDENCIES

A feedback mode was available which made the experimental aircraft lightly susceptible to PIO tendencies of the longitudinal motion. The oscillations appeared for any target configuration.

When the targets were placed crossways, disturbances of the longitudinal motion were small but the oscillations in pitch were clearly visible (Fig. 19).

When the mean radii ρ_r were calculated by averaging, the oscillations, as shown in Fig. 20 became reduced, the decrease of the radius became unmistakable and the determination of the align-time became more reliable.

6.4 COMPARISON OF PILOT RATING AND MISSION PARAMETERS

During flight tests separate ratings for roll, pitch, yaw, and normal acceleration of precision tracking and manoeuvrability were accomplished.

Averages of pilot ratings during precision tracking are represented in Fig. 21 (upper half). Mean values of CEP_{MAP} and their standard deviations are shown beneath them. All diagrams have a similar shape. Therefore pilot ratings and CEP_{MAP} values are correlated. An average of the presented ratings was calculated and drawn against the mission parameter CEP_{MAP} in Fig. 22.

The values of the feedback modes of B, C, and D yield a straight line. Its slope is a sensitivity of pilot rating.

If the CEP_{MAP} increases by 1 mil - that is an error deviation of 3 feet in a distance of 3000 feet - than the pilot rating will increase by 0.5. These values may also represent the resolution of the pilot.

The PIO tendencies of the feedback mode A resulted in a high workload during precision tracking. Therefore the pilot rating increased disproportionately and the dot appears over the line drawn in Fig. 22.

Averages of pilot ratings of manoeuvrability for roll, pitch, yaw and vertical acceleration are shown in Fig. 23 (upper half). Mean values of the align-time and their standard deviation are shown beneath them. Pilot ratings and align-times of the feedback mode A and B are slightly smaller than of C and D.

The mean value of the presented ratings was calculated and drawn against the mission parameter of align-time in Fig. 24. The slope of the straight line is again a sensitivity of the pilot rating.

If the align-time increases by 0.2 sec the pilot rating will increase by 0.5. These values may also represent the resolution of the pilot. Note feedback mode A with PIO tendencies yielded the smallest pilot rating of manoeuvrability and has a good aligntime.

7. SIMULATOR TEST PROGRAM

The purpose of the simulation was to compare the advantages of the target configurations A and B and to evaluate the GRATE by varying flight configurations with known handling qualities in a precise and repeatable manner. Different levels of time delay and turbulence intensity were investigated relative to the baseline configurations of a ground attack and a fighter aircraft.

The transfer functions for the low-order simulations of the baseline aircraft are provided in Appendix B. They were implemented with simple, linear equations of motion in the LAMARS.

7.1 MODIFICATIONS OF SYSTEM AND TEST CHARACTERISTICS

Three time delays (TAU1, TAU2, TAU3) were investigated relative to the baseline configurations by adding a first-order Pade approximation to the model transfer functions in the longitudinal and the lateral/directional axes. Additionally, each time delay was flown in three different levels of turbulence (ATM1, ATM2, ATM3). The basic test matrix is shown in Table 1.

In addition to the basic tests, approaches were flown with and without turbulence with all the lamps on to suppress the GRATE. At approximately 250 meters AGL, the pilot switched from aiming at the closest lamp to the farthest lamp to simulate an offset manoeuvre. In this way, it could be determined whether the offset manoeuvre was as effective as the GRATE in unmasking poor handling qualities. Due to time constraints, only a small portion of the test matrix could be flown for these investigations.

7.2 RESULTS AND EVALUATION

Dives were simulated with target configurations A and B displayed in the HUD. Pilots liked the target configuration B better than A. Approximately constant step sizes within a dive was the advantage of this arrangement. So lamps were mounted in a pattern of configuration B on the terrain board for more realistic simulations.

Results for the baseline ground attack and fighter aircraft with no added time delay are shown in Table 2. Each pilot rating applies to a set of four data runs and includes an assessment of alignment and final tracking in the pitch and yaw axes.

The application of turbulence to the GRATE did not significantly affect the flying qualities evaluation. The turbulence affected performance slightly, but making the small changes necessary to track the target lamps was relatively easy with or without turbulence. These results coincide with those of the flight tests for low to moderate turbulence.

Table 3 summarizes the results of varying time delay and turbulence levels for the ground attack configuration. Predictably, the effect of increasing time delay was to induce closed-loop oscillations in both pitch and yaw axes. By reading the ratings horizontally, it is concluded that the GRATE for no turbulence is as effective as the application of turbulence in identifying degradations in handling qualities caused by increasing time delays. There is no tendency for increasing turbulence intensity to make ratings worse, at least in terms of the Level rating. The pilot, therefore, apparently rated a qualitative degradation in flying qualities due to time delay which was not significantly affected by turbulence.

Results for the fighter configuration using the GRATE were similar to those for the ground attack configuration and are shown in Table 4. Except for the pilot rating of 6 for the large time delay (TAU3), once again there are no variations in the ratings which depend on turbulence.

To check that the GRATE was truly effective in unmasking poor handling qualities, GRATE was suppressed by flying with all the lamps on. An offset was simulated by transitioning from the closest to the farthest lamp at about 250 meters AGL. The results are shown in Table 5. Note that with no turbulence, the offset simulation failed to unmask the degradation in rating due to TAU1. For TAU2, the offset resulted in a degradation in flying qualities but not to the same extent as with the GRATE. The optimistic ratings for the offset are due to the pilot compensating for time delay and/or turbulence by flying very smoothly, an option which is not available with the high-bandwidth input excitation of the GRATE. It is apparent from the last row of Table 5 that the offset manoeuvre, even with turbulence, did not consistently unmask flying qualities deficiencies.

8. CONCLUSIONS

The Ground Attack Technique (GRATE) for evaluating closed-loop handling qualities was tested in a flight test and a simulation program. The **qualitative** assessment of this test method is summarized as follows:

- Pilots agree that this technique effectively portrays the flying qualities in a ground attack dive under realistic conditions.
- Due to its reliance on visual cues providing a high bandwidth input excitation to the pilot-aircraft system, GRATE is more thorough than turbulence inputs in unmasking poor flying qualities.

From the quantitative evaluation of aiming errors it can be concluded that the circular error probability around the centre of the mean aiming point (CEP_{MAP}) and the align-time are correlated with mean pilot ratings of precision tracking and manoeuvrability. The comparison yielded the following sensitivities:

- A change of 1 mil of CEP_{MAP} resulted in a change of 0.5 in the rating of precision tracking.
- A change of 0.2 sec in the align-time resulted in a change of 0.5 in the rating of manoeuvrability.

In the future, GRATE will be used in critera investigations and flying-quality assessments. Additionally, a modified version called ATLAS (Adaptable Target Lighting Array System) is being developed by NASA Ames-Dryden and will be used in a NASA/AFTPS/DFVLR cooperative program under the umbrella of the USAF/FRG MoU.

9. *REFERENCES*

1. Koehler, R., and E. Buchacker, "A Flight Test Method for Pilot/Aircraft Analysis", 21st Annual Conference on Manual Control, Ohio State University, Jun 1985.

2. Biezad, D.J., and D.K. Schmidt, "Normalizes Predictive Deconvolution: A Time Series Algorithm for Modeling Human Operator Dynamics", AIAA Journal of Guidance, Control, and Dynamics, Vol. 8, No. 6, Nov-Dec 1985, pp. 768-776.

3. Mc Ruer, D.T., "Human Dynamics in Man-Machine Systems", Automatica, Vol. 16, May 1980, pp. 237-253.

4. Berry, D.T., B.G. Powers, K.J. Szalai, and R.J. Wilson, "In-Flight Evaluation of Control System Pure Time Delay", Journal of Aircraft, Vol. 19, No. 4, Apr 1982, pp. 318.

5. Koehler, R., "Design and Implementation of Input Signals for Identification of Pilot/Aircraft Models", DFVLR-FB 84-08 (1984), ESA-TT-880 (1984).

Time Delay (τ/δ_c)	Turbulence Intensity (fps rms) ATM0 (none)	ATM1 (light-1.5)	ATM2 (mod.-3.0)	ATM3 (heavy-5.0)
TAU0 (0.0 sec)	X	X	X	X*
TAU1 (0.1 sec)	X	X	X	-
TAU2 (0.2 sec)	X	X	X	-
TAU3 (0.3 sec)	X	X	X	-

* ATM3 unrealistically severe

Table 1 Test Matrix

Configuration	ATM0	ATM2	ATM3
Ground Attack Conf.	3	3	-
Fighter	2	3	4*

* "literally driving task with turbulence too high for flight test

Table 2 Turbulence Influence on Baseline Pilot Ratings No Time Delay - TAU0

Time Delay	ATM0	ATM1	ATM2
TAU0	3	3	3
TAU1	3/3*	3	-
TAU2	4/6	4	5
TAU3	7/9**	8**	8**

* "slight pitch bobble for tracking"
** "could not perform task within reasonable tolerance"

Table 3 Turbulence and Time Delay Influence on the Ground Attack Configuration Pilot Ratings

Time Delay	ATM0	ATM1	ATM2
TAU0	2	2	3
TAU1	5	5	5
TAU2	7	7	7
TAU3	8	6*	8

* "not really able to do the task, but a little better than with higher turbulence level [ATM2]"

Table 4 Turbulence and Time Delay Influence on the Fighter Configuration Pilot Ratings

Time Delay	GRATE Technique ATM0	GRATE Technique ATM1	Offset Manoeuvre ATM0	Offset Manoeuvre ATM1
TAU1	5	5	2	5
TAU2	7	7	4	3

Note: Offset ratings based on two data runs

Table 5 Fighter Configuration Pilot Ratings Comparison

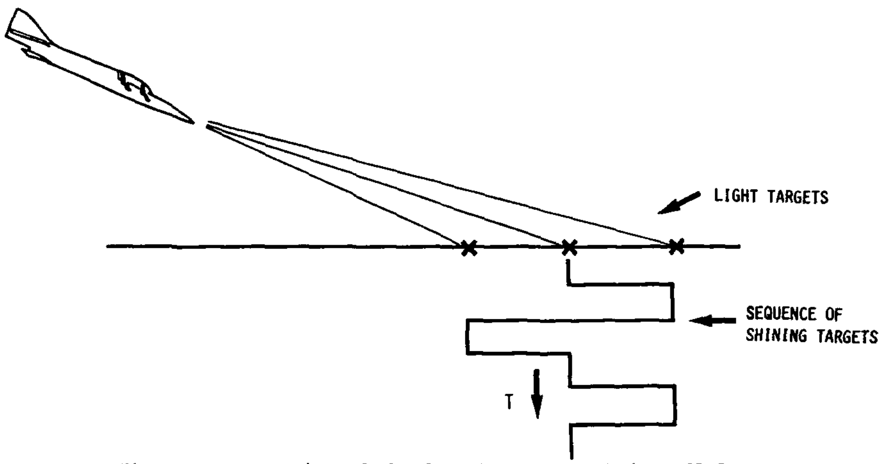


Fig. 1 Illustration of the Ground Attack Technique GRATE

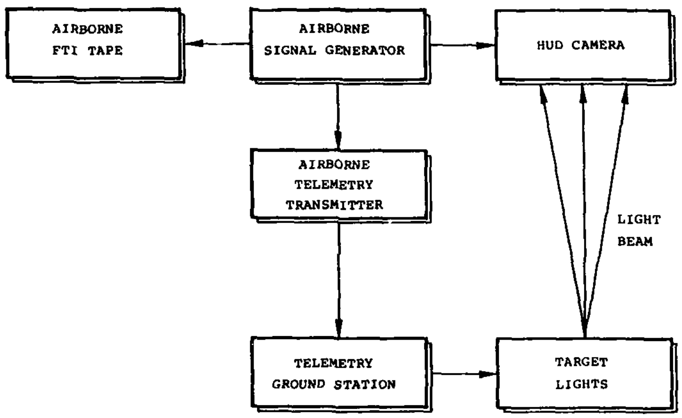


Fig. 2 Test Setup

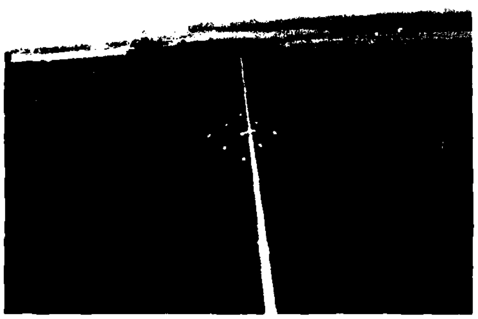


Fig. 3 Arrangement of Target Lights on the Ground

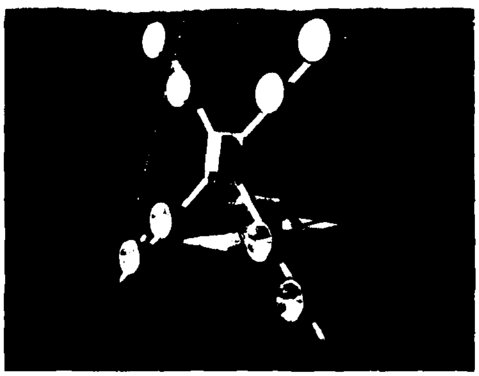


Fig. 4 A Lamp Cross

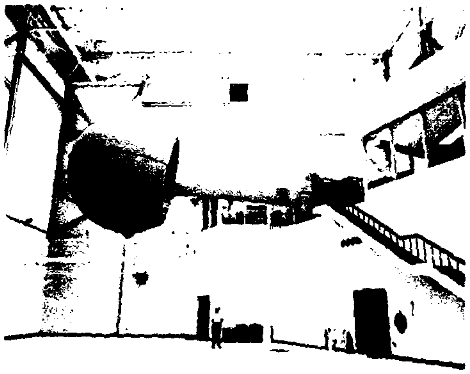


Fig. 5 Motion System of LAMARS

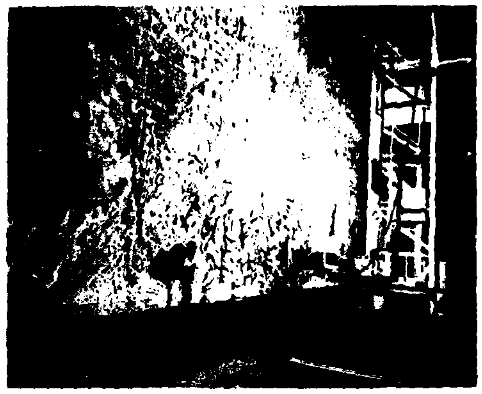
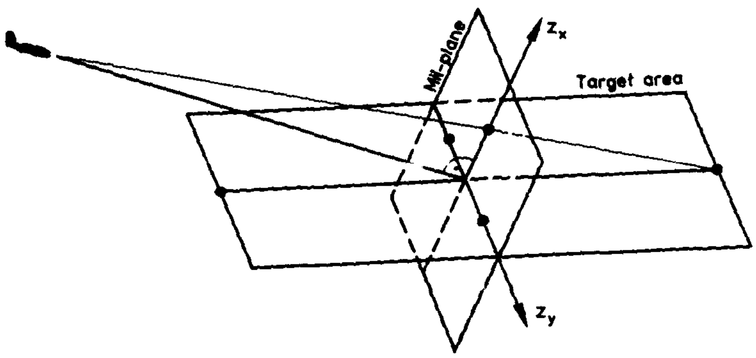


Fig. 6 Terrain Board of LAMARS



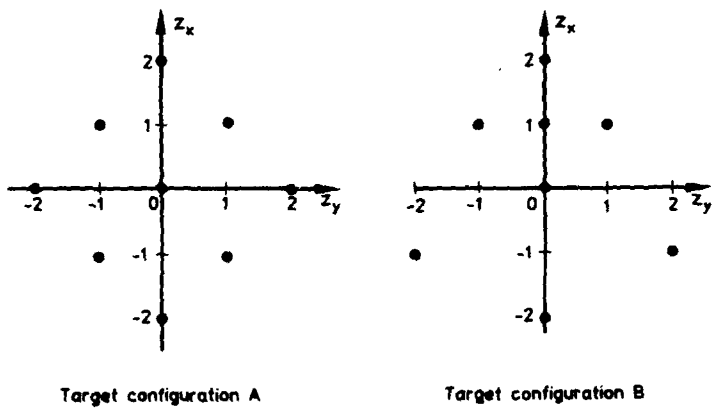
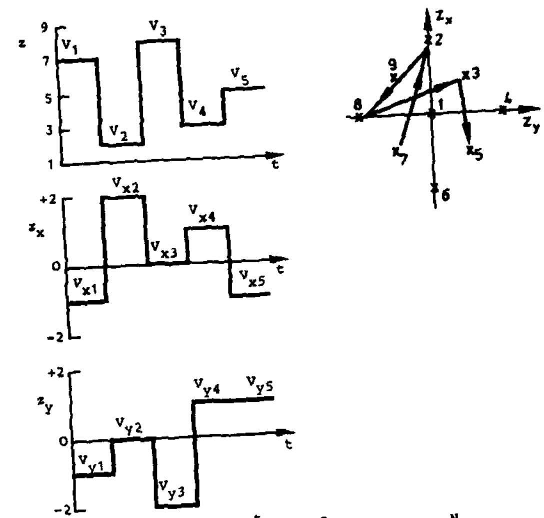


Fig. 7 Target Representations in the Mil-plane

• Targets



	t					Σ
v_{xi}^2	1	4	0	1	1	7
v_{yi}^2	1	0	4	1	1	7
$v_{xi}\ v_{yi}$	1	0	0	1	-1	1

$$\rho_{xy} = \frac{\sum_{i=1}^{N} v_{xi}\, v_{yi}}{\sqrt{\sum_{i=1}^{N} v_{xi}^2 \sum_{i=1}^{N} v_{yi}^2}}$$

$$\rho_{xy} = \frac{1}{7}$$

Fig. 8 The Correlation Coefficient of Target Configuration A

Fig. 9 Distribution of Distances for Configuration B

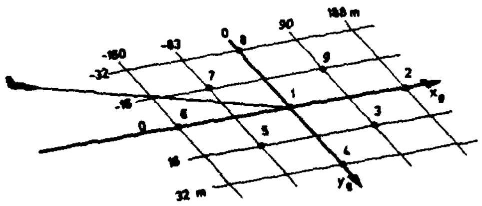


Fig. 10 Target Area of Configuration A

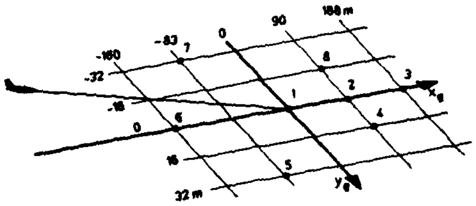


Fig. 11 Target Area of Configuration B

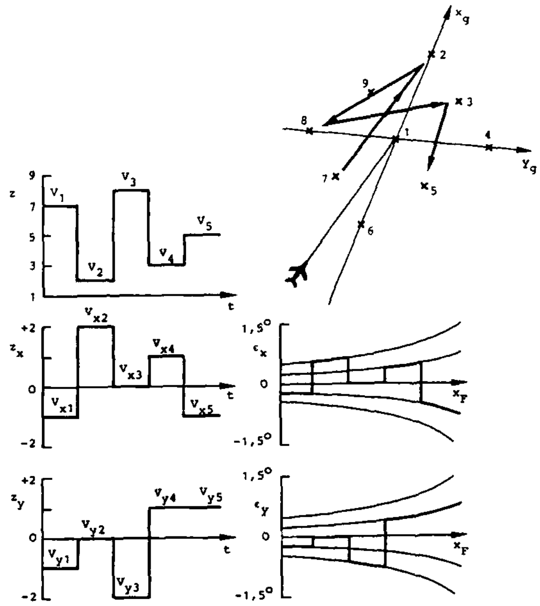
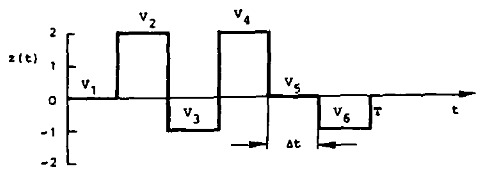


Fig. 12 Input Signal and Visual Angles



$$\frac{|Z(\omega)|^2}{T} = \underbrace{2\Delta t\,\frac{1-\cos\Omega}{\Omega^2}}\left[\frac{1}{N}\sum_{i=1}^{N} V_i^2 + \frac{2}{N}\sum_{i=1}^{N-1} \cos i\Omega \sum_{j=1}^{N-i} V_{i+j}\, V_j\right]$$

with $\Omega = \omega\Delta t$

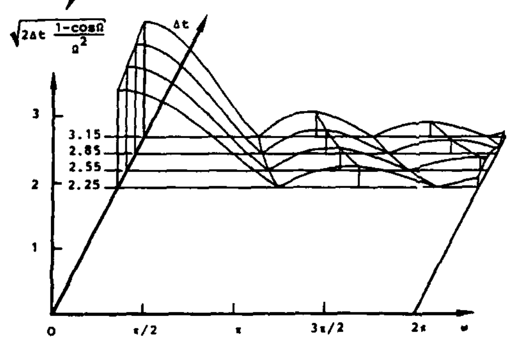


Fig. 13 Power Spectrum of a Multi Step Input Signal and a Presentation of the Dominant Factor

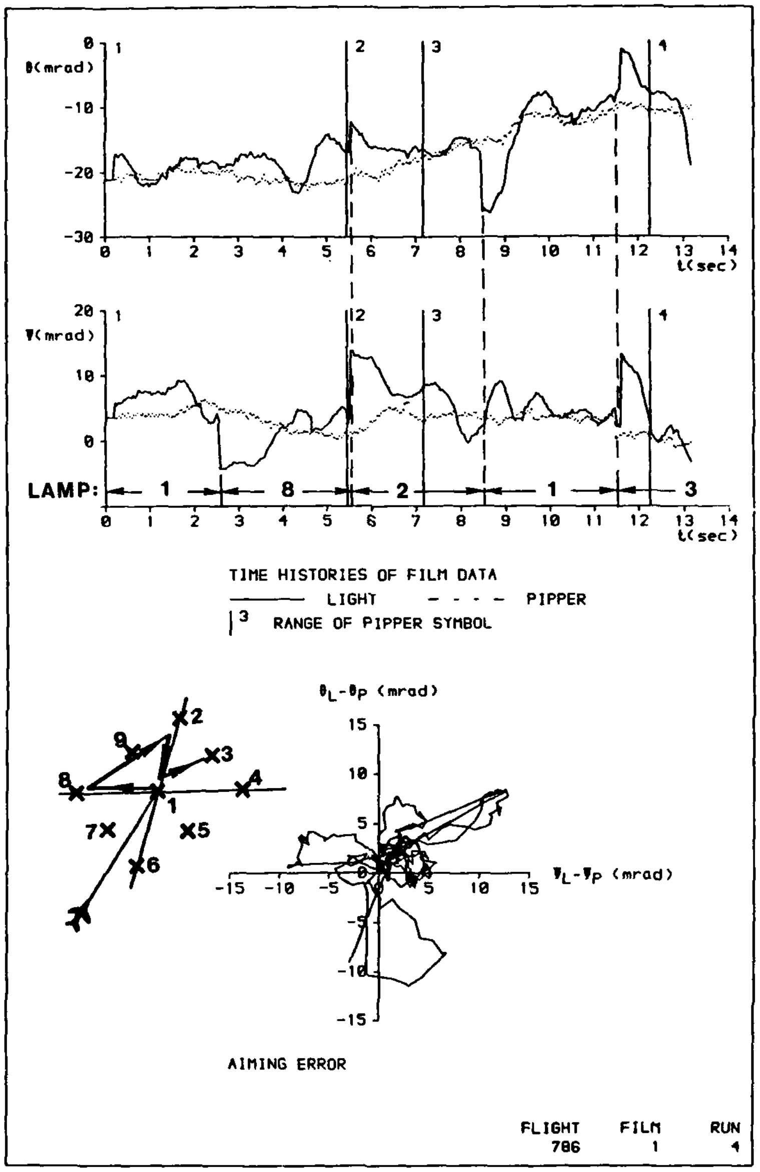


Fig. 14 Plots of Film Data - Feedback Mode C

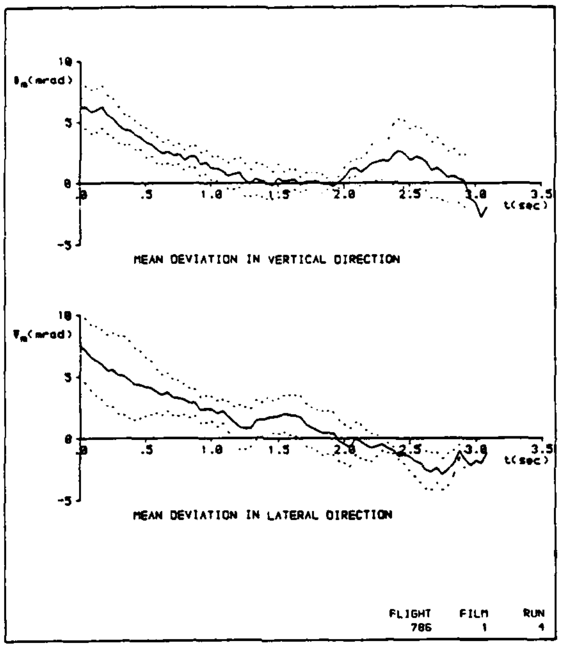


Fig. 15 Averages of Time Histories after a Target Leap
Feedback Mode C

———— Mean Values
-------- Limits of Confidence

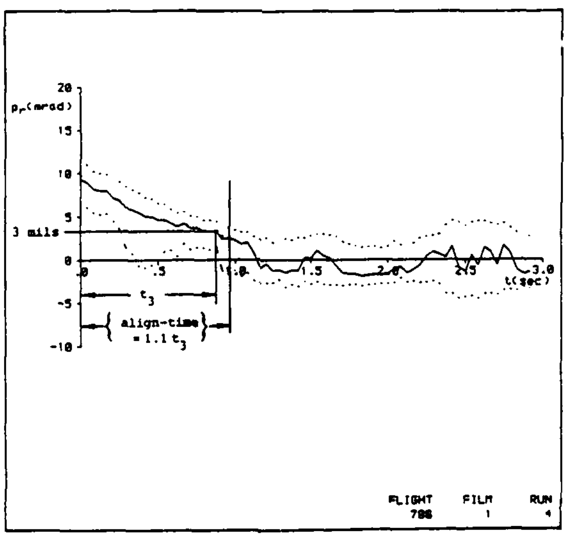


Fig. 16 Definition of Align-Time
Feedback Mode C

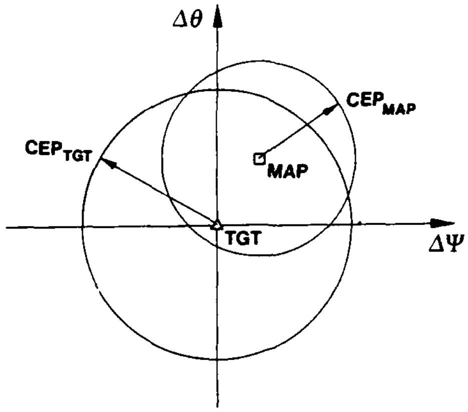


Fig. 17 Circular Error Probabilities

MAP Mean Aiming Point
TGT Target

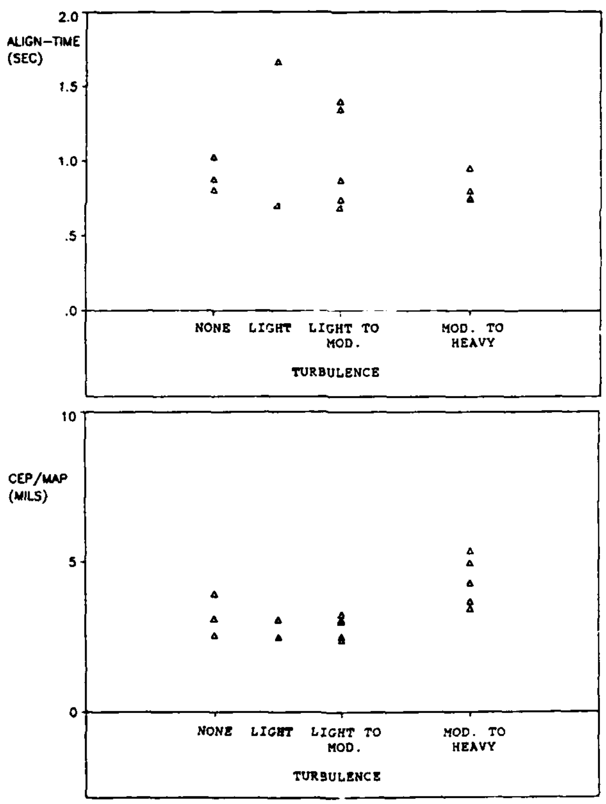


Fig. 18 Influence of Turbulence on Alignment and Stationary Tracking

Feedback Mode C
Target Configuration A

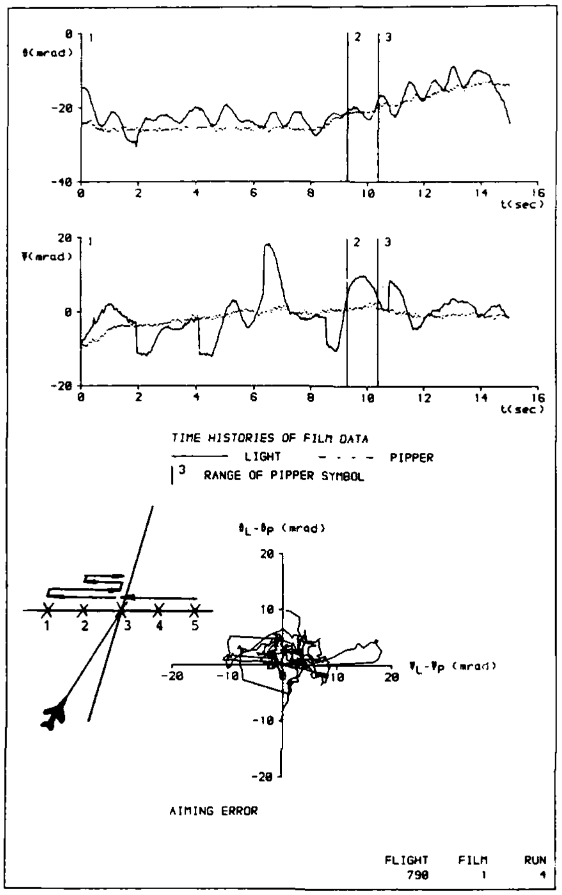


Fig. 19 Film Data with Pitch Oscillations
Feedback Mode A

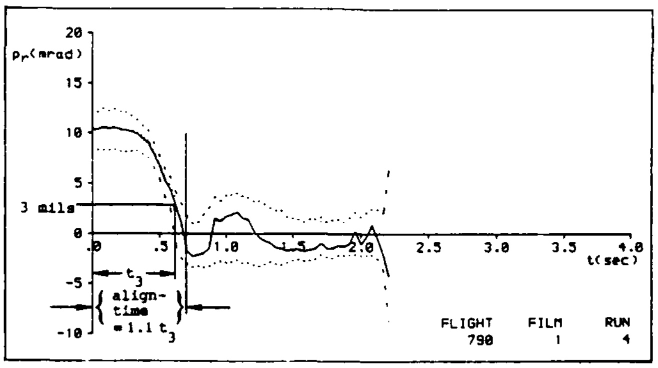


Fig. 20 Determination of Align-Time
Feedback Mode A

(Note: Oscillations are Suppressed by Averaging)

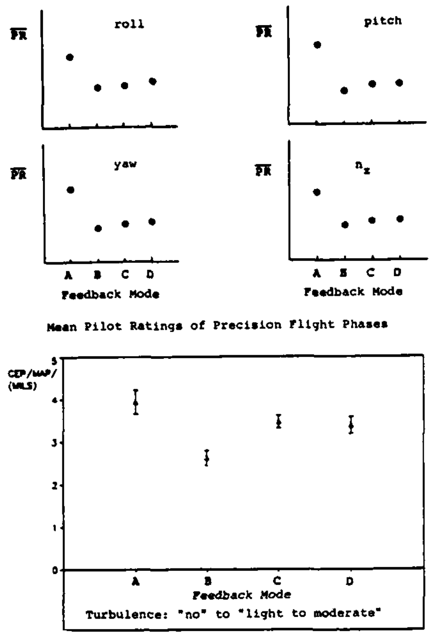


Fig. 21 Pilot Ratings and Results from Film Data of Stationary Tracking

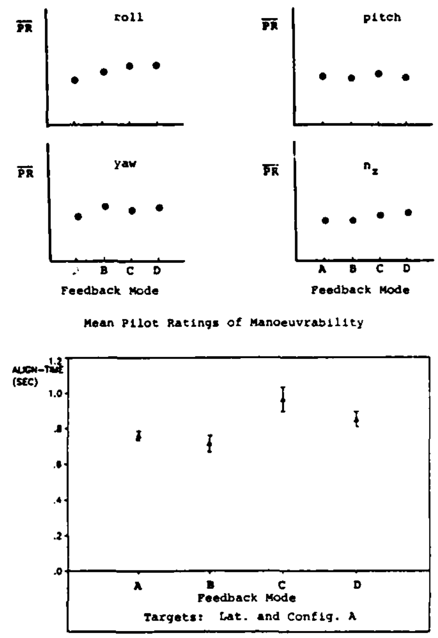


Fig. 23 Pilot Ratings and Results of Film Data of Alignment

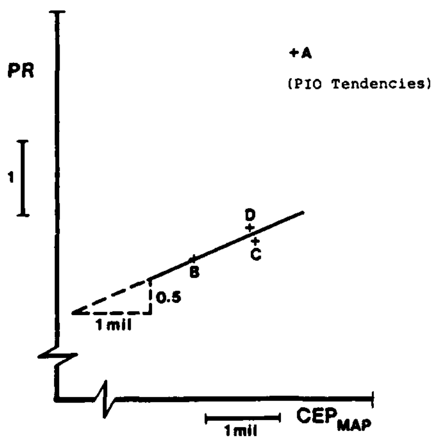


Fig. 22 Interdependence of Circular Error Probabilities and Mean Pilot Ratings of Precision Tracking
A, B, C, D Feedback Modes

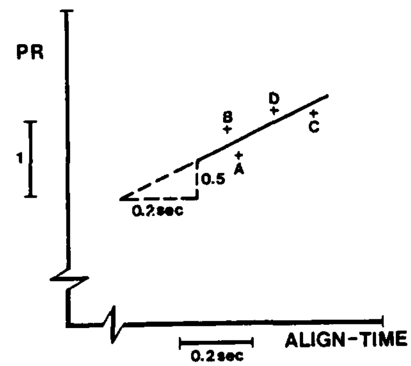


Fig. 24 Interdependence of Align-Time and Mean Pilot Rating of Manoeuvrability
A, B, C, D Feedback Modes

Appendix A

Determination of a Mean Radius $\rho_r(t)$

After a target leap the aiming errors in pitch and yaw have a deterministic and a random portion

$$\Delta\theta = \Delta\theta_d + \Delta\theta_r$$

$$\Delta\Psi = \Delta\Psi_d + \Delta\Psi_r .$$

The components can be utilized for a definition of a deterministic and a random portion of the radius

$$\rho_d = \sqrt{\Delta\theta_d^2 + \Delta\Psi_d^2}$$

$$\rho_r = \sqrt{\Delta\theta_r^2 + \Delta\Psi_r^2} .$$

A quantity Q is utilized for a representation of the deterministic portion

$$Q = \frac{1}{n(n-1)} (S_\theta^2 + S_\Psi^2 - S_{\theta\theta} - S_{\Psi\Psi})$$

where

$$S_\theta = \sum_{i=1}^{n} \Delta\theta(i) \operatorname{sgn} \Delta\theta_0(i)$$

$$S_\Psi = \sum_{i=1}^{n} \Delta\Psi(i) \operatorname{sgn} \Delta\Psi_0(i)$$

$$S_{\theta\theta} = \sum_{i=1}^{n} \Delta\theta^2(i)$$

$$S_{\Psi\Psi} = \sum_{i=1}^{n} \Delta\Psi^2(i) .$$

The quantities $\Delta\theta_0$, $\Delta\Psi_0$ are defined by the pitch and yaw leap at the time when the leap occurs.

The number n is defined by the numbers of leaps within a dive.

The statistical average of Q

$$E\{Q\} = \rho_d^2$$

represents the deterministic time history of the characteristic motion and has no bias caused by the random process.

The quantity Q and its standard deviation can be calculated from test data.

A radius could be ob ained from Q by taking the square root. However, the quantity Q may be negative. All values can be transformed using the equation

$$\rho_r = \sqrt{|Q|} \operatorname{sgn} Q .$$

This quantity ρ_r may be regarded as a mean radius which is diminished due to estimated noise effects. Time histories are shown in Fig. 16 and 20.

Appendix B

Ground Attack Configuration:

	s^0	s^1	s^2	s^3	s^4	
$\alpha/\delta es$	-.0086	-1.5	-.019			deg/mm
$\theta/\delta es$	-2.38	-1.406				deg/mm
Δ_{long}	.0985	23.5	5.67	1.0		
$\beta/\delta as$	-2.15	-21.8	.275	.0143		deg/mm
$p/\delta as$	-.592	-125	-49.3	-5.64		deg/sec/mm
$r/\delta as$	-.4	2.09	2.2	-.0327		deg/sec/mm
$\phi/\delta as$	-128	-52.5	-5.59			deg/mm
$\beta/\delta rp$	1.07	33.9	4.89	.114		deg/mm
$p/\delta rp$	-.749	-135	15.6	1.27		deg/sec/mm
$r/\delta rp$	-4.24	-1.86	-23.9	-2.53		deg/sec/mm
$\phi/\delta rp$	-135	19.8	1.71			deg/mm
Δ_{lat}	13.9	262	49.5	12.5	1.0	

Fighter Configuration:

	s^0	s^1	s^2	s^3	s^4	
$\alpha/\delta es$	0.0	-.89	-.0198			deg/mm
$\theta/\delta es$	-1.137	-.948				deg/mm
Δ_{long}	0.0	9.0	4.2	1.0		
$\beta/\delta as$	0.0	-.3863	-.1385	-.0507		deg/mm
$p/\delta as$	0.0	-134.8	-29.72	-10.05		deg/sec/mm
$r/\delta as$	0.0	-3.115	-1.56	-.369		deg/sec/mm
$\phi/\delta as$	-134.8	-29.72	-10.05			deg/mm
$\beta/\delta rp$	0.0	5.746	1.5	.0179		deg/mm
$p/\delta rp$	0.0	-29.26	.449	.8638		deg/sec/mm
$r/\delta rp$	0.0	-1.33	-5.43	-1.27		deg/sec/mm
$\phi/\delta rp$	-29.26	.449	.8638			deg/mm
Δ_{lat}	0.0	49.0	26.25	7.5	1.0	

IN FLIGHT RELIGHT TESTS ON AM-X SINGLE ENGINE FLY-BY-WIRE AIRCRAFT

G. Menso and G. Gamalero
AERITALIA
Combat Aircraft Group
Flight Test Department
I-10072 Caselle Torinese (Italy)

SUMMARY

The in flight relight evaluation of a single engine aircraft could be considered one of the more interesting aspect of flight testing.
When the engine is intentionally shut down in flight to attempt the relight, possibility of an unsuccessful one must always be taken into account and therefore a flame out landing must be contemplated, otherwise the aircraft is lost.
This event may be particularly serious on a fly-by-wire aircraft on which the loss of the engine and all its driven accessories will make control very difficult if not impossible.
Therefore the planning and the execution of this kind of test is not only concerning the propulsion aspects, but involves all the aircraft, because full evaluation (both theoretic and flight test) of aircraft and systems performance and reliability in engine out condition must be carried out in advance of the actual relight tests.
Besides the relight evaluation must be carried out as early as possible on the program to allow safe performing of those tests that may cause a flame out.
This paper describes all the activities carried out by AERITALIA Flight Test Department in connection with relight tests of AM-X and may therefore be looked upon as a reference for smooth test planning and execution.

1. INTRODUCTION

The AM-X is a single engine fly-by wire attack fighter with a limited mechanical back-up on pitch and roll controls. (Fig. 1).
The handling of the aircraft in case of total power loss (engine flame out) is degraded to level 3 and the pilot work load may become very high, especially in the landing phase.
The engine R.R. RB168-MK807 fitted to the aircraft has no provision for assisted in flight restart and therefore it may be relit only if sufficient residual rotation speed remains either for inertia (hot relight case) or windmill (cold relight case).
The Overall Flight Test Plan (OFTP) includes several tests that may cause an engine flame out, that cannot be performed within the glide recovering range of the aircraft. These tests include high angle of attack/spin evaluation, high altitude engine handling, gun and missile firing, etc..
All these tests were therefore considered "high risk", unless the engine was proved to be easily relightable and safe procedures were defined.
A careful approach to the planning of the test activity was therefore made to allow achievement of the best results with the minimum risk.
The basic programme requirements may be summarized as follows:

- Maximum safety was required, as the implications of a prototype loss in the early stage of the programme were considered unacceptable.
- In flight relight tests had to be performed as soon as possible in the OFTP to allow a smooth prosecution of the high flame out risk tests.

These basic requirements lead to follow four lines of decision analysis as hereinafter described:

A) Reduction to a minimum of the risk of unsuccessful relight.

B) Choice of operation range and/or revision of test aircraft configuration to:
   - Assure the best capability of engine flame out landing.
   - Avoid unnecessary cost and complexity, thus minimizing the impact of the tests on the OFTP.

C) Accurate planning of the subject tests on the OFTP to:
   - Learn before commencing the actual relight tests all what was required to safely performe (if required) a flame out landing.
   - Define a sufficient relight boundary and optimize the associated procedures in time to allow prosecution of the other tests.

D) Flight test instrumentation (FTI)/Telemetry coverage improvements to:
   - Assist the pilot on the pre-relight checks and in any case of failure.
   - Extensively monitor the ancillary systems to detect any dormant failure or to take care of incipient failure.
   - Analyse in real time the tests results thus speeding up test progress.

The decisions taken and the following actions on the above lines will now be described in detail.

2. MINIMIZING THE UNSUCCESSFUL RELIGHT RISK

An unsuccessful relight may be caused by one or more of the following reasons:
- Insufficient rotational speed on windmilling condition.
- Engine seizure due to mechanical damage or oil system malfunction.
- Failure of the ignition system.
- Failure of the fuel system.
- Misprocedure.

All these possibilities were carefully examined and a number of actions carried-out to minimize the probability to face them in anger.
To verify the existence of sufficient engine windmilling speed in the required envelope an extensive evaluation was planned on the Rolls Royce Altitude Test Facilities (ATF) in Derby (U.K.) using an engine with all the aircraft loads and bleeds properly simulated.
These tests will be described in details in the following paragraph 2.1.
The possibility of engine mechanical damage or failure was considered very unlikely because the engine to be used for the in flight trials was new; in any case continuous monitoring of starting and run down time and of the vibration level was carried out.
To avoid any possibility of systems failure that may cause unsuccessful relight, appropriate procedures were drawn-up and used during the ground preflight checks as detailed on chapter 2.2 and before each in flight shut down as detailed on chapter 2.3.
These procedures were performed using telemetry facility to simplify the pilot job and to cover also those items not checked by cockpit panel (like igniters operation, battery voltage, hydro accumulators pressures etc.).
In addition the igniter plugs were periodically inspected to verify their condition; this gave an opportunity to assess plug wear rate.

2.1. ATF Testing

An envelope in which sufficient windmilling was expected to assure cold relight was provided by the engine Contract Spec. although it was drawn not taking into account air bleed and mechanical loads effects consequent to engine installation.
An extensive campaign on Altitude Test Facilities (ATF) was carried out to verify the effect of installation on relight characteristic.
Inlet distortion was simulated through specially cutted distortion plates, while dummy loaded accessories/hydraulic pumps and IDG's were fitted on Accessories Gear Box (AGB).
Air from HP and LP compressors was also extracted to simulate the actual aircraft bleed.
One of the igniters was at the limits of operability, to simulate the "worn" case.
Cold and hot relights have been performed inside and outside the Contract Spec. envelope with the aim to determine a successful relight boundary, to gather good data about windmill speed and to demonstrate repeatibility of the data.
For this reason 3 attempts have been made for each test point and only when all three were successful, the point was considered achieved.
During this test the performance of hydraulic pumps at very low engine drive speed was assessed. Some checks were also done to verify the capability of the engine to recover windmill speed by increasing Mach number from condition well outside relight envelope, simulating an actual dive profile of the aeroplane.
The tests were carried out in July 1985 and the following results were achieved:
- The Contract Spec. relight envelope was demonstrated also for the installed engine.
- The minimum windmill speed to achieve positive relight was found to lay well outside this envelope. Also high Mach number tests were successful.
- A good definition of the actua windmill speed to be expected on aircraft was achieved (with accessories loaded and unloaded).
- Effect of fuel type was assessed and found negligible.
- Effect of air bleed on windmill speed/relight characteristics was found negligible, while accessories mechanical drag was found to reduce the achievable windmill speed by about 3%.
- The capability of the engine to recover its windmill speed consequently to an airspeed undershoot was demonstrated.
- The capability of hydraulic pumps to deliver enough pressure/flow to power primary flight control at low windmill speed was demonstrated.

A summary of performed tests is shown at fig. 2.

2.2. Procedure pre-flight

Before each of the dedicated flight, in addition to the standard preflight procedure, some special checks were scheduled.
These checks included:
- A full check on ignition operation, checking each single plug by disconnection of the appropriate circuit breaker.
- A check of the emergency rudder operation, for what concern both hydraulic supply and computer DC powered operation.
- A check of all emergency trims.
- A check of the operation of DC supplied emergency fuel pumps.
- A check of the operation of the special battery supplied FTI system.
- A check of the proper auto reversion of the essential DC bus to battery supply consequent to IDG's deselection.
- A check of the emergency demist.
- An engine hot relight to overcheck the complete operation of engine and related systems.

During the taxi phase a check of emergency brakes was also performed.
All the above checks, although time consuming, were found very useful to assure perfect serviceability of all emergency devices and to give confidence to the pilot about the safety devices and the procedures.
The careful monitoring of all these procedures through the telemetry gave confidence about the real time facilities and the correct operation of the FTI.

2.3. In flight check

The first test of each flight was performed simulating an engine out descent, approach and touch and go with the engine at idle and mechanical flight controls.
This with the purpose to familiarize the pilot with the profile to be adopted for the specific test, taking all the necessary reference points, and to check the weather conditions in terms of visibility and umidity (to evaluate possible misting condition in the ECS off condition).
When these tests were successfully performed the actual relights were carried out.
In addition and in particular for the first phase of testing the system to be lost as consequence to engine flame out (ECS, HYDRAULIC PUMPS and GENERATORS) were all selected off before actual shut down to check proper switch over to emergency, thus avoiding non return situation.
The flaps were preselected in the MANOEUVRE position, that was found the optimum configuration for an engine out approach and landing.

3. CHOICE OF AIRFIELD AND AIRCRAFT CONFIGURATION

Consideration was made early in the test planning whether to perform the tests on other airfield than the AIT Caselle home base.
Caselle, (see fig. 3) is a civil airport with a 1.6 mile single runway and with the over-the-field area subjectet to civil traffic restrictions. The AIT test are normally conducted on a test area,the closest corner of which being about 20 miles from Caselle.
The use of airfields like Edwards AFB or Istres (France) that have their own over-the-field area fully available for test purposes and unlimited landing runways, was undoubtly safer in case of forced engine out approach and landing.
This opportunity was disregarded because of the complexity and the cost of such an operation and because it was not felt compatible with the OFTP. This kind of operation would infact imply to dedicate one prototype and all associated personnel and equipment to one task only for a significant period of time.
An agreement was reached with the airport authority to allow priority to the test aircraft on predetermined slots in the part of the day when civil traffic is minimum, while a study of meteo statistic did show the summer weather to be ideal.
The performance evaluation carried out in advance confirmed the validity of this decision, allowing the definition of a satisfactory emergency procedure from the test range to the Caselle airfield.
Modifications were then introduced to the concerned prototype to optimize the flame out performance.

3.1. Aircraft configuration in engine out condition

The flame out of the engine causes in the AM-X, in addition to primary power loss, a degradation of the basic systems as hereinafter detailed:

Primary flight controls.

Longitudinal control: In normal condition the longitudinal controls functions are performed through stabilizer and elevator.
In engine out condition the control signal of the stick is mechanically transmitted to the elevator which is therefore controlled; the graduality of the transient between hydraulically powered to manual control is guaranteed by two accumulators.
Stabilizer surface and its functions (i.e. longitudinal control and artificial damping) are lost. The stabilizer actuation system is still able to exercise a trimming function, through an emergency electrical motor powered by the battery.

Lateral control: In normal condition, the lateral controls functions are performed through spoilers and ailerons.
In engine out condition the control signal of the stick is mechanically transmitted to the ailerons which are manually controlled; the transient from hydraulical power to manual control is guaranteed by two accumulators.
Spoiler surfaces and their functions (i.e. later control, artificial damping, air brakes and lift dumpers) are lost.

Directional control: In normal condition, the directional functions are exercized through rudder actioning.
In engine out condition, the rudder control is no more operative and the surface will lay in the fin wake.

Secondary flight controls (flap and slat):
It will remain locked on preselected position.

Environmental Control System (ECS) & Pressurization:
It will be lost in a time dependant on leakage. An emergency ram air scoop is available at low speed/altitude. No demist is available.

Electrical power:
Alternate Current (A.C.) power is lost. All essential equipments are battery supplied for at least 20 min. by auto switchover to essential Direct Current (D.C.) bar.

Fuel system:
Normal fuel boosting is A.C. power operated and will therefore be lost.
Emergency D.C. operated pump is available and will automatically be switched on when normal boosting stops, by means of a pressure switch on supply line.

Undercarriage (U/C):
Could be lowered with a dedicated accumulator.

Brakes:
Emergency brakes will be operated through a dedicated accumulator (antiskid will be lost).

Steering:
Steering operation is lost due to either hydraulic and electric power loss to the system.
Aircraft can be controlled using asymmetric emergency braking.

3.2. Modifications to aircraft standard introduced

In accordance with the results of ground and flight investigatory tests, in order to bring the aircraft to the maximum safety standard envisaged, a number of modifications have been embodied on the aircraft in order to allow the availability of some system, considered highly desirable in case of engine out landing.
The modifications embodied are the following (as shown on fig. 4):

Emergency rudder.
The unavailability of rudder control in engine out condition was considered hazardous for landing, specially in the case of lateral gusts during the final approach phase.
The aircraft was therefore implemented with an "emergency rudder" system introducing the following modifications:
a) - Connection of the rudder actuator to the hydraulic gun accumulator (not utilized, since the gun is not fitted on the prototype).
b) - Connection of the n° 2 Flight Control Computer electrical supply to the battery bus bar.
c) - Installation in the cockpit of an "emergency rudder" switch.

The operation of the above switch determines: hydraulic power supply to the rudder actuator, automatic reset of the rudder failure on the interested Flight Control Computer, full rudder authority (± 30°) also in the manoeuvre flaps configuration, otherwise limited to + 7.5°.

Emergency steering.
The availability of the steering during the ground roll in case of engine out landing was considered mandatory, and for this reason the "emergency steering" system was embodied on the aircraft.
The following modifications have been therefore introduced:
a) - Connection of the steering actuator to the gun accumulator (as already done for the "emergency rudder").
b) - Connection of the steering system electrical supply to the battery bus bar.

Selection of the emergency steering was left on the same switch on the stick used for normal steering engagement.
The tests carried out, results achieved and selection procedure of these two emergency systems are described in para 4.

Emergency demist.
An emergency D.C. powered demist fan was installed on the aeroplane, replacing the standard A.C. powered system.
This new system, tested on actual misting conditions, was found to be only partially effective and therefore a special anti misting fluid was used (CHEMICAL COMMODITIES AGENCY INC. MIL-A-21071B).

Emergency bus bar (PP3) and battery bus (PP4).
A revision of the load distribution between the TRU supplied bars and the battery supplied bars (PP3 / PP4) was done.
This led to the decision to partly revise this distribution, connecting all ignition associated items on the PP4 (always battery supplied) and adding on the PP3 emergency bus some load like the pitot heater, the above mentioned emergency demist, flight computer and steering box, plus same cockpit instrument judged useful for the purposed test.

3.3. Modification disregarded

In the early phase of the relight test planning some further modifications to aircraft standard were considered but then disregarded.
It is worth to mention such modifications and the reasons for disregarding.

In flight engine starter.
Before reliable data on the windmilling speed of the loaded engine became available from ATF tests, the possibility to fit a special starter to motor the engine in any case to the required speed was taken into account.
The complexity of design, purchase and qualification of such a system, associated with the good results from ATF tests and the demonstration of the good capability of the aircraft to safely land in engine out condition, led to disregard this modification.

Duplication of plugs power supply.
Consideration was also made whether to separate the power supply of the two igniter boxes and plugs.
Also this modification, even if felt of some utility in order to increase safety, was disregarded because of the complexity and the very long time required to realize such a system.
Instead a procedure to closely check ignition system serviceability was adopted.

Lift dumpers and airbrakes emergency operations.
These systems were proved to be not mandatory for a safe recovery of the aeroplane and therefore any attempt to assure some hydraulic supply to them was disregarded.

Additional hydraulic power supply.
Again the preliminary tests performed were very encouraging because of the long endurance of the already available accumulators and of the hydraulic supply characteristics at low engine speed. Therefore any attempt to install an emergency hydraulic power source (RAT or EPU) was judged unnecessary and disregarded.

4. RELIGHT TEST PLANNING

An extensive plan of investigatory ground and flight test was drawn and scheduled to be performed at the early possible stage of prototype activity to collect all the information considered necessary for a safe recovery and landing of the aircraft in case of unsuccessful relight.
These test include basically:
- Evaluation of glide descent and landing performance with various flap setting.
- Evaluation of degraded controls handling qualities.
- Ground and inflight evaluation of emergency systems operation and performance.

All these tests are extensively detailed in the following subchapters.

4.1. Evaluation of glide performance

4.1.1. Choice of aerodynamic configuration

In engine flame out flaps and slats will remain locked in the preselected position; therefore an analysis was carried out to identify the best aerodynamic wing configuration to be adopted during the relight trials.
For this reason, using flight matched aerodynamic data, max efficiency, best glide speed and maximum glide distance were estimated for the selectable flap/slat position.
According to the results of this analysis, the high lift configuration was rejected because of the low efficiency and therefore reduced glide distance capability.
Then the manouvre wing configuration was preferred to the clean, mainly because the lower touchdown speed allowed, combined with the sufficient glide distance capability.
For these reasons all the investigatory tests, as well as the actual relight test, have been performed in the manouvre wing configuration.

4.1.2. *Saw-tooth descent*

Purpose of these tests was the experimental verification of the rate of descent (R/D) and dive angle (GAMMA) as a function of speed and engine rating, in order to identify the most representative configuration to be used during the following investigatory trials aimed to assess flame out recovery profile.
For this reason a number of test has been carried out at different engine rating (NH), with undercarriage UP and DOWN at several altitudes.
The tests results did show the most representative configuration of engine out gliding to be the 82% NH and undercarriage DOWN (fig. 5a).

Nevertheless because of the undercarriage speed limitation it was decided to simulate the engine out condition in the 60% NH and undercarriage UP, bearing in mind that this condition was conservative against actual (fig. 5b).

4.1.3. Descent profile for relight test

The choice of the profile for the first relight flight has been dictated by the need to performe the first attempt in the center of the envelope investigated during the ATF tests (see fig. 14).
Besides it was decided to plan the tests at such an altitude that would allow to perform a second attempt in case of need, increasing the windmilling speed by a constant Mach descent.
This second attempt should have been terminated at an altitude (10000 ft) sufficient to acquire if necessary, the flame-out landing circuit (FOLC) keypoint, i.e. 8000 ft, 200 Kts.
As a consequence of above requirements, the flight profile for the first relight flight resulted as shown in fig. 6.
This profile has been repeated several times to check its feasibility and to collect all the useful data, like total distance covered, time elapsed from engine shut down to touch-down, duration of the elevator and aileron accumulators, dive angles to be maintained to acquire step-by-step the profile key-points, etc.
For each of the points performed a descent profile similar in principle to the one above discussed has been identified and actually performed before shut down at least up to the FOLC key-point.

4.1.4. Flame-out landing circuit (FOLC) from the key-point 8000 ft/200 Kts

According to the theoretical and flight test data, and for similarity to other aircrafts, the FOLC procedure has been choosen as shown in fig. 7.
The feasibility and the adequacy of this profile have been initially checked flying with the F.C.S. in normal conditions and with the engine as required; then the engine rating has been reduced to idle and the F.C.S. has been degraded to the configuration most representative of the engine out conditions, i.e. stabilizer, spoilers and rudder disconnected.
Initial simulated flame-out tests showed an area of possible danger when landing without rudder, specially in the case of lateral wind gusts during the final approach phase. For this reason the modification already described to make available (at least for the final approach) the rudder, has been studied and implemented on the aircraft before the first relight flight.
Flight data gathered during these investigatory tests showed that the selected flame-out procedure was easy and successfully carried out also in case of an offset of up to 50 degrees in the heading and up to ± 1500 ft in the altitude at the keypoint.
In this last case, it becomes very important the choice of the moment for undercarriage lowering, since it causes a variation of the rate of descent of 1000 ft/min. which at the speed of 180 Kts (typical of the engine-out final approach) represents a variation of 3° in the flight path angle.
This would allow an adjustment of the flight path of the aircraft in order to achieve the best touch-down point in the runway.

4.1.5. Final approach and landing

As previously mentioned, all the tests were carried out in the manoeuvre flaps configuration and were mainly aimed to define the best incidence/speed during the final approach and at touch-down and to evaluate the adequacy of the emergency longitudinal trim speed.
Results gathered allowed the definition of the suggested conditions for a possible engine out landing in manoeuvre flaps, which are the following.

Stabilizer trim $\delta s = -4 \div -5°$
approach $\alpha = 10°$
touch-down $\alpha = 12°$

These incidence values correspond to an approach speed of approximately 180 Kts and at a touch-down speed of about 160 Kts, which are acceptable specially for an emergency landing.

4.1.6. Aerodynamic braking

These tests were originally planned up to incidences of 14° α , but were interrupted at 11° α , because showed poor efficiency and may cause an overcontrol, with consequent longitudinal pilot induced oscillation (PIO). Nevertheless the tests carried out have been useful to make an estimation of the barrier engagement speed, in the case of landing without brakes (see fig. 8).
Assuming infact a touch-down point located at 1/3 of the runway, and decelerating the aircraft without braking in the remaining runway, the barrier engagement speed was found to be about 110 Kts.

4.2. Evaluation of degraded controls handling qualities.

As said above early evaluation of full mechanical controls confirmed the prediction of poor handling qualities and very high pilot work load to fly in this condition.
The handling qualities on powered controls (no FCS) were instead found good enough to attempt a full landing, provided that rudder is available.
Therefore, after the introduction of the already described emergency systems, the evaluation was concentrated on manual powered controls handling mainly for what concerns the duration of accumulators and hydraulic supply available on engine out condition.

4.2.1. Low engine speed hydraulic power supply.

An investigation was carried out to verify the capability of one hydraulic system to supply emergency flight controls at low engine speed.
The engine windmill speed condition was simulated by supplying a reduced airflow to the engine starter from an external air supply trolley.
The tests showed that at low windmill down to the minimum for relight (about 10%), the flight controls demand could be more than met by one hydraulic system.

4.2.2. Verification of elevator and aileron accumulators endurance.

This verification has been carried out in conjunction with the accomplishement of the descent profile (see para 4.1.3.), continuing the flight after the wawe-off on the runway in hydraulic off condition, up to the reversion in manual mode.
Data obtained, gave the evidence that the duration of both aileron and elevator accumulators before discharge and reversion to manual mode, was of the order of 9 min.
Since the average duration of the various flight profiles from shut-down to eventual landing was found around 7 min., it was concluded that the charge of the accumulators itself was capable to assure hydraulically powered flight controls for the complete engine out descent up to the touch-down. This in association of what said at 4.2.1. gave full confidence about ability to land on powered controls.

4.2.3. Verification of emergency rudder & steering accumulator endurance.

Ground test carried out demonstrated the capability of the system implemented on the aircraft to guarantee the availability of rudder and steering during the approach, landing and ground rolling phases.
Fig. 9, referring to a low speed taxi, shows that the hydraulic power available allows quite a number of rudder and steering manoeuvres before discharging.
Nevertheless, due to the big amount of leakages present in the hydraulic line of the accumulator, this endurance is remarkably reduced if the system is connected long before it is used.
For this reason it was decided to recommended the selection of the emergency rudder steering 1 min. before the foreseen touch-down.

4.3. Emergency systems evaluation.

4.3.1. DC Pump supply characteristics.

A full evaluation of the feeding characteristics in emergency condition (DC pump) was carried out on the ground and in flight.
The results showed that, as predicted, enough fuel feeding was guaranteed at all engine speeds and altitudes.

4.3.2. Battery endurance.

A battery endurance test was performed on ground running the engine with IDG's OFF and simulating full in flight load with externally connected resistance.
The endurance was found to be twice the minimum assumed necessary for relight purposes.

4.3.3. Other emergency utility systems.

Test of the other emergency systems like:

- Emergency demist
- Emergency undercarriage
- Arrestor hook

were successfully performed at an early stage of the program and therefore were not included in this part of investigatory test.

5. FLIGHT TEST INSTRUMENTATION (FTI) / TELEMETRY COVERAGE

5.1. Airborne instrumentation facilities.

In the standard configuration, in case of total A.C. power failure (engine out condition), only the Accident Data Recorder (ADR) remains operating through the standard aircraft battery.
To cope with the requirement of the relight test and to reduce the load on aircraft battery, improving its endurance, it was decided to provide the FTI with its own power source. Therefore dedicated FTI battery (of the same type of normal aircraft battery for commonality reason), additional FTI inverter to supply A.C. powered items and all the required control devices were embodied.
This alternative power source was operable by the pilot through a simple special panel in the cockpit and used only during the specific phase of the flight when it was required (eng. out conditions). The FTI equipement supplied with this alternative power source was:

- the ADR system
- the telemetry to the Ground Station (both transmitters; pilot speech included)
- the C.G. package accelerometers and rate gyros
- the CROUZET digital and analog transducers (Airspeed and Altitude)
- the Time Coding System
- the FM Multiplexing System
- the Tape Recorder
- the PCM System, including all the transducers power supply
- the SYP 820 inertial platform.

All the unnecessary FTI devices like cameras, flutter stores controls, bonkers etc. were automatically switched off at the selection of emergency FTI to minimize the battery load and to increase endurance.
The supply of SYP 820 was found useful also to maintain attitudes display to the pilot in addition to emergency stand-by instruments.
A scheme of the special supply system is shown at fig. 10.
It is worth mentioning that a specific calibration of the pilot engine speed indicator (that was not required to be particularly accurate in the low speed range) was done.
The units (two were tested) to be used were found to be particularly accurate less than ± 0.5%) in all the range, down to about 4% where the indicator drops to zero.

5.2. Real time monitoring facilities.

Extensive use of real time data transmission via telemetry link is made in the AM-X Flight Test Programme.
For the specific relight tests phase this facility was considered mandatory and to cope with program requirement some special features were introduced.
The test aircraft was provided with a dual-channel transmitter allowing continuity of data link with the Control Room where a large number of parameters were displayed throughout the flight.
The standard "Control Room" facility consists of a number of work stations provided with an 8 channels strip chart pen recorder, a video display with capability to recall different "menus" of parameters and in addition, a number of digital displays to monitor general interest flight parameters. A picture of the control room is shown at fig. 11.
A fifth work station, provided with the same facilities but not the pen recorder, is available to the test conductor, giving the possibility to monitor aircraft configuration, flight conditions etc..
During the relight tests the work stations have been used to monitor engine, systems, flight control and flight mechanic parameters, giving the flight test engineers all the information necessary to help the pilot in carrying out the test in the safest way.
Parameters like hydraulic accumulators residual pressures, aircraft and FTI battery voltage, emergency fuel pumps pressures, were carefully monitored to alert the pilot in case of any early sign of system degradation and also to allow decision about the number of actual attempt to be carried on during the flight.
A data bank containing the windmilling data gathered on ATF tests was prepared and this allowed a continuous display of the expected windmill speed to be compared with the actual.
A graphic presentation of the actual aircraft flight condition against envelope and expected windmill speed was also obtained using video graphic display facilities.
Due to the low resolution of the standard pen recorded, extensive use of a ES 1000 electrostatic recorder was made. This gave possibility to display essential parameters like NH, TGT, Throttle on high resolution time histories being paper size of 300 mm fully available for each trace.
Furthermore a temporary memory area was available allowing telemetry real time data storage. This allowed, immediately after each test, more sophisticated analysis in "quasi-real time" (QUARTAS) mode, using special tailored software.
All these facilities allowed to adopt a build up of the test programme whithin the same flight, maintaining the required maximum safety criterion. Some of the menus used for this exercise are shown at fig. 12 and 13.

6. RELIGHT TEST CARRIED OUT

The objectives of the in flight relight test carried out were:
- To demonstrate the hot relight capabilities on the whole operational envelope of the aeroplane.
- To demonstrate the cold relight envelope drawn by the engine manufacturer and successfully tested in ATF, verifying the consistency of the predicted windmil speeds.

Particular care in the test planning was adopted in order to allow a safe build up.
The first test point for a cold relight attempt was chosen with the following criteria:
- To be well inside the predicted envelope.
- To be at an altitude and Mach number that could allow more than one attempt, in case of failure, maintaining constant Mach by loosing altitude (and therefore increasing windmill speed).

The first hot relight test point was planned to be executed before the cold attempt in a condition allowing to get the cold relight condition in case of failure.(See fig.14).
This initial point was successfully repeated three times to assure full consistency of the results.
When this part of the envelope was considered achieved, two more cold attempts were done at higher and lower altitude following the line of constant windmill speed drawn from ATF tests, to fully confirm that line.
Each cold relight was always preceded by an hot attempt, that would had lead to the next planned cold point in case of unsuccess.
When this central area of the envelope was cleared and windmill speed well defined, all the other points were performed reaching in steps the corners of the envelope.
They were always planned in such a way to allow recovery to the already acquired area in case of failure.
This first exploration was done reducing to a minimum the airbleeds and the power off-take from the engine and using JP8 (AVTUR) fuel that, while shown irrelevant insofar the relight itself was concerned by the ATF results, allows a minor improvement of the emergency fuel supply characteristics by the DC powered fuel pump.
Subsequently a number of points on the corners of the envelope have been repeated with alternate fuel JP4 (AVTAG) and then applying the full bleed and load to the engine.
This because, although the pilot could switch bleeds and loads off, it was considered essential to verify that no dramatic deterioration of the relight characteristics would be caused by a pilot misprocedure.
Autoignition system provided on the aeroplane was also tested in the latest phase of testing.
This system provides automatically ignition to the engine when throttle is above IDLE and NH falls to sub idle (NH<50%) thus restarting the engine whitout any pilot operation.
No tests were conducted below 10000 ft because relight performance below that altitude is not expected to worsen and relight done at low altitude, if for any reason unsuccessful, may lead to lose the aircraft, being glide range not enough to get back to the field.
It should always be born in mind that, when an engine is shut down, a remote possibility that it will not relight always exists for a number of unpredictable reasons including misprocedure.

7. TEST RESULTS

All the tests performed were successful and fully demonstrated the expectation and ATF prediction.
In particular the engine windmill speed was exceptionally close to what found on ATF, demonstrating how representative these tests were.
The measured peak turbine gas temperature (TGT) was found about 50 °C warmer than on ATF, but this was clearly due to the soak period used in flight (shorter in respect to ATF).
The data gathered during the tests gave full confidence in the in-flight relight capability of the engine and defined the procedure to be inserted in the pilot handbook.
The reached level of confidence allowed to progress in the testing of the aircraft including high incidence and weapon firing trials.
Summary of the test point is shown at fig. 15.

8. CONCLUSIONS

It can be concluded that the approach followed on planning and performing the tests was very successful.
It was possible to perform on full safety the risky relight test in a relatively short period of time (about 4 months) without interfering with the other aircraft activities (see flights summary at fig. 16) and to get the results in perfect phase with the other program requirements.
The results achieved demonstrated that a careful planning of the tests and an early consideration of the even most remote possibility of failure or error can permit a smooth execution of any flight test programme also when risks are associated.

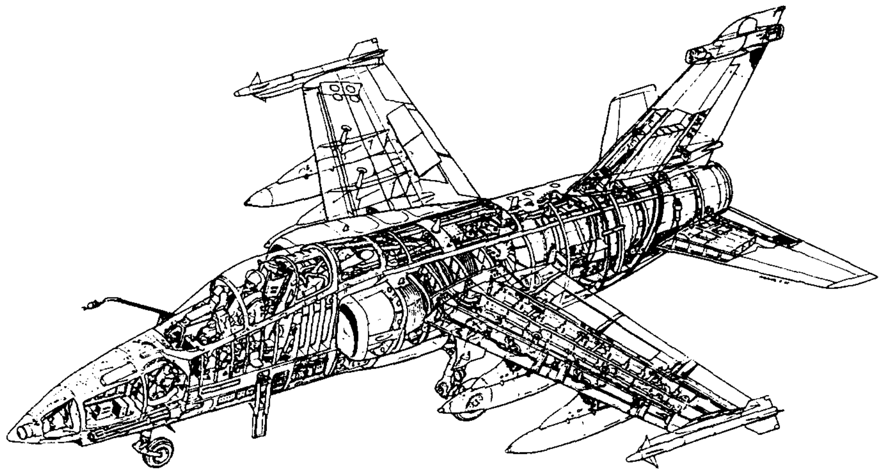


Fig. 1 - AM-X ATTACK FIGHTER

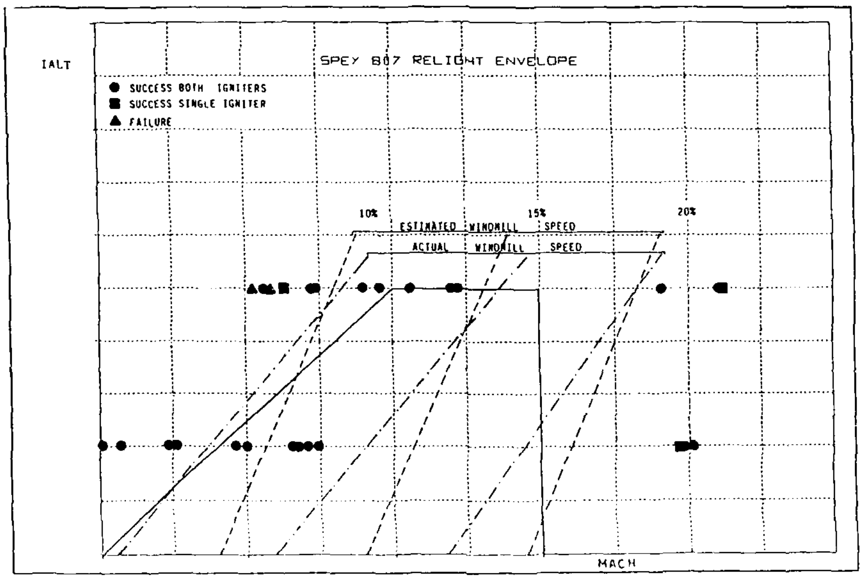


Fig. 2 - ALTITUDE TEST FACILITY (ATF) RESULTS

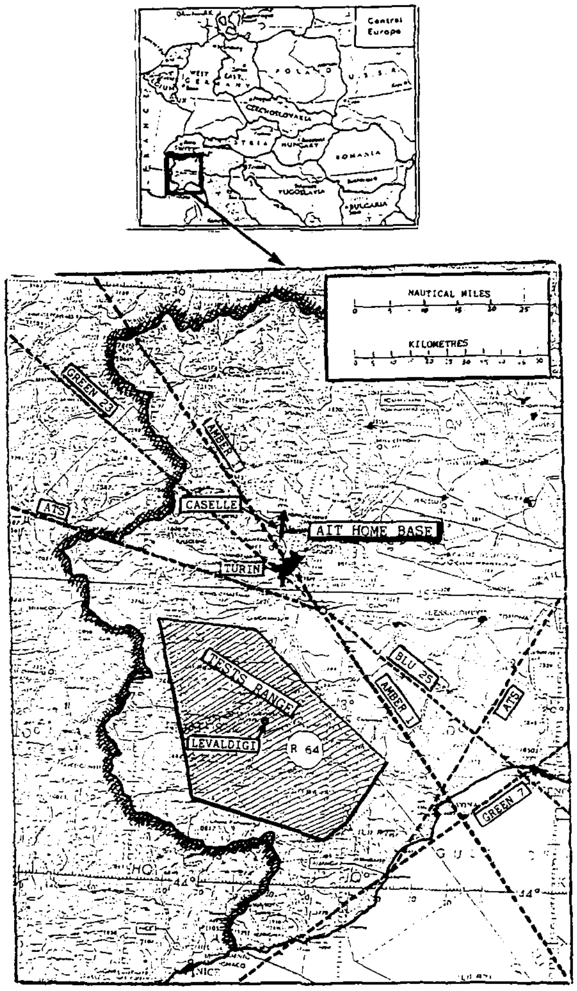


Fig. 3 - CASELLE AIRFIELD LOCATION

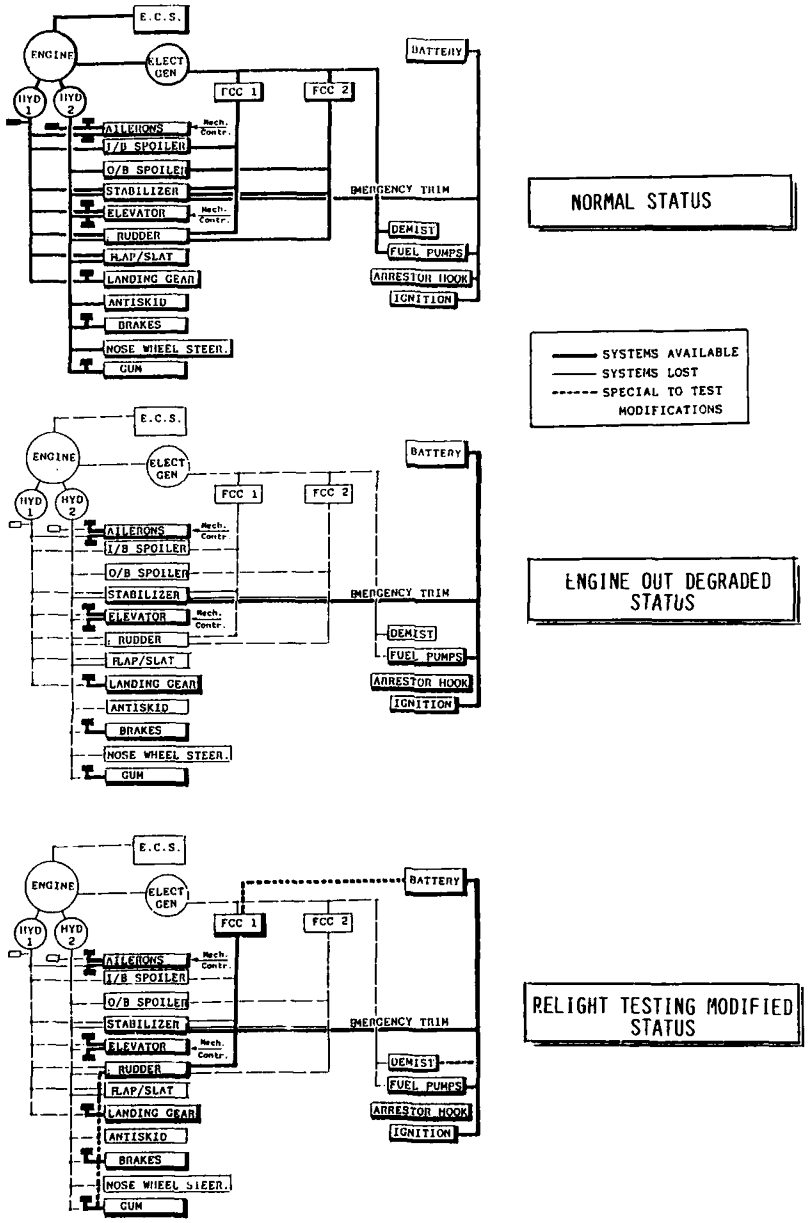


FIG. 4 - MODIFIED FLIGHT CONTROLS & SYSTEMS

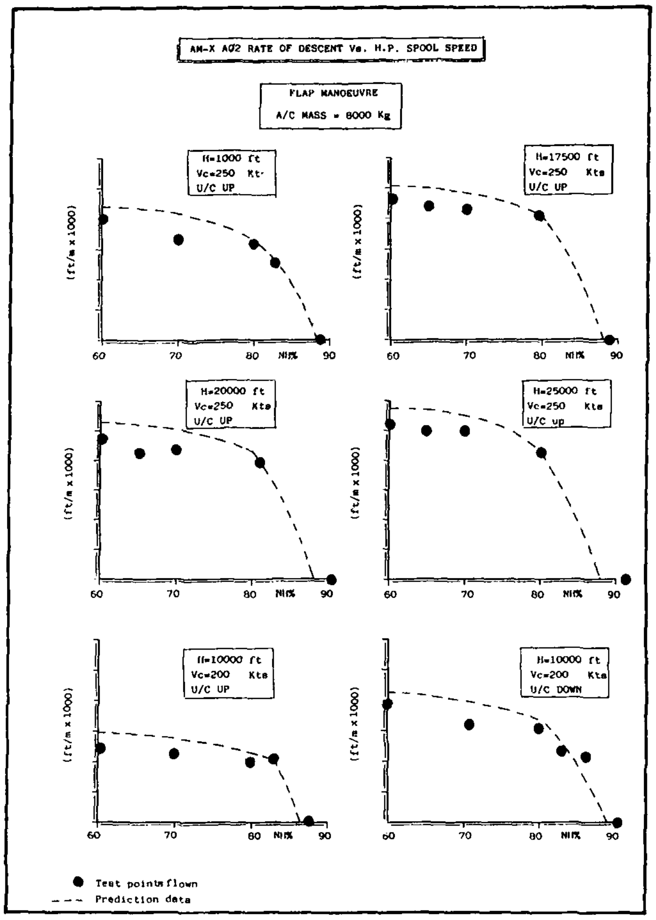


FIG. 5A - SAW-TOOTH DESCENT PERFORMANCE

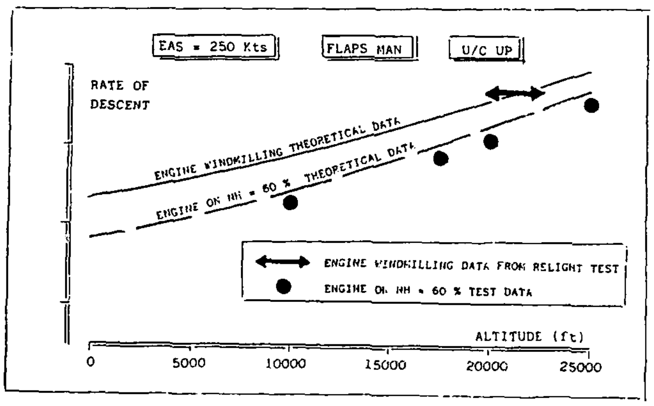


FIG. 5B - RATE OF DESCENT IN ENGINE-OUT CONDITIONS

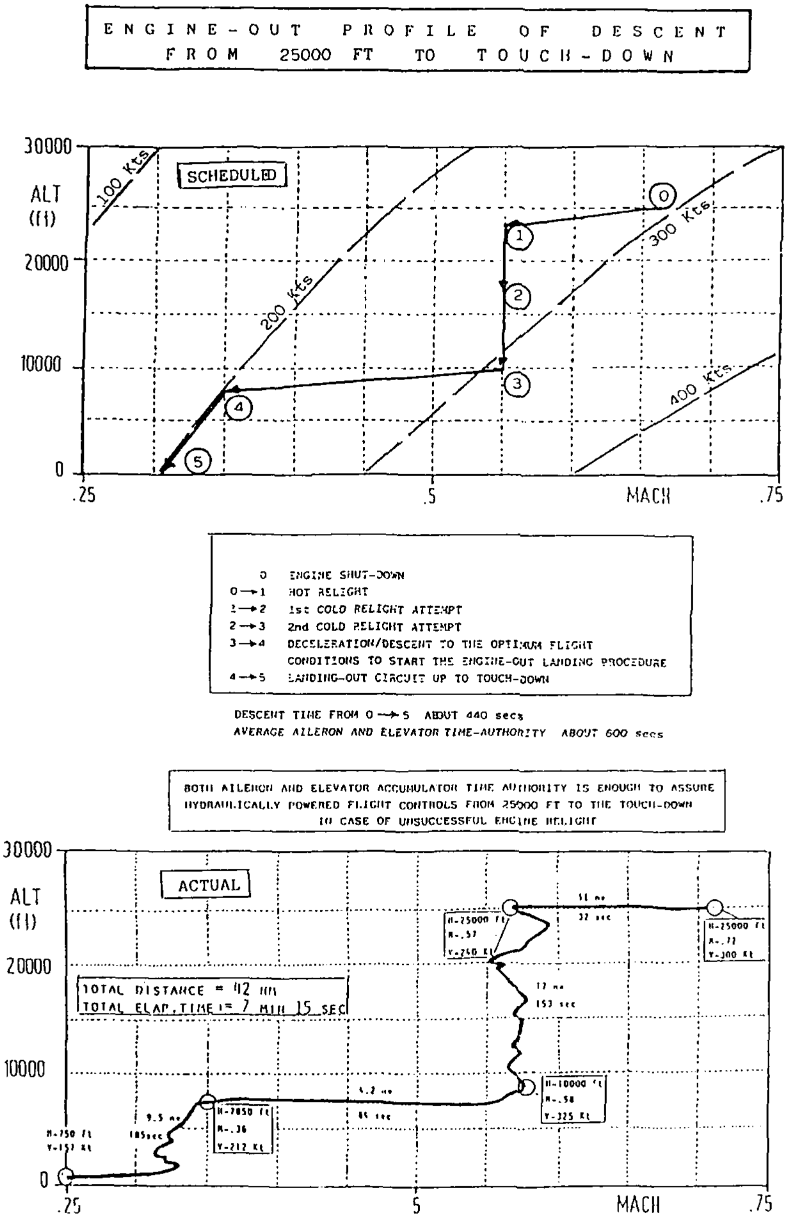


Fig. 6 - 1st RELIGHT PROFILE

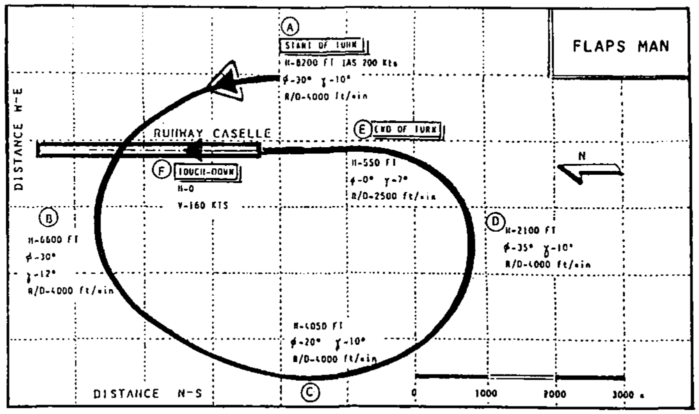
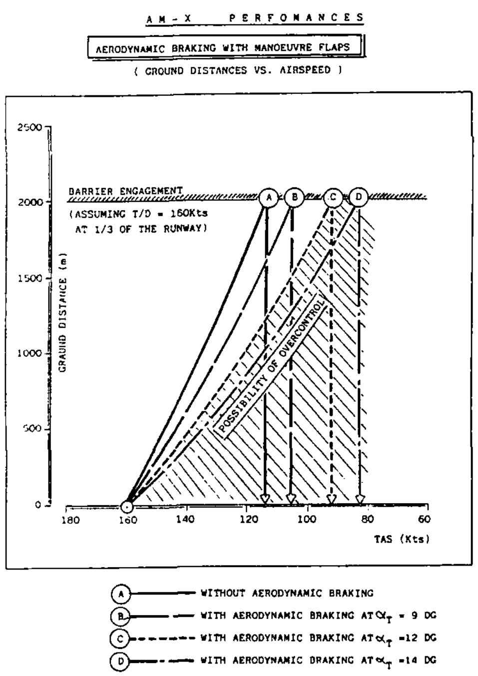


FIG. 7 - FLAME-OUT LANDING CIRCUIT (FOLC)



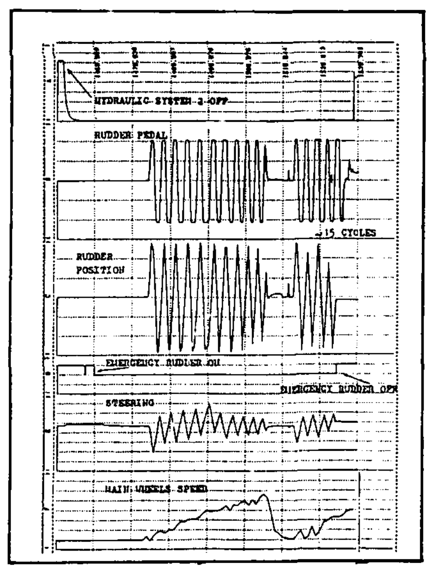


FIG. 8 - BARRIER ENGAGEMENT SPEED

FIG. 9 - RUDDER ACCUMULATOR ENDURANCE

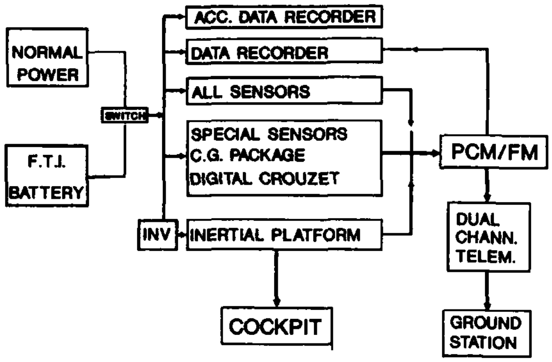


FIG. 10
MODIFIED FTI POWER SUPPLY

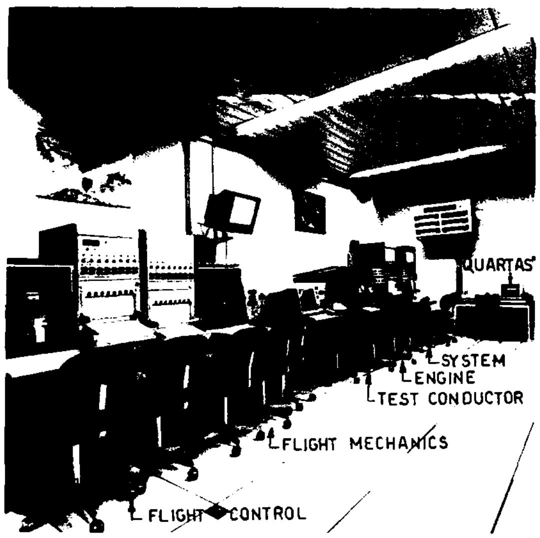


FIG. 11
CONTROL ROOM LAY-OUT

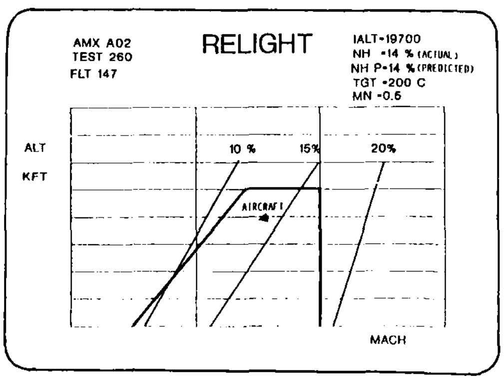


FIG. 12 - REAL TIME MONITORING
(RELIGHT ENVELOPE/PREDICTED WINDMILL)

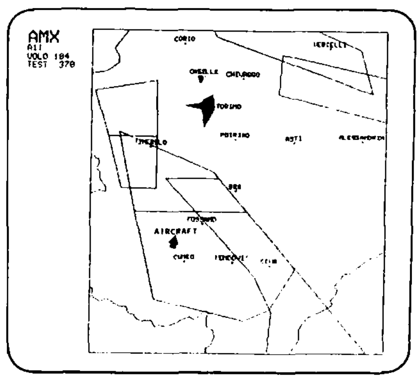


FIG. 13 - REAL TIME MONITORING
(PRESENT POSITION)

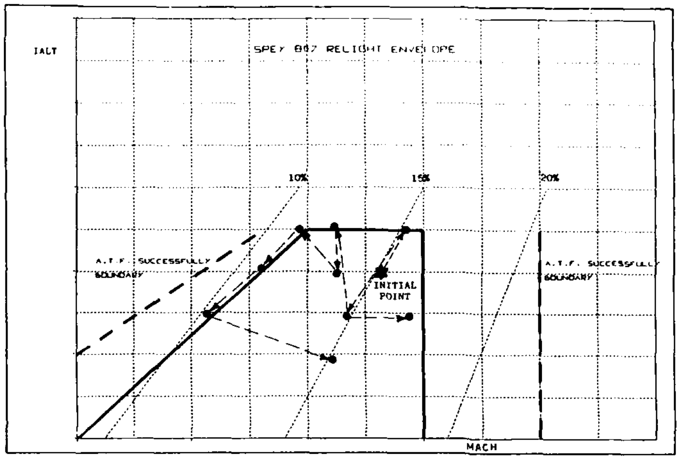


FIG. 14
RELIGHT ENVELOPE
EXPANSION APPROACH

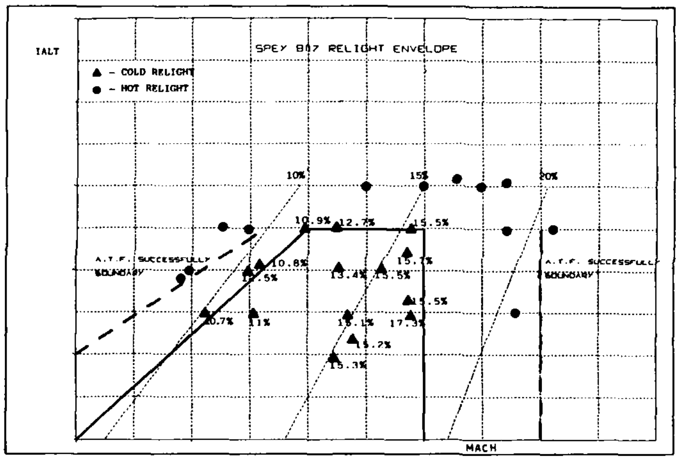


FIG. 15
RELIGHT TEST RESULTS

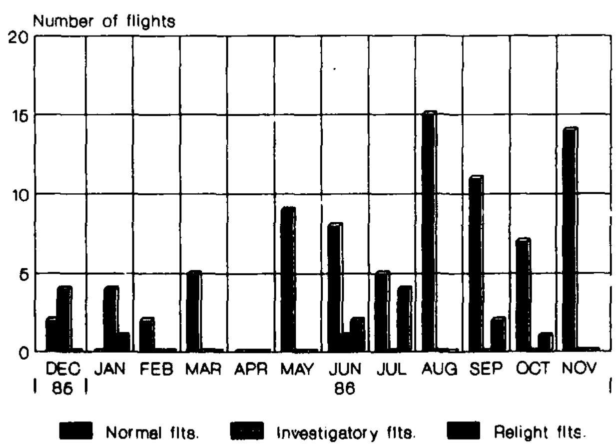


FIG. 16
TEST SUMMARY

FLIGHT TEST RESULTS OF A COMPLEX PRECISE DIGITAL FLIGHT CONTROL SYSTEM

Gunther Schänzer
Director of the Institut for Flight Guidance and Control
Technical University Braunschweig
Germany

ABSTRACT

The presented flight control systems consist of an open loop control system and a more conventional state vector feed back closed system. The open loop actuator control includes a quasistationary aircraft model and a full state command model. For the flight tests of this control system, the minimization of the state vector error was the basic target. To improve the flight test quality, an onboard realtime wind and turbulence measurement system has been used.

The flexible flight test instrumentation including sensors and computers will be described and some typical flight test results will be demonstrated.

LIST OF SYMBOLS

C_D	drag coefficient	$\underline{V}$	airspeed vector
C_L	lift coefficient	$\underline{\dot{v}}_c$	commanded acceleration
D	drag	$\underline{V}_K$	flight path speed vector
g	acceleration due to gravity	$\underline{V}_W$	wind speed vector
ILS	instrument landing system	W	weight
MLS	microwave landing system	w_W	vertical wind speed
n	load factor		
T_c	commanded thrust	γ	glide path angle
u_W	horizontal wind speed		
$\dot{u}_W$	change in horizontal wind speed	$\underline{\Delta}$	vector of the state error

1. INTRODUCTION

Flight control systems are more or less a conventional tool to improve the aircraft characteristics as well as to achieve a more precise guidance and control. For most applications e. g. all weather ILS approach, standard flight control systems are precise enough. For future 4D-navigation, windshear response alleviation, space reentry vehicle control or geodetic applications e. g. measurement of the earth's surface on board the aircraft for digital mapping, a more precise flight control system is required. And last but not least, it is an attractive scientific task to demonstrate theoretical approaches in realistic flight tests.

The flight control system I like to present here in more detail has been developed on the basis of a very simple description of the scientific task: the flight control system should be as precise as possible in flight path and aerodynamic flow control under the constraints, that passenger comfort and throttle activity are acceptable.

2. FLIGHT CONTROL SYSTEM

The basic principle for the developement of the flight control system was to introduce all the knowledge into the control system design we have, concerning an aircraft's behaviour.

The input signals for the systems are guidance functions for flight path and the aerodynamical flow conditions. In the given example, the aerodynamic flow condition is presented in a conventional manner as a spatial airspeed function [1,2,3]. The flight path is determined as an earth fixed spatial function. A precise flight control system has to include the following control system elements (fig. 1)

- open loop determination of the required actuator displacement (fig. 2)
- open loop calculation of the commanded state vector (fig. 2)
- state vector feed back, including sensors and observers
- integral trim to improve static accuracy and to reduce control column forces in the event of system failures

With a precise open loop control, the state error Δ (fig.1) is neglibly small. In this case it is possible, to design the closed loop for excellent eigenvalue distributions and a low feed back gain.

The main key for the flight testing is the state error Δ, which is a substantial value to estimate the quality of the flight control system.

With the knowledge of the aircraft's non linear equations of motion, we can derive the required deflections of the control surfaces and throttle. This inverse flight mechanical problem cannot be solved in general. A. Redeker [2] has developed a complex quasistationary inverse model of the aircraft longitudinal motion that shows excellent results (fig. 3). To demonstrate typical characteristics of this quasistationary inverse aircraft model, we will calculate the commanded thrust.

$$T_c = W\left[n\,\frac{C_D}{C_L} + n\,\frac{w_W}{V} + \left(1 + n\,\frac{u_W}{V}\right)\gamma + \frac{\dot{u}_W}{g} + \frac{\dot{v}_c}{g}\right] \qquad (1)$$

This thrust equation can be derived from the aircraft forces parallel and perpendicular to the flight path. We get the commanded thrust as a function of the aircraft weight (W), the load factor n, the drag to lift ratio, the flight path angle γ, the ratio of vertical (w_W) and horizontal (u_W) wind including turbulence to airspeed V, the windshear $\dot{u}_W$ and the commanded acceleration $\dot{v}_c$. The aircraft drag D can be written in a typical flightmechanical expression (fig. 3).

$$D = n\,W\,\frac{C_D}{C_L} \qquad (2)$$

To solve the thrust equation (1) with sufficient accuracy, the aircraft parameters C_D/C_L, W, the aircraft motion (n, γ, V) as well as the wind situation w_W , u_W , $\dot{u}_W$ must be measured and identified. The wind vector can be measured on board with sufficient precision [4] (see chapter 4). The aircraft parameter have to be derived as well from theoretical calculations, as from simulations and flight tests. In the newest version of the described flight control systems, the aircraft parameters are calculated by a learning procedure [5]. In principle, the variance of the state error Δ (fig. 1) will be minimized in real time. Without any previous knowledge, the system needs appoximately 2000 seconds for an on line identification of the non linear aircraft parameters. In fig. 4 the required thrust versus airspeed is presented. The better the initial knowledge, the quicker is the learning procedure. This system is something like an adaptive flight control system without any stability problems.

The second major feature of the presented flight control system is the determination of the commanded state vector (fig. 1 and 2). With a complete state vector command the control system reponse can be very quick without overshoot. A simple example of a state vector command is given in fig. 5a for a typical stability augmentation system. The commanded pitch rate may be proportional to the stick force. The relevant state vector elements are pitch attitude Θ and pitch rate q, which can be measured easily by gyros. As for a symmetric flight condition (wings level) pitch attitude is the integral of the pitch rate, the state vector command calculation is very simple. The response of the closed control loop is quick without overshoot (Θ_2 in fig. 5b). In contrast to the demonstrated example, in most conventional stability augmentation systems there is only a pitch attitude but no pitch rate command. This means, that the commanded pitch rate is zero. This conflicting situation results in a sluggisch response and in an overshoot (Θ_1 in fig. 5b).

The total open loop subsystem for actuator command and state vector command is presented in fig. 2. The system will be completed by a conventional state vector feed back and an integral trim system. With a precise open loop control, the difference Δ between commanded and actual state vector will be zero. In this case the feed back gain matrix is only responsible for a sufficient eigenvalue characteristic and for compensation of small calculation errors. To identify the non linear functions of the inverse aircraft model, the state vector error Δ will be reduced in a trial and error procedure from the flight test. With some theoretical knowledge this job had been done in less then ten flight test hours [2].

3. FLIGHT TEST VEHICLES

The identification of the aircraft parameters and the flight testing of the flight control system has been realized with the two test aircraft DO 28 (fig. 6) and DO 128 (fig. 7) owned by the Institute for Flight Guidance and Control (appendix 1). Both aircrafts are equipped with flexible general purpose test equipment.

- sensors
- data acquisition
  calculation
  storage
  operation
- actuators
- control panels and displays

The sensor group contains

- inertial
- position
- aerodynamic flow

systems, that are integrated in a navigation system (fig. 8). All control surfaces and the throttle are controlled by electrical servo motors.

- elevators
  elevator trim
- aileron
- rudder
- fast flaps
- throttle

The elevator trim system is required especially for automatic landings. The data processing systems (fig. 9) consists primarely of

I/O processor including interfaces
maincomputer
terminal
cockpit and operator displays
data recording

In the main computers high level computer languages will be used. In the older DO 28 aircraft with a Norden/DEC 11/34 main computer the programs are written in FORTRAN 77 language. The newer DO 128 aircraft has been equipped with an Aerodata/DEC Mikrovax II main computer and MODULA 2 language. The more transparent structure of MODULA 2 compared to FORTRAN 77 is the major reason for an important reduction of programming and testing time. The hardware location is given in fig. 10. The listing of the hardware is presented in appendix 1.

4. WIND MEASUREMENT

The quality of flight tests can be strongly influenced by disturbances, especially wind and turbulence. Additionally the nonlinear open loop control requires the information of the wind vector (see eq. 1). Parameteridentification without knowledge of the wind disturbance is only sufficient for flights in calm air. Real calm air is very rare in central europe and this limits the efficiency of the flight test tremendously. Therefore both test aircrafts are equipped with an on line realtime wind measurement system [4]. The wind vector $\underline{V}_W$ can be derived from the airspeed vector $\underline{V}(V, \alpha, \beta)$ and the flight path speed vector $\underline{V}_K$ (fig 11)

$$\underline{V}_W = \underline{V}_K - \underline{V} \qquad (3)$$

This type of measurement is based only on kinematics and is therefore independent from aircraft motion. The handicap is the small wind vector as the difference of the large speed vectors. The measurement of the two speed vectors must be very precise. On line calibration is as well a need as complementary filtering [4] and coordinate transformations [6].

An example for an on board wind measurement is the low level jet in fig. 12. Here the on board measurement is compared with a mast anemometer measurement [7] and a low level jet model [8,9].

This type of on board wind and turbulence measurement will be applied beside the flight testing of guidance and control systems for meteorological and environmental (pollution) measurement.

5. FLIGHT TEST RESULTS

Finally, some flight test results shall be demonstrated. Figure 13 shows the high accuracy of the complex digital flight control system in calm air. In a 9 minutes flight period, the maximum altitude deviation was less then 1 meter. The altitude deviation is in the range of the resolution of the barometric altimeter. Figure 14 shows the aircraft response in altitude, airspeed and thrust at the begin of a turn flight in moderate turbulence. An altitude-acquire manouevre shows fig. 15 for strong turbulence. Typical for this test aircraft is the high gust sensitivity of the uncontrolled aircraft due to the low wing load. An older version of the complex flight control system (with a simple open loop control) is shown in fig. 16 in an curved MLS-approach [11] passing a moderate wind shear without any flight path and speed deviation. This MLS-approach has been finished by an automatic landing.

REFRENCES

[1] Redeker, A. A New Control System for Flight in Turbulences and Windshear. 2nd International Symposium for Aviation Safety, Toulouse, November 1986

[2] Redeker, A. Beiträge zur Verbesserung der Flugführungsgenauigkeit von Flugreglern. Dissertation at the Institute for Flight Guidance and Control, TU Braunschweig, 1986

[3] Schänzer, G. Design Criteria for Multi-Loop Flight Control Systems, AGARD, 42nd Guidance and Control Panel Symposium, Brussels, June 1986

[4] Vörsmann, P. An On-Line Realization for Precise Wind Vector Measurements on Board the DO 28 Research Aircraft, 14th Congress of the ICAS, Toulouse, France, September 1984

[5] Haverland, M. Ein lernender Regler für die Flugzeuglängsbewegung. Dissertation at the Institute for Control, TU Braunschweig, 1988

[6] Redeker, A., Vörsmann, P. Precise Vertical Speed Reconstruction Based on Vertical Acceleration and Barometric Altitude, Z. Flugwiss. Weltraumforsch. 9, 1985, pp. 243 - 251

[7] Brunner, D. Untersuchung zur Wind u. Turbulenzstruktur in der planetaren Grenzschicht, DGLR Vortrag Nr. 84 - 100, Hamburg, 1984

[8] Swolinsky, M. Beiträge zur Modellierung von Scherwind für Gefährdungsuntersuchungen, Dissertation at the Institute for Flight Guidance and Control, TU Braunschweig, 1986

[9] Swolinsky, M. Wind Shear Models for Aircraft Hazard Investigation, 2nd International Symposium for Aviation Safety, Toulouse, November 1986

[10] Redeker, A. Real Time Processing and Display, AGARD, Flight Mechanics Panel Symposium, Aircrafts Testing Techniques, Edwards, California, 1988

[11] Schänzer, G. The Influence of Microwave Landing Systems on Guidance and Control, AGARD CP, Guidance and Control Panel, Cheltenham Spa, UK, 1978

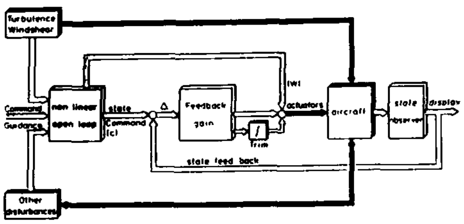


Fig 1: Block diagramm of the control loops

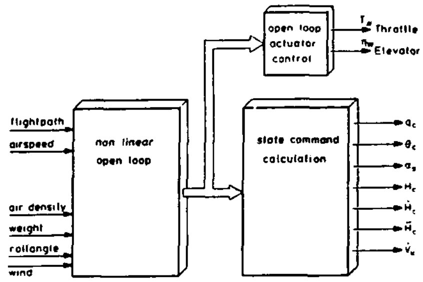


Fig 2: Non linear open loop control state and state command calculation

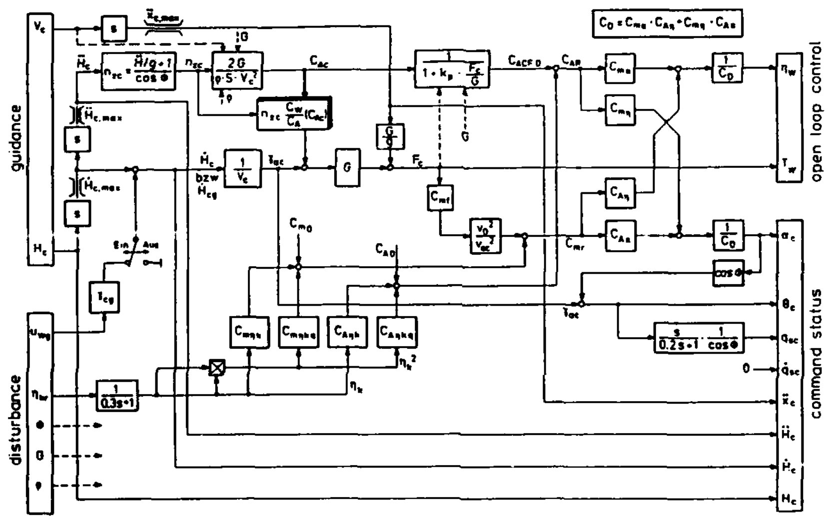
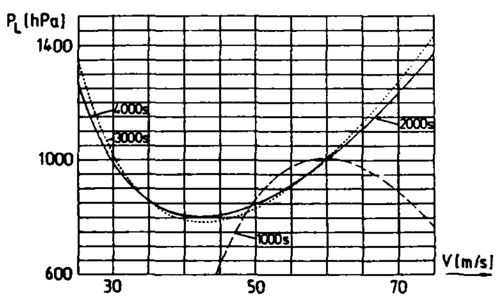


Fig 3: Inverse aircraft model



$$P_L = a_{-2}V^{-2} + a_{-1}V^{-1} + a_0 + a_1 V + a V^2$$

Fig 4: Manifold pressure versus airspeed

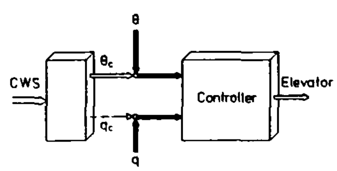


Fig 5a: State vector command

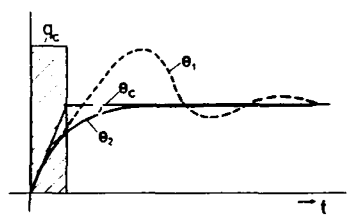


Fig 5b: Pitch response

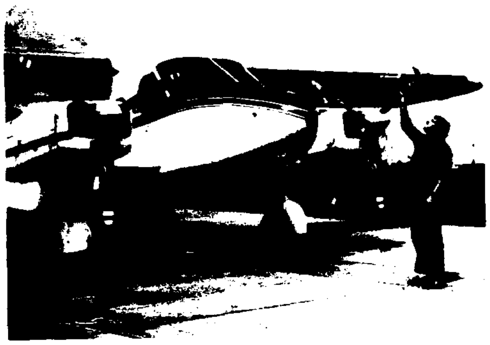


Fig 6: DO 28 Research aircraft

Fig 7: DO 128 Research aircraft

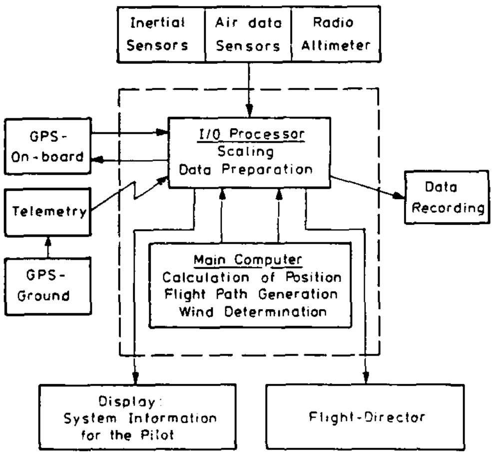


Fig 8: Concept for the DO 128 integrated navigation system

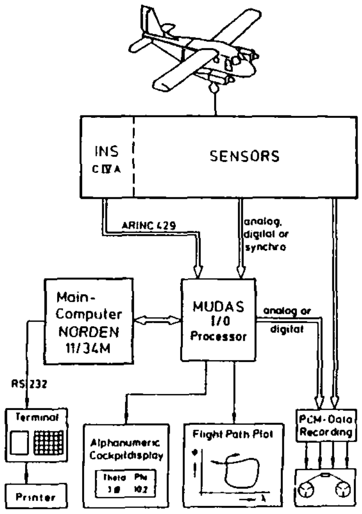


Fig 9: DO 28 Data processing system

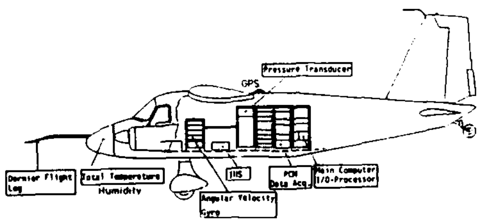


Fig 10: Hardware location in the DO28 aircraft

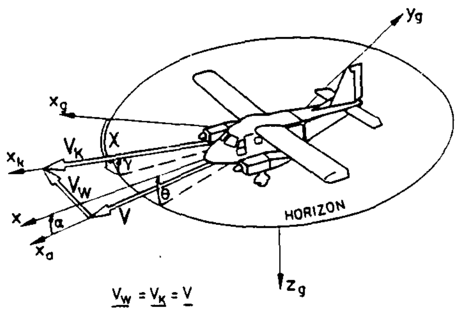


Fig 11: Determination of the wind vector

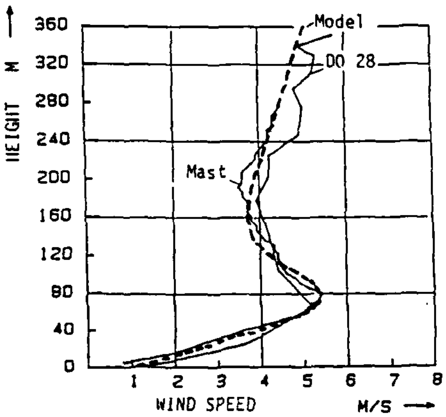


Fig 12: Comparison of aircraft and mast measurement with a low level jet model

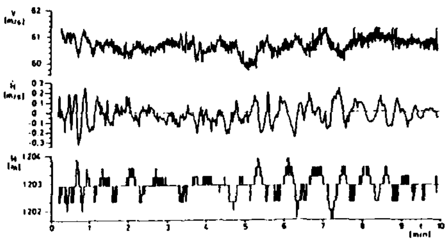


Fig 13: Altitude and speed hold (calm air)

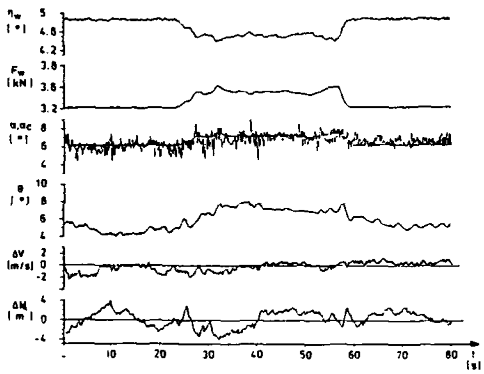


Fig 14: Altitude and speed hold in turn flight (moderate turbulence)

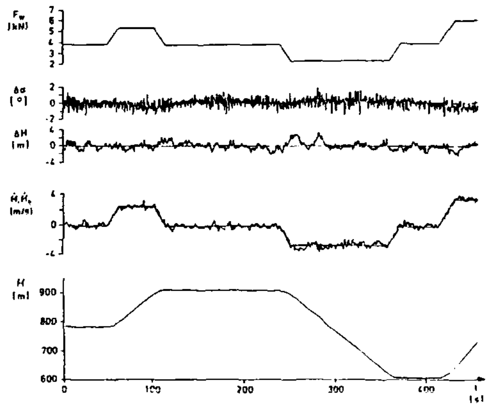


Fig 15: Altitude acquire (strong turbulence)

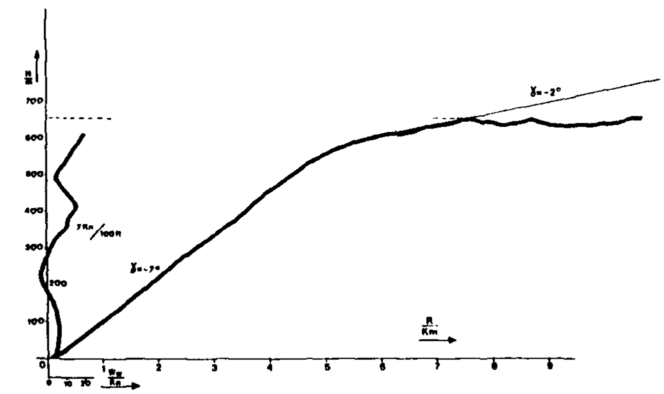


Fig 16: Curved MLS approach (passing a moderate windshear)

APPENDIX 1

Research Aircraft Dornier 128-6 / Call sign D-IBUF

Technical Data and Dimensions

Twin turboprop-engine-powered STOL aircraft, non steerable tailwheel

Dimensions:

Overall Length	11.71m (without noseboom)
Overall Height	3.90m
Wing Span	15.55m
Wing Area	29.00sqm

Weight:

Maximum Take-Off Weight	4350kg
Empty Weight (Incl. Research Equipment)	3080kg
Maximum Fuel (1500l)	1150kg

Engines:

2 x Pratt&Whitney Aircraft of Canada PT6A-110 turbo-prop engine with 400Shp continuous power.
Electrical power supply for experiments c. 3kVA :
28V DC
115V/400Hz AC
220V/ 50Hz AC

Performance:

Max. Operating Altitude	20000ft	(6100m)
Max. Speed	150kts	(278km/h)
Min. Speed	63kts	(117km/h)
Cruise Speed	130kts	
Take-Off-Distance	550m	
Landing Distance	600m	
Endurance (at FL 100)	6.5h	
Location:	DFVLR-Flugabteilung, Braunschweig Flughafen	

Research Tasks

The twin-engined STOL-Aircraft is an universal tool for airborne investigations at the Institute for Flight Guidance and Control and other research groups. The main tasks are:

- meteorological measurement (on-line determination of wind speed and direction, temperature, humidity, measuring of air pollution)
- investigation of the aircraft motion under influence of turbulent wind conditions
- airborne investigation of different sensor- and avionic-systems
- identification of the aerodynamical parameters of the aircraft

Research Equipment

Sensors:

Sensor	Parameter
DORNIER Flight Log	Angle of Attack Angle of Sideslip Pressure Port for Static and Total Pressure
DFVLR 5-Hole-Probe with ROSEMOUNT 1221	Static Pressure Dynamic Pressure Differential Pressure for Angle of Attack and Sideslip
HONEYWELL Lasernav	Pitch Angle and Angular Velocity Bank Angle and Angular Velocity Yaw Angle and Angular Velocity Acceleration in x,y,z-Direction Position, Groundspeed True Track Angle Magnetic Heading Vertical Velocity Inertial Altitude Wind Speed and Direction
PAROSCIENTIFIC Digiquarz Mod. 1023A	Static Pressure
ROSEMOUNT 1221	Dynamic Pressure
ROSEMOUNT 102EJ1BB	Temperature
VAISALA/AERODATA Humicab	Humidity Temperature
SPERRY Radar Altimeter	Height

Computer Systems:

Main Computer: AERODATA-PDP 11/73 / microVAX, ON-Line-Graphic, Possibility for Hard-Copy
Communication Computer: LANGE WL 2001 / IAM VME-Bus Processor

Data-Recording:

- Streamer Tape Recorder 64 channels with 23 Hz sampling rate

Additional Equipment:

- Standard-IFR-Instrumentation with DME (Distance Measuring Equipment) and Radar-Altimeter, VLF-Navigation-System, HF-Communication
- VHS-Video Recording Equipment
- SERCEL-GPS (Satellite-Navigation-System)
- Until the end of 1989 the aircraft will be equipped with actuators and position indicators for elevator, rudder, aileron and throttle. Then the flight testing of autopilot systems will also take place on the Do 128-6.

Research Aircraft Dornier 28D1 / Call sign D-IBSW

Technical Data and Dimensions

Twin piston-engine-powered STOL-Aircraft, non steerable tailwheel

Dimensions:

Overall Length	11.37m (without noseboom)
Overall Height	3.90m
Wing Span	15.50m
Wing Area	28.00sqm

Engines:

2 x Lycoming IGSO-540 A1E, six cylinder aircooled turbocharged piston engine with fuel injection with 380 hp max. continuous power.

Electrical power supply for experiments c. 3kVA :
28V DC,
115V/400Hz AC,
220V/ 50Hz AC

Weight:

Maximum Take-Off Weight	3700kg
Empty Weight (incl. Research Equipment)	2950kg
Maximum Fuel (820l)	600kg

Performance:

Max. Ceiling	24000ft	(7300m)
Max. Speed	177kts	(328km/h)
Min. Speed	56kts	(104km/h)
Cruise Speed	130kts	
Take-Off-Distance	530m	
Landing Distance	600m	
Endurance (at FL 100)	6h	
Location:	DFVLR-Flugabteilung, Braunschweig Flughafen	

Research Tasks

The twin-engined STOL-Aircraft is an universal tool for airborne investigations at the Institute for Flight Guidance and Control, the Special Research Group for Flight Safety and other research groups. The main tasks are:

- meteorological measurement (on-line determination of wind speed and direction, temperature, humidity, measuring of air pollution)
- investigation of the aircraft motion under influence of turbulent wind conditions
- flight testing of autopilot systems (auto-land, direct-lift-control, fly-by-wire, side-stick-control, etc.)
- airborne investigation of different sensor- and avionic-systems
- identification of the aerodynamical parameters of the aircraft

Research Equipment

Sensors:

Sensor	Parameter
DORNIER Flight Log	Angle of Attack Angle of Sideslip Pressure Port for Static and Total Pressure
ROSEMOUNT 102 AU1BZ	Total Temperature
2 NORTRONICS G5ATH7	Pitch Angular Velocity Bank Angular Velocity
SFENA 39-05V1M1	Yaw Angular Velocity
3 DONNER Mod. 4310	Aircraft Fixed Acceleration in x,y,z-Direction
INS CAROUSEL IVA	Pitch Angle Bank Angle Vertical Acceleration Along Track Angle Ground Speed True Track Angle True Heading Angle Wind Speed Wind Direction Position
ROSEMOUNT 1241M4B1	Static Pressure (Altitude)
STATHAM PM69TC	Dynamic Pressure
COLLINS Radar Altimeter AL101	Height

Computer Systems:

Main Computer: NORDEN 11/34 (mil. version of a DEC PDP 11/34) 2 Floppy-Drives, Floating Point Processor
Background Computer: J11, Q-Bus-Processor
Communication Computer: MUDAS , Fa. Dornier (Modular Universal Data Aquisition System)

Data-Recording:

- PCM Data Aquisition System IM 16K3, 32 channels with 92 Hz sampling rate
- Streamer Tape Recorder 64 channels with 23 Hz sampling rate

Additional Equipment:

- Standard-IFR-Instrumentation with RCA Primus 10 DME (Distance Measuring Equipment), ARC 41A VOR
- actuators for elevator, rudder, aileron, throttle
- electrical elevator trimm
- measurement of engine manifold pressure and rotating speed
- VHS-VIDEO recording equipment

ANALYSE DES ESSAIS DE FLOTTEMENT A L'AIDE D'UN SYSTEME DE TELEMESURE

par

Alain Bucharles
Départment Automatique — ONERA/C.E.R.T.

Gérald Lignon et Henri Cassan
Essais en Vol — Division AVION — AEROSPATIALE
316 Route de Bayonne
31060 Toulouse, Cedex 03
France

RESUME : La nécessité d'analyser les résultats de chaque vol avant d'autoriser le point de vol suivant est le principal inconvénient de l'ensemble des procédures d'Essais en Vol de flottement. Elles conduisent toutes à une durée globale d'ouverture du domaine de vol importante et incompatible avec les objectifs de certification des avions. Aussi AEROSPATIALE et le C.E.R.T. ont mis en commun leur expérience pour développer des outils et des méthodes d'analyse en temps réel des essais à l'aide d'un système de télémesure associé à de puissants moyens de calcul.

INTRODUCTION

Les essais en vol de flottement ont pour but :

- l'ouverture du domaine de vol c'est-à-dire la démonstration de l'absence de flottement dans tout le domaine de vol jusqu'à MD/VD.
- la fourniture au Bureau d'Etudes de résultats suffisamment précis pour pouvoir les extrapoler jusqu'à 1,2 VD.

L'ouverture du domaine de vol se fait pas à pas par observation de l'évolution de l'amortissement de chaque mode avec la vitesse pour diverses configurations de l'avion. L'autorisation de procéder à l'essai suivant (augmentation de vitesse) n'est donnée que si cette évolution ne présente aucun risque d'instabilité.

Autrement dit l'exploitation complète d'un vol doit être terminée et analysée avant de procéder au vol suivant. Ce qui peut conduire, suivant la complexité des problèmes rencontrés, à des délais importants.

Or la recherche d'une plus grande compétivité impose de produire des appareils de plus en plus performants à un coût de production moindre. D'où la nécessité de réduire la durée de développement et d'essai en vol. Pour l'A320 l'objectif était de certifier l'avion seulement onze mois après le premier vol.

Pour mener à bien un projet aussi ambitieux pour un avion entièrement nouveau, AEROSPATIALE s'est dotée d'un système de traitement en temps réel par télémesure (pouvant répondre aux besoins des diverses disciplines des essais en vol) et en étroite collaboration avec le département Automatique de l'ONERA/CERT a développé de nouvelles méthodes d'exploitation des essais.

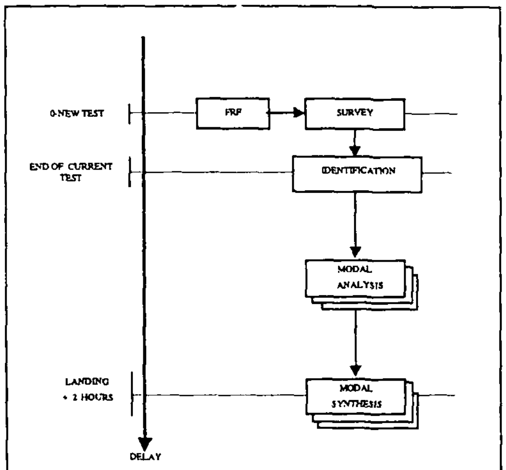


Fig.1-Objectifs pour l'analyse des essais de flottement

Ces méthodes, concues comme aide à la décision, automatisent le traitement des données numériques issues du message télémesure. Elles permettent ainsi au spécialiste de se consacrer pleinement à la conduite de l'essai et à l'analyse et l'interprétation des résultats. L'objectif visé est de pouvoir réaliser deux points du domaine dans le même vol et d'être en mesure de donner la "clearance" pour le vol suivant dans un délai de deux heures après l'atterrissage de l'avion (fig. 1) de manière à pouvoir effectuer si nécessaire deux vols dans la même journée.

Elles consistent essentiellement en :

- la surveillance en temps réel des paramètres les plus significatifs.

- l'analyse et la synthèse modale en temps légèrement différé sur l'ensemble des paramètres (50).

ESSAI EN VOL DE FLOTTEMENT

1. PRINCIPE D'ESSAI

Les essais de flottement sont réalisés aux différents points du domaine de vol par excitation de la structure de l'avion. La mesure simultanée des forces d'excitation et des réponses accélérométriques de la structure permet de calculer les fonctions de transfert $H = \Phi_{es} / \Phi_{ee}$ d'où l'on extrait les paramètres modaux F (fréquence) et α (amortissement réduit).

2. MOYENS D'ESSAI EMBARQUES

Les moyens d'excitation utilisés sont :

- des volets aérodynamiques oscillants (tip-vane) installés à chaque extrémité de voilure, pilotables en fréquence (choix Fmini et Fmaxi), en mode (symétrique ou antisymétrique), en vitesse de balayage. Ils permettent de réaliser des excitations harmoniques.
- des impulseurs à poudre montés en extrémité de plan horizontal et de dérive. Ils génèrent des excitations impulsionnelles.

Les réponses de l'avion sont mesurées par des accéléromètres disposés sur toutes les parties de l'avion comme le montre la figure 2.

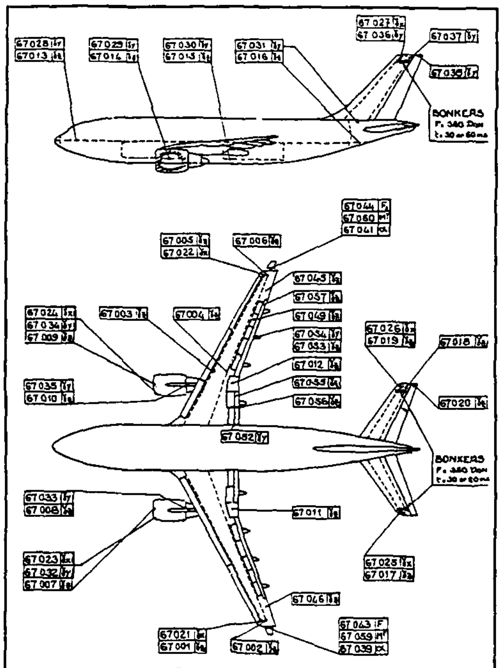


Fig 2 - Installation d'essais en vol

3. Procédure de l'essai

Les sollicitations couvrent la plage de fréquence 0/30 Hz. Chaque excitation harmonique est limitée à une bande de fréquence de deux octaves de manière que la dynamique des accélérations mesurées soit suffisamment faible. La vitesse de balayage utilisée de 30 secondes par octave est un bon compromis entre la vitesse nécessaire à une bonne excitation, le nombre de balayages et la durée globale de l'essai.

Pour chaque gamme de fréquence les essais sont répétés deux fois : une première fois en symétrique puis en antisymétrique.

Les tirs d'impulseurs sur la partie arrière sont faits en symétrique et en antisymétrique sur l'empennage horizontal et en latéral sur l'empennage vertical.

Donc un point d'essai nécessite : (Fig. 3)

- deux balayages "basses fréquences" (symétrique et antisymétrique).
- deux balayages "haute fréquence" (symétrique et antisymetrique).
- deux tirs d'impulseurs empennage horizontal (symétrique et antisymétrique)
- un tir d'impulseur empennage vertical.

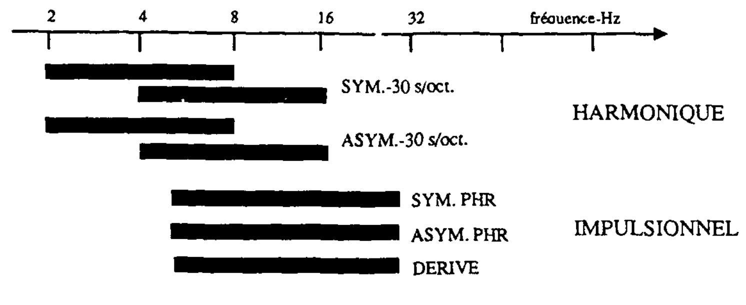


Fig.3-Essai en vol de flottement "type"

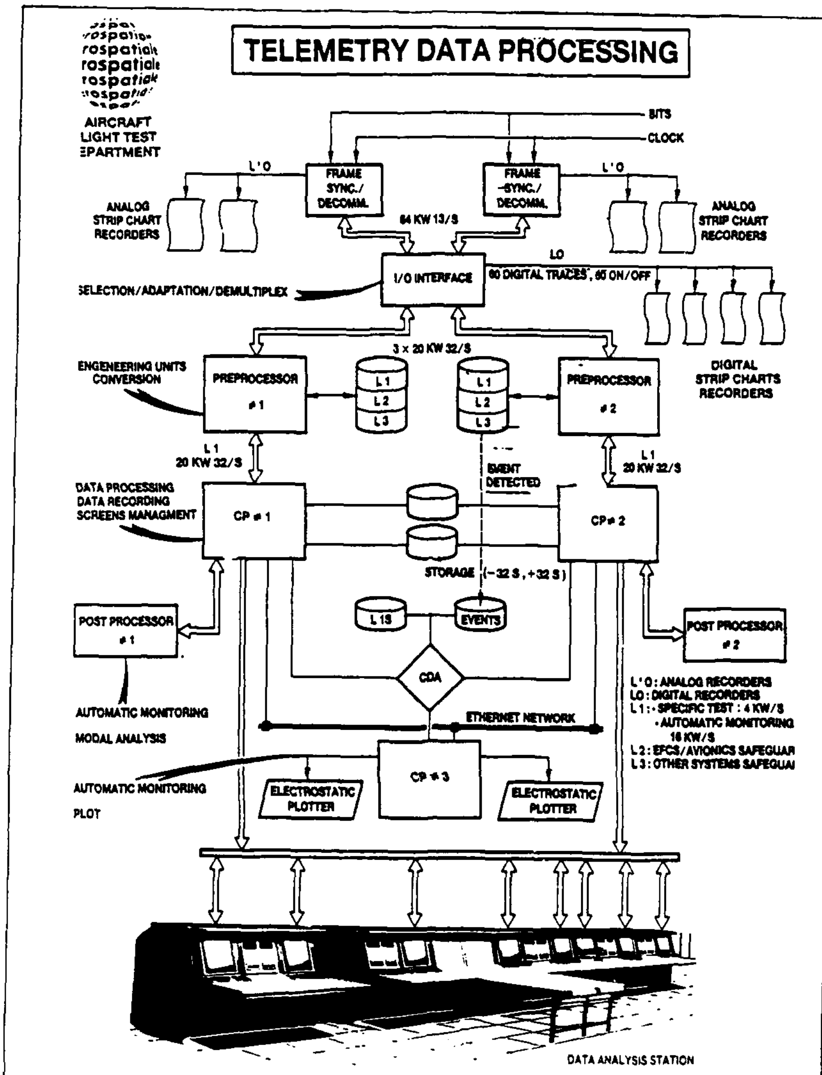


Fig 4 - Organisation générale télémesure

TELEMESURE

1. DESCRIPTION GENERALE

Le message télémesure (PCM de 64 Kmots de 13 bits) est émis par l'avion à l'aide de deux antennes : une sous le nez de l'avion, l'autre divisée en deux demi-antennes de part et d'autre de la dérive.

La réception se fait soit directement à Toulouse (station principale) soit à Bordeaux ou à Saint Nazaire (stations télécommandées depuis Toulouse). Dans ce dernier cas la liaison avec la station principale est assurée par satellite.

Le centre de traitement du message reçu est organisé autour d'un calculateur central et de deux calculateurs vectoriels (pré et post processeur). Chaque calculateur est doublé pour pallier à la panne éventuelle d'un ou plusieurs éléments.

La salle d'exploitation comprend six postes de travail interchangeables plus le poste du chef d'écoute. La figure 4 montre la description de l'ensemble.

2. ORGANISATION "FLOTTEMENT"

Le message télémesure spécifique au flottement contient :

- les paramètres généraux avion (Vc-Mach-masse-centrage-réservoirs-...)

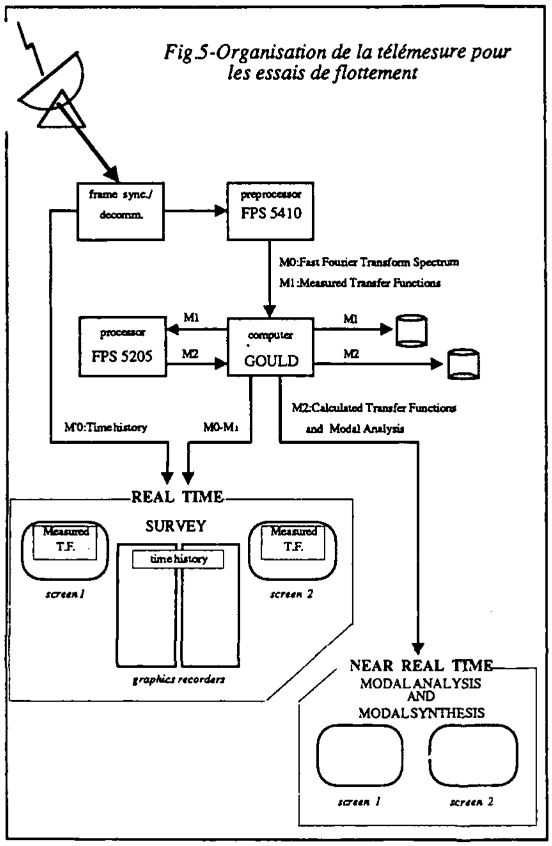


Fig 5-Organisation de la télémesure pour les essais de flottement

- les forces d'excitation et les réponses avion à la cadence de 128 p.p.s. (prélèvements par seconde) pour le calcul des fonctions de transfert.
- une force d'excitation et quelques réponses choisies avant les essais à la cadence de 512 p.p.s. pour le suivi en fonction du temps.

La figure 5 décrit l'organisation de la station pour les essais de flottement. Cette organisation est faite autour de deux postes correspondant aux deux objectifs de surveillance en temps réel de l'essai et d'analyse et synthèse en temps très légèrement différé.

Le poste temps réel reçoit les messages :
- M'0 directement en sortie du décommutateur pour visualisation sur les enregistreurs graphiques des évolutions en fonction du temps de 16 paramètres à 512 p.p.s.
- M0 élaboré par le préprocesseur et contenant 8 spectres en fréquences (4 maxi par écran) pour la surveillance de l'avion au cours des augmentations de vitesse.
- M1 élaboré également par le préprocesseur et contenant 8 fonctions de transfert (4 maxi par écran) pour la surveillance de l'essai proprement dit.

Le poste temps différé reçoit le message M2 élaboré par le postprocesseur et contenant les fonctions de transfert mesurées et calculées ainsi que les résultats de l'analyse modale.

Les messages M1 et M2 sont également enregistrés sur disque pour traitement éventuel ultérieur sur le site télémesure lui-même ou sur les ordinateurs du centre de calcul "sol".

ESTIMATION DES REPONSES EN FREQUENCE

L'ensemble des signaux mesurés et transmis au sol par l'intermédiaire de la télémesure font l'objet d'un traitement numérique, schématisé figure 6, qui permet de dégager pour chacun d'entre eux un sous ensemble de modes. A partir de ces informations et de la connaissance de l'avion et de la structure, le spécialiste est alors à même d'effectuer la synthèse modale pour le point de vol concerné.

Nous allons détailler par la suite les différentes étapes du traitement.

1. Mise en forme des signaux

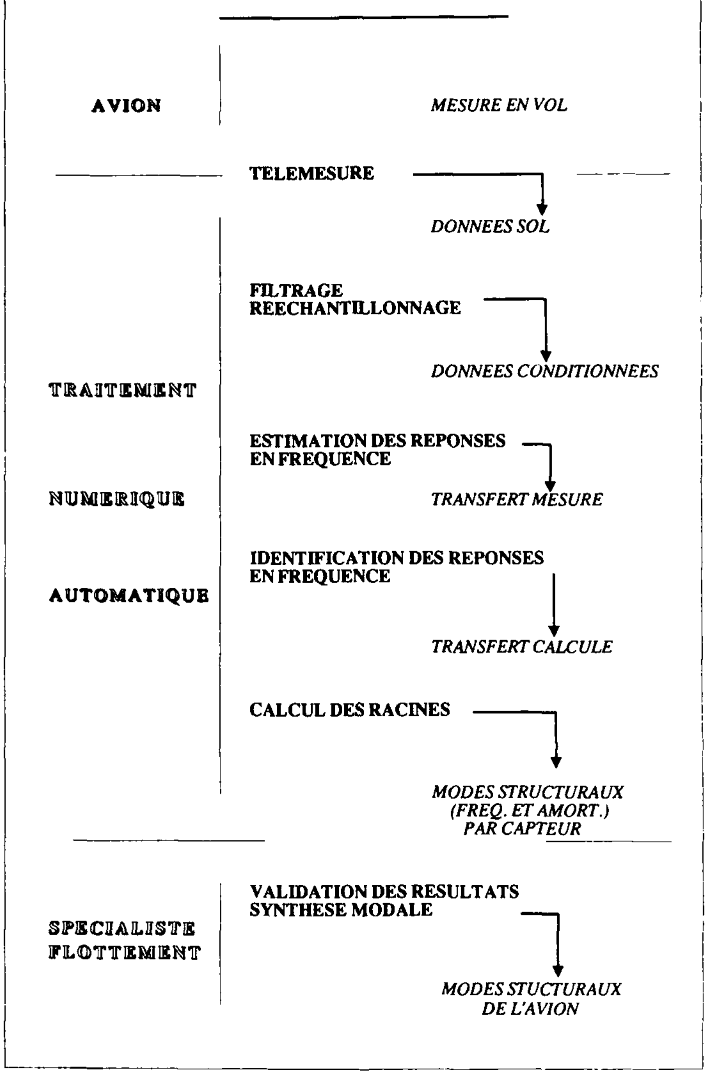
Les signaux reçus au sol sont échantillonnés à la cadence de 128 p.p.s. Or l'information intéressante se situe dans la bande de fréquence de l'excitation qui s'étend de 2 à 8 Hz ou bien de 4 à 16 Hz selon le type d'essai, c'est-à-dire bien en deçà de la demi-fréquence d'échantillonnage. Ils apparaissent donc suréchantillonnés et de manière à réduire les temps de calcul il sont reéchantillonnés, après filtrage, à 32 p.p.s. (2-8 Hz) ou 64 p.p.s. (4-16 Hz).

Le filtrage nécessaire avant reéchantillonnage doit répondre à certaines contraintes :

- ne pas introduire de déformation importante de la bande utile,
- couper suffisamment pour éviter des repliements de fréquence gênants dans la gamme de travail.

Le choix s'est porté sur un filtre numérique de BUTTER WORTH d'ordre 6.

Fig 6 - principe de traitement des essais de flottement



2. ESTIMATION DE LA REPONSE EN FREQUENCE

A ce stade, l'information se présente sous la forme de signaux temporels échantillonnés et il serait tout à fait possible de la traiter telle quelle pour obtenir les caractéristiques modales de la structure moyennant la manipulation d'un volume de données important conduisant à des temps de calcul élevés. C'est la raison principale pour laquelle nous avons préféré réaliser une identification dans le domaine fréquentiel. Un autre avantage de ce type d'identification est qu'il est possible de ne l'effectuer que dans une bande de fréquence limitée, permettant de s'affranchir de certains modes (modes rigides avion). Cependant elle conduit à une étape supplémentaire : l'estimation de la réponse en fréquence entre l'excitation et les réponses de la structure supposée linéaire. Si e(t) désigne l'excitation et s(t) une réponse, un estimateur de la réponse en fréquence est obtenu en faisant le rapport du spectre croisé excitation-réponse et du spectre de puissance de l'excitation.

$$H(\omega) = \Phi_{es}(\omega) / \Phi_{ee}(\omega)$$

Le problème est donc ramené à celui de l'estimation des spectres Φ_{es} et Φ_{ee} pour laquelle différentes techniques sont envisageables. La méthode utilisée est la méthode de WELCH : les K séquences temporelles de M points issues du fractionnement du signal de départ sont multipliées par *une fenêtre de HANNING puis transformées* par Fast Fourier Transform. Le spectre est calculé en moyennant les périodogrammes ainsi obtenus. Dans le cas d'un avion, le faible amortissement des modes et leur proximité éventuelle impose un choix judicieux de K et M pour assurer un bon lissage et une résolution suffisante (de l'ordre de 0,05 Hz).

Outre l'estimation de la réponse en fréquence, l'analyse spectrale permet le calcul de la fonction de cohérence $C(\omega)$

$$C(\omega) = [\Phi_{es}(\omega)]^2 / [\Phi_{ee}(\omega)] * [\Phi_{ss}(\omega)]$$

La cohérence constitue une mesure fréquentielle du rapport signal/bruit. Elle fournit donc une mesure de la qualité de l'estimation de la réponse en fréquence pour chaque fréquence et peut être utilisée pour introduire des pondérations sur les données lors de l'identification.

IDENTIFICATION DES REPONSES EN FREQUENCE

A la suite de l'analyse spectrale la réponse en fréquence d'un capteur quelconque est définie sur un ensemble discret de fréquences sous forme partie réelle/partie imaginaire. Pour avoir accés aux caractéristiques *modales, on va* en rechercher un modèle fraction rationnelle $F(p) = N(p) / D(p)$, les racines de $D(p)$ fournissant ultérieurement les modes. On se restreint bien sûr à la plage couverte par l'excitation. Le problème posé est double : il faut d'une part choisir les degrés de $N(p)$ et $D(p)$, d'autre part, ces degrés étant fixés, identifier les paramètres de $N(p)$ et $D(p)$.

1. IDENTIFICATION PAR UN TRANSFERT D'ORDRE FIXE

On s'intéresse à une liaison entrée/sortie particulière et soit $\{H(\omega_k), k=1,M\}$ la réponse en fréquence correspondante. On cherche à représenter cette liaison par une fonction de transfert fraction rationnelle

$$F(p) = \frac{N(p)}{D(p)} = \frac{a_0+a_1p+\ldots\ldots a_mp^m}{1+b_1p+\ldots\ldots b_np^n} \qquad m \leq n$$

Aux points d'échantillonnage, $F(j\omega_i)$ doit être aussi proche que possible de $H(\omega_i)$. La qualité de l'approximation est donc directement liée à l'écart entre $F(j\omega_i)$ et $H(\omega_i)$. Ceci nous conduit naturellement à prendre comme critère d'identification la somme des carrés des modules des écarts sur *l'ensemble des fréquences*.

$$J = \Sigma \left\| \varepsilon(\omega_k) \right\|^2 = \Sigma \left\| H(\omega_k) - F(j\omega_k) \right\|^2$$

soit
$$J = \Sigma \left\| H(\omega_k) - N(j\omega_k)/D(j\omega_k) \right\|^2$$

$$J = \Sigma \left\| H(\omega_k) - \frac{a_0+a_1j\omega_k+\ldots\ldots+a_m(j\omega_k)^m}{1+b_1j\omega_k+\ldots\ldots+b_n(j\omega_k)^n} \right\|^2$$

On est amené à rechercher l'ensemble des paramètres $(a_o, \ldots, a_m, b_o, \ldots b_m)$ qui minimisent le critère J. Le problème ainsi posé est non linéaire par rapport aux paramètres (b_i), mais il est toutefois possible de le résoudre simplement par moindres carrés *de manière itérative*. Cette résolution consiste à minimiser à l'itération l le critère J_l.

$$J_l = \Sigma \frac{1}{\left\| D_{l-1}(j\omega_k) \right\|^2} \left\| D(j\omega_k) H(j\omega_k) - N(j\omega_k) \right\|^2$$

D_{l-1} étant le *dénominateur calculé* avec les paramètres obtenus à l'itération (l-1). Si l'on pose:

$$D(j\omega_k) = DR(\omega_k) + jDI(\omega_k)$$

$$N(j\omega_k) = NR(\omega_k) + jNI(\omega_k)$$

$$H(\omega_k) = HR(\omega_k) + jHI(\omega_k)$$

La *minimisation* du critère J_l conduit à résoudre le double système d'équations suivant :

$$\begin{cases} \dfrac{1}{\left\| D_{l-1}(j\omega_k) \right\|} [DR(\omega_k) HR(\omega_k) - DI(\omega_k) HI(\omega_k) - NR(\omega_k)] = 0 \\[2ex] \dfrac{1}{\left\| D_{l-1}(j\omega_k) \right\|} [DR(\omega_k) HI(\omega_k) + DI(\omega_k) HR(\omega_k) - NI(\omega_k)] = 0 \end{cases}$$

C'est un système de 2M équations à (m+n+1) inconnues de la forme $Y = XA$

où A désigne le vecteur des paramètres. La solution est donnée par

$$A = (X^T X)^{-1} X^T Y$$

Cette procédure de résolution nécessite l'initialisation du dénominateur. En l'absence de connaissance a priori, on peut choisir $D = 1$. La procédure itérative est arrêtée dès que le critère n'évolue plus. A la convergence on a

$$D_l = D_{l-1}$$

de sorte que le critère minimisé J_1 est bien équivalent à J.

2. CALCUL AUTOMATIQUE DE L'ORDRE DE LA FONCTION DE TRANSFERT

Le calcul structural permet d'avoir une idée assez précise du nombre de modes présents dans une gamme de fréquence donnée. Toutefois, chaque capteur n'est sensible qu'à un nombre limité de modes en fonction de sa position sur la cellule ; il se pose donc le problème du choix optimal des degrés du numérateur et du dénominateur de la fonction de transfert devant être justifiée à chaque réponse en fréquence.

Le degré du numérateur est systématiquement pris égal à celui du dénominateur : on se réserve ainsi le plus grand nombre possible de degrés de liberté.

La détermination de l'ordre du modèle, a priori différent pour chaque réponse en fréquence, n'est pas un problème simple. La théorie fournit bien tout un ensemble de méthodes délivrant directement cette grandeur, mais ces méthodes, si elles fonctionnent bien sur des données de simulation, ne sont pas satisfaisantes pour l'exploitation de signaux réels du fait des bruits (parasites, perturbations atmosphériques, effet mach, ...) et parce que, surtout, il n'existe pas de modèle exact du système à représenter. On va plutôt essayer de définir une recherche intelligente de l'ordre du modèle qui conduise finalement à un modèle physiquement acceptable, composé d'une somme de systèmes du second ordre résonnants.

L'heuristique mise au point consiste à démarrer l'identification à ordre élevé et à faire décroître par pas de deux l'ordre jusqu'à l'obtention d'une solution satisfaisante. Ne sont pris en considération que les points dont la cohérence est supérieure à 0,6. La réduction de l'ordre d'identification est obtenue par élimination successive :

- des modes réels
- des modes instables
- des modes causant la plus faible dégradation du critère
- des modes doubles.

A chaque étape d'élimination, le dénominateur est initialisé à l'aide des modes non éliminés à l'ordre supérieur.

La procédure est arrêtée lorsqu'il n'est plus possible d'éliminer de mode. Les fréquences et amortissements des modes structuraux sont alors extraits des racines du dénominateur.

EXPLOITATION D'UN ESSAI

Les essais en vol de flottement de l'AIRBUS A 320 ont constitué la première utilisation du système complet. Nous allons nous servir de ces essais pour illustrer les différentes procédures utilisées. Ces procédures qui permettent la surveillance en temps réel de l'essai et l'élaboration des résultats finaux à partir des résultats élémentaires calculés automatiquement ont été conçues de façon à être le plus interactives et conviviales possible :

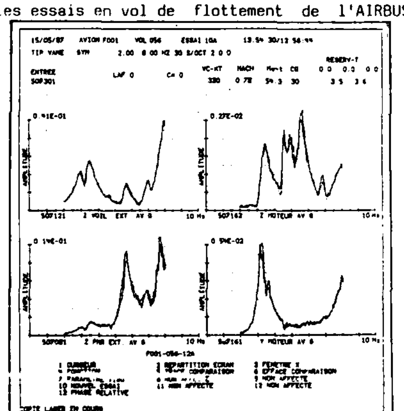


Fig 7-Présentation suivi temps réel

* les programmes et les différentes fonctions à l'intérieur d'un programme apparaissent dans des menus dont la disposition à l'écran rappelle la position de la touche sur le clavier permettant de les activer.

* toutes les instructions nécessaires à l'exécution de diverses tâches sont entrées en mode conversationnel.

1. SUIVI TEMPS REEL

* tracés en fonction du temps. Ils constituent l'aide la plus efficace pour la détection d'anomalies comme la panne d'un capteur, la présence de turbulence,... En un mot ils permettent de valider l'essai. Ils permettent également de visualiser les niveaux vibratoires obtenus ainsi que leurs évolutions.

* fonction de transfert. L'opérateur a la possibilité de présenter 1, 2 ou 4 fonctions de transfert par écran sous diverses formes : amplitude, phase, cohérence, diagramme de Nyquist... La possibilité de superposer à l'essai en cours les fonctions de transfert obtenues à l'essai précédent permet de porter un jugement qualitatif sur l'évolution des modes de l'avion. De plus un curseur sur l'axe des fréquences permet d'afficher pour chaque mode les valeurs d'amplitude, phase, cohérence... Le top de fin d'essai donné par l'opérateur déclenche le calcul d'identification de chaque fonction de transfert et des paramètres modaux associés.

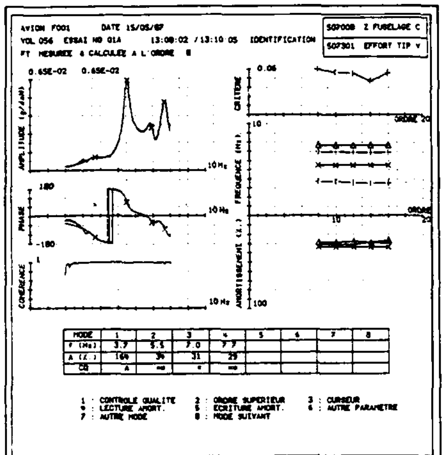


Fig 8- Présentation analyse modale

2. ANALYSE MODALE

Le premier écran permet de choisir le paramètre (réponse avion) que l'on veut examiner.

Apparaît ensuite l'écran analyse (fig. 8) sur lequel on trouve :

- les fonctions de tranfert mesurée et calculée à l'ordre optimum.
- la fonction de cohérence
- l'évolution du critère d'identification
- les évolutions en fonction de l'ordre d'identification des fréquences et des amortissements retenus
- un tableau des valeurs fréquences et amortissements.

A partir de toutes ces informations les spécialistes flottement valident l'analyse et attribuent à chaque mode un critère de qualité : A annulation du mode, 0 annulation de l'amortissement, 1 ou 2.

3. Synthèse d'un essai

L'ensemble des résultats de l'analyse d'un essai sont rassemblés dans un tableau (Fig. 9) où chaque colonne est attribuée à un paramètre. Un algorithme permet de trier les fréquences par ordre croissant et d'afficher un mode par ligne. La dernière colonne (RESULTAT) donne sur chaque ligne la synthèse des résultats calculée comme suit :

- fréquence $F = 1/N \, \Sigma \, F_i$
- amortissement $A = \dfrac{\Sigma A_i \; CQ_i}{\Sigma \, CQ_i}$

avec CQ = critère de qualité = 0,1 ou 2.

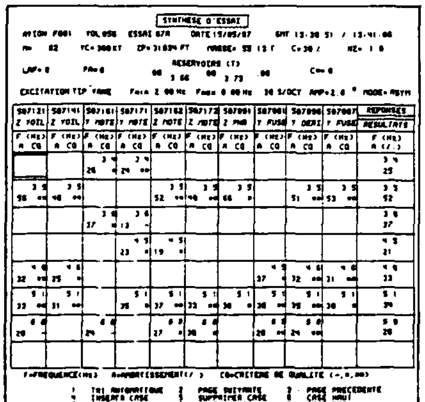


Fig 9 - Présentation synthèse d'un essai

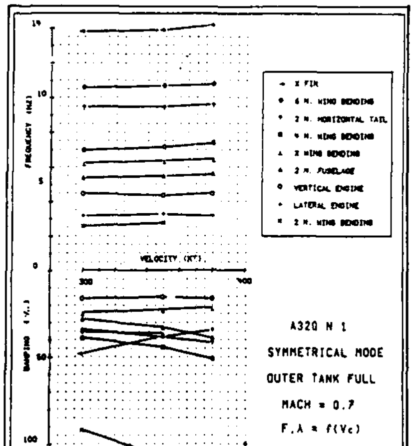


Fig 10 - présentation synthèse des essais

4. Synthèse des essais (Fig. 10)

Ce programme permet de tracer à partir des résultats de la synthèse d'un essai les évolutions des fréquences et amortissements en fonction de la vitesse de l'avion, de ses différentes configurations et des conditions de vol.

BILAN DE L'UTILISATION SUR A320

La disponibilité du système a été totale durant toute la campagne d'essai puisqu'il n'a à aucun moment retardé un vol.

La surveillance temps réel avec la possibilité d'intervention immédiate auprès de l'équipage s'est révélée très complémentaire du rôle de l'ingénieur navigant d'essai. Ceci a permis le déroulement des essais avec une sécurité accrue et de faire l'économie de vols qui se seraient révélés inexploitables (turbulences, pannes capteurs...).

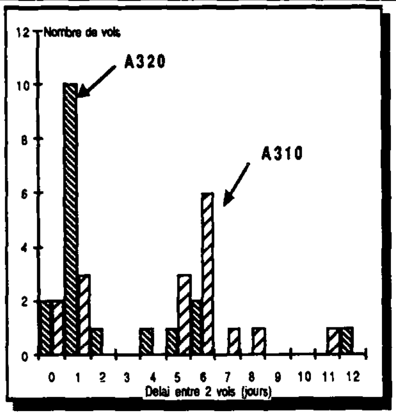


Fig 11- cadence des vols flottement A320/A310

La comparaison avec les essais de flottement de l'A310 fait ressortir les faits suivants :

- un nombre total de vols identique pour les deux avions malgré une complexité beaucoup plus grande de l'A320 (commandes de vol électriques, "load alleviation function").

- une augmentation de la cadence des vols. Cette cadence qui était en moyenne de 1 vol par semaine sur l'A310 a été ramenée à 1 vol parjour sur l'A320.

(Fig 11 ci-contre)

- une facilité accrue de la recherche des paramères les plus représentatifs du comportement de la structure.

- une plus grande homogénéïté et précision des résultats d'amortissement.

CONCLUSION

L'outil développé pour l'exploitation en temps réel des essais en vol s'est révélé performant et bien adapté aussi bien en ce qui concerne le traitement automatique des données que dans l'aide apportée à l'élaboration des résultats et à la compréhension du comportement en flottement de l'avion.

Il a ainsi contribué à obtenir la certification de l'A320 dans les délais impartis.

L'augmentation de cadence de vols qu'il a autorisé ne s'est faite ni au détriment de la sécurité, ni à celui de la qualité des résultats.

Aussi pour l'avenir nous continuons à améliorer nos outils :

- développement de moyens d'excitation encore plus performants et adaptés à tous types d'avion.

- intégration d'une identification multi-réponses pour faciliter la synthèse d'un essai.

Ceci permettra sans aucun doute à AEROSPATIALE de rester compétitive pour les projets futurs.

BIBLIOGRAPHIE

G. HARDIER

"Recherche du modèle aérolastique d'un avion à partir de données de vol"
Thèse de Docteur Ingénieur ENSAE Mai 1984

M. LABARRERE - J.P. KRIEF - B. GIMONET

"Le filtrage et ses applications"
Cepadues édition 1978

M. LABARRERE - A. BUCHARLES

"Analyse modale des essais en vol de flottement"
Rapport DERA Décembre 1984

A. BUCHARLES

"Exploitation par télémesure des essais de flottement de l'A320"
Rapport DERA Février 1986

IEEE Press 1979

"Programs for Digital Signal Processing"

Y. NEGRE - D. LAFOURCADE

"Digital telemetry system for real time analysis of AIRBUS A320 flight test results"

H. CASSAN - G. LIGNON

"A320 N°1 - Flottement - Synthèse de l'exploitation des essais en vol de flottement"
Note interne AEROSPATIALE

H. CASSAN

"A310 N°162 - Synthèse des travaux d'exploitation par DEV des essais en vol de flottement"
Note interne AEROSPATIALE

THE AIR FORCE FLIGHT TEST CENTER

FLIGHT TEST SAFETY PROGRAM

KELLY J. ADAMS, GM-13, SYSTEM SAFETY ENGINEER
AFFTC/SES
EDWARDS AFB CALIFORNIA 93523-5000

MARK R. CRAWFORD, GM-13, PERFORMANCE AND FLYING QUALITIES ENGINEER
6520 TEST GROUP/ENF
EDWARDS AFB CALIFORNIA 93523-5000 USA

SUMMARY:

The success of the AFFTC flight test safety program is based on :

A. Careful attention to the safety planning aspects of testing by each program manager, with thorough review of any applicable formal safety analyses, investigation of more experienced personnel in the subject of test and review of past programs of similar nature for successful and faulty safety planning.

B. Management review of the safety aspects, endorsement and approval of every new and modified test program.

C. The establishment of a staff system safety division which facilitates an independent review of every new and modified test program. For a small increase in manpower a significant increase in safety awareness has been achieved. The safety planning is still primarily the program managers responsibility. SES facilitates the review, documentation, and approval of the program managers safety planning.

D. Flexibility in the safety documentation when programmatic changes occur.

E. Attention to test disciplines of control room procedures, conduct and communication.

F. The presence of a Unit System Safety Officer at each project to review the safety documentation for completeness and accuracy.

GENERAL:

The Air Force Flight Test Center (AFFTC) currently employs a flight test safety program which uses the principles of system safety in both the planning and conduct of all flight test programs. This program attempts to achieve the optimum degree of safety within the constraints of test effectiveness, time and cost attained through specific application of system safety management and engineering principles whereby hazards are identified and risk minimized throughout all phases of the test program. The Air Force development test safety record has improved dramatically over the years. More test hours were flown in 1986 and 1987 at Edwards than in any previous year, without the loss of a test aircraft. There are two basic reasons for the impact in safety: technology and management procedures. Technology in the form of telemetry gives test personnel the ability to monitor critical parameters in real time. But monitoring is not enough. The system must be designed to minimize recognition time, to identify the proper corrective action, and to initiate the action. Recognition time is minimized by prominently displaying limit values of critical parameters. The proper corrective actions must be defined in advance with the test director given the responsibility for notifying the pilot immediately. More important to the test program is the up-front planning process.

Several years ago, after several accidents and near accidents, a decision was made at the AFFTC to establish a separate flight test safety organization. The objective was to create a small organization with some degree of independence from the test managers. The organization was and is staffed by experienced pilots and engineers who are on rotational assignments. They have current experience and have a guaranteed "return" ticket to their parent functional organization. Civilian engineers are given a temporary promotion. The combination of a rotational assignment and temporary promotion attracts highly qualified individuals.

Every test program manager is responsible for the safety planning for his project. A thorough review of past programs of similar nature, any formally prepared contractor system safety analyses and investigation of more experienced personnel in the subject test is the responsibility of the program manager. Every test program undergoes a rigorous safety review by project personnel and senior supervisors which is chaired by people from the safety organization. The review system brings to bear all expertise, government and contractor. The review process ensures that critical conditions are approached in small steps. The safety track record is significantly improved in comparison with the past because of a concerted effort to consider the entire system. With todays complex aircraft, there is the potential for interaction among subsystems. A systems approach is taken during the safety review by including people from a variety of test disciplines in the review process. As an example, propulsion and flying qualities experts are included in the review of gun fire tests. Secondly, people who have been involved in tests of a given type (i.e., flutter, high angle of attack) on a wide variety of aircraft are a part of the review process.

Tests are categorized as low risk, medium risk, or hazardous based on the severity of potential hazards and probability of occurrence. Examples of tests which have demonstrated higher than normal risk include first flights, flight envelope expansion, flutter tests, high angle of attack testing, rejected takeoffs, and tests with explosive warheads. Minimizing procedures to prevent a mishap from occurring or to reduce the consequences of a mishap are developed for each test hazard. All tests are thoroughly reviewed by the senior staff and Flight Test Center Commander prior to accomplishment.

HISTORY:

In March of 1978 during a gunfire test on the A-10 aircraft, both engines flamed out from gun gas ingestion. Due to insufficient altitude the engines were not able to be restarted, the pilot ejected safely and the A-10 was destroyed. The ensuing mishap investigation made several important findings. First there were several unrecognized hazards, second, the associated minimizing procedures had not been established and third, the command authority had not been advised of the risk involved. In summary, a perfectly good airplane was lost because the test had been conducted at an altitude too low for engine restart when the altitude was not critical to the test. Soon thereafter the AFFTC Commander directed the establishment of the system safety division.

AUTHORITY AND CHARTER:

AFFTC REGULATION (AFFTCR) 127-3, Safety Planning for AFFTC Tests, contains the authority and charter for the AFFTC System Safety Division (SES). AFFTCR 127-3 outlines the procedures and responsibilities for all involved agencies for safety planning at the AFFTC. This includes reviewing the safety aspects of testing, identification of the hazards involved, establishment of applicable minimizing procedures and corrective actions, an assignment of a risk level and a final review and approval by the AFFTC commander. SES is a small liaison staff organization and consists of four people: 1) the division chief, who is permanently assigned and is a rated crewmember, 2) a system safety officer, who is temporarily assigned and is a rated crewmember, and 3) two engineering representatives who are temporarily assigned and are selected from the engineering community and are temporarily promoted to a management level. This is not a "classical" System Safety Office but rather a flight test safety office which uses the principles of system safety. The office personnel primarily provide guidance to project personnel, provide a safety database (lessons learned), chair safety review boards and inform upper management of risks. In summary, for very little increase in manpower a more thorough level of safety planning has been achieved. Safety planning is still the primary responsibility of the program manager, however SES facilitates a more disciplined and detailed review, documentation, and approval process.

Additionally one or more Unit Systems Safety Officers (USSOs) are located at each Test Organization or specific engineering discipline. These USSOs ensure that safety documentation is reviewed and signed prior to the safety review board(SRB), after the SRB has convened, and all subsequent safety documentation amendments are complete and accurate. Additionally the USSOs in each organization maintain a book or log of their organizations safety documentation and provide a common documentation and storage system.

RESPONS BILITIES

The responsibilities of SES are to:

A. chair the SRBs,

B. provide guidance to project personnel :

1. to ensure that there is consistency in documentation and planning, and

2. to facilitate coordination of the documentation through the approval cycle, and

C. provide a safety database which consists of:

1. a file containing all of the safety related documentation for each project from 1974 to the present and,

2. a computerized database where each piece of documentation is logged according to a control number, risk level, test completion date, title of the overall program, number and title of any subsequent changes, open or closed status, the originating project office, the name and phone number of the project manager, the test vehicle type (i.e., F-16, B-1B, etc.) and the Job Order Number. This database serves as a useful tool to both the program managers and SES to search for previous projects of a similar nature which helps in identifying hazards, minimizing procedures, corrective actions and risk levels.

D. Advise the center commander of project safety risk levels.

SES is involved with other activities which to some extent directly or indirectly promote the AFFTC flight test safety program:

A. Liaison with the National Aeronautics and Space Administration (NASA) Ames/Dryden Flight Research Facility, United States Army Engineering Flight Activity, Air Force Astronautics Laboratory and other agencies.

B. Training for the USSO program.

C. Training for the Program Managers Course.

D. Attendance and input at Technical Review Boards.

E. Review and approval of aircraft Class II Modifications.

F. Attendance and input at System Safety Working Groups, Preliminary Design Reviews, Critical Design Reviews for new and modified weapon systems.

TEST SAFETY PLANNING

Prior to a safety review board the project manager schedules and completes a technical review board. The program manager also assures that:

A. Potentially hazardous testing has been identified in the test concept documentation.

B. Advanced coordination has occurred for requirements listed in the Statement of Capability to ensure that the safety requirements of each facility are addressed in a timely manner.

C. Other personnel have been contacted with experience in similar activities or testing to assist in potential hazard identification and risk reduction procedures. He has also reviewed the AFFTC/SES database for hazards previously identified and lessons learned in other AFFTC tests of a similar nature.

D. The contractor system safety plans are used to identify potential hazards during test activities. This is in accordance with Military Standard (MIL STD) 882B in which the contractor is required to include as part of the Contract Data Requirements List, various Preliminary Hazard Analyses, System Hazard Analyses and Subsystem Hazard Analyses.

E. The safety requirements and coordination for facilities other than the AFFTC if used.

F. The technical adequacy of the test plan is reviewed and modified as necessary. Technical concerns such as test objectives, instrumentation requirements, test conditions and procedures, sequence of testing, safety related factors, preparation or training requirements, resources required, operations and communications security, and reporting requirements.

SAFETY REVIEW BOARD:

The primary purpose of the Safety Review Board (SRB) is to conduct an independent review of the safety aspects of flight test activities. The SRB is comprised, as a minimum, of a chairman (from SES), an operations representative, and an engineering representative. Additionally the following apply:

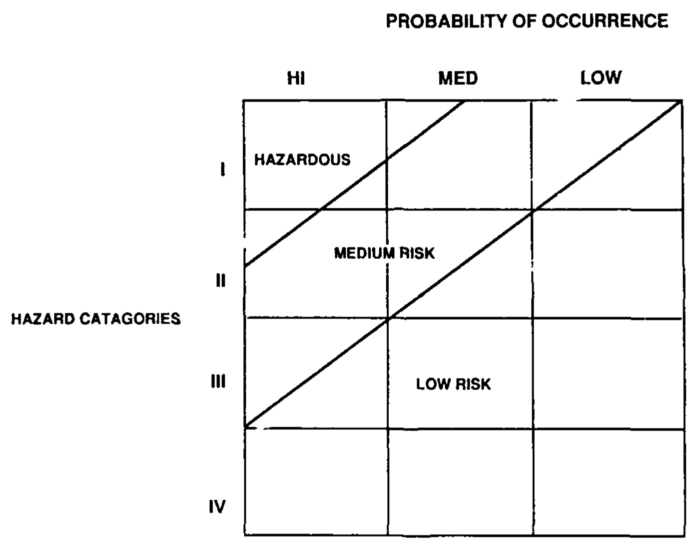
A. All voting members are AFFTC personnel.

B. The AFFTC Contract Management Division may designate a voting member if they have been designated as the Government Flight Representative for contractor flight operations.

C. SRB members are normally selected from agencies having project responsibilities with significant test experience. Operations and engineering test agencies will provide experienced personnel to function as SRB members. These people should be senior in tenure or test experience to the project personnel. They should have experience in the type of test activity to be reviewed. If possible they should be selected from intermediate supervisory personnel with project familiarity but without sufficient project involvement to present a personal conflict of interest.

D. Additional SRB members will be designated from support agencies within the AFFTC if considered appropriate (i.e., bioenvironmental, airfield management, fire department, range safety).

Prior to the start of the SRB the project prepares all of the safety related documentation which includes an analysis of all hazards unique to a specific test activity. The SRB members are supplied with this information along with the test plan as approved by the TRB. The project manager presents to the board which agencies have mishap responsibilities for the test article, the test plan objectives, proposed tests, test methods, test item description, buildup rationale and any other test unique items which are pertinent to the test or activity. The board then reviews the test unique safety aspects of the testing. This review does not include the hazards of normal operation or flying. Specifically those aspects reviewed include the test procedures including test point to test point or test mission to test mission buildup techniques (i.e. airspeed, altitude, dynamic pressure) and how the test plan proceeds from least to most critical test points as far as hazard or severity is concerned. Any predictions based on analysis, simulator, wind tunnel, or laboratory results are reviewed by the board. The procedures used for command and control of the testing are reviewed with particular emphasis on control room procedures and identification of the test personnel responsibilities. The board reviews all hazards identified by the project, identifies any additional hazards that may be discovered, and makes suggestions on how to improve the test from a safety point-of-view. It is not the intent of the board dictate to a project how to do a test or demand that certain safety concerns be incorporated. The boards primary function is to conduct an independent review. This is not to say that the board is without a certain amount of inherent power because, after all, the board members do determine the risk level associated with the tests. If they feel that the project has not taken prudent safety precautions, then they will assess a risk level higher than they might have had prudent safety precautions been taken. Finally, the board makes an assessment of the risk level associated with the testing. The risk level may be broken down to a specific flight, types of tests or even specific test points. The risk is defined as either LOW, MEDIUM or HAZARDOUS risk. Risk assessment is determined as a function of hazard category level and each hazards probability of occurrence. Figure 1 summarizes the risk levels from a matrix of hazard category and probability of occurrence.



The following definitions are also considered when assigning a risk level:

A. LOW risk is defined as tests or activities which present no greater risk than normal operations. Routine supervision is appropriate. The program manager, appropriate test force director or their designated representative will be briefed and will approve all LOW risk tests.

B. MEDIUM risk is defined as tests or activities which present a greater risk to personnel, equipment, or property than normal operations and require more than routine supervision. Any activity or test determined to be MEDIUM risk will be briefed to and approved by the Test Wing Commander within one working day before their anticipated accomplishment.

C. HAZARDOUS is defined as tests or activities which present a significant risk to personnel, equipment, or property, even after all precautionary measures have been taken. Close supervision is required at all levels. Appendix I lists hazards of some tests by discipline. Activities or tests determined to be HAZARDOUS are briefed to and approved by the AFFTC Commander within one working day before their anticipated accomplishment. Tests which should be considered potentially HAZARDOUS are:

1. First flights of new aircraft configurations.

2. Flight envelope expansion.

3. Flutter testing.

4. Rejected takeoffs at high brake energy levels.

5. Single-engine aircraft airstart envelope determination.

6. High angle-of-attack, spin prevention, and out-of-control recovery tests.

7. Helicopter height-velocity envelope determination.

8. Ground and air minimum control speed determination.

9. Flight tests of developmental or prototype unmanned vehicles.

10. Explosives tests.

11. Tests involving high energy LASERS, MASERS, electromagnetic emitters, or hazardous (toxic, radioactive, etc.) materials.

12. Armament testing.

a. Testing with live explosive warheads.

b. Powered flight of developmental or prototype missiles.

c. Flight envelope clearance tests of new armament or release systems.

13. Initial flights after completion of Class II modifications (AFSCR 80-33) which could affect structural integrity, aerodynamic stability, or safety of flight.

14. Initial investigations of stall characteristics or minimum usable flying speeds of new aircraft designs, or of modified aircraft which are predicted to have degraded stall characteristics.

15. Low altitude tests of terrain avoidance/terrain following radars or systems and night low altitude tests.

A coordination package is then prepared with all the above carefully detailed. The package is then coordinated through the projects Test Organization, The Test Group (engineering), the Test Wing (operations), the Directorate of Safety, and then the Center Commanders staff. The project manager and the SRB chairman then brief the program to the AFFTC commander. The Commanders approval is clearance to begin testing.

PROGRAM CHANGES:

Even though all of the above planning may appear to be exhaustive and all inclusive, programmatic changes, test vehicle changes and even unexpected test results occur which may affect the previous safety planning. The proposed changes are documented in a Test Project Safety Review Amendment. If the changes affect the assigned risk level, depending on the extent of the changes:

A. A SRB may be reconvened or,

B. Coordination of the SRB members may be required.

If the changes merely modify some phase of test conduct with no effect on the original safety considerations the the amendment will not require that an SRB reconvene or the board members signatures. The amendment is coordinated again through the Test Organization, through the responsible parent organization: the Test Wing or Test Group, and is approved by the test wing commander if the changes do not significantly impact the original safety considerations. If an SRB had to be reconvened, the same coordination, briefing and approval process that was required for the original package will be duplicated. All amendments are attached to the original safety documents and in most cases all previous amendments as well as the original package are intended to apply to the new amendment. If a change increases the risk level or changes a test previously considered HAZARDOUS, the Center Commander is the final approval authority for amendments.

PERFORMANCE:

One might think that the above process is too complicated and time consuming to be effective. The AFFTC flight test safety record, however has greatly improved since the implementation of SES. The following table summarizes the number of flight test hours, support test hours, SRBs and amendments for the last eight and one-half years at the AFFTC.

YEAR	FLIGHT TEST HOURS	SUPPORT TEST HOURS	SRBs	AMENDMENTS
1980	1836	4964	82	163
1981	2394	5451	67	236
1982	2677	4175	73	284
1983	3894	5451	57	327
1984	3978	4289	60	250
1985	4208	5293	42	220
1986	5697	5452	57	242
1987	5188	5852	85	214
1988	4C97*	6655*	64	172

* Results are as of 29 August 1988.

TESTING:

GENERAL.

Test procedures are addressed in the SRB and it is because they are developed and reviewed prior to testing that they have become effective in minimizing the probability of encountering a given hazard during inherently hazardous testing. AFFTCR 55-23, Test Control and Conduct, specifies the responsibilities and procedures for the control and conduct of flight testing for which the AFFTC is the responsible organization. A test director is designated by the Test Organization director or program manager who executes the test activity as stated in the test plan. The test director ensures key personnel (CTF director, project manager, test controller, project engineer, pilot, etc.) attend pre- and post-flight briefings. The test conductor is responsible for the conduct of these briefings. Test mission conduct is categorized into two types:

A. Those that require only ground monitoring or occasional radio communication between the test aircraft and ground personnel (communication, antenna patterns, navigation, simulated weapon delivery, some performance and radar functional tests) and,

B. Those that require test data to be telemetered and some form of ground control from test personnel (high angle-of-attack, first flight, maximum performance braking, cruise missile free flights, envelope expansion, weapons separation, some gun-fire, and some air-to-air and air-to-ground missile firings). Most tests determined to be HAZARDOUS by the SRB require ground control. Because each test and test program is unique AFFTCR 55-23 is only a guideline for control room procedures and policies. Each CTF generally writes their own operating instructions (OI) for its specific control room procedures. AFFTC experience in mission control of sophisticated aircraft performing complex test maneuvers has given the personnel unique insight into mission control operations. The OI must contain a list of unambiguous commands to be used, along with their meanings.

PERSONNEL.

Key individuals and duties typically found at the AFFTC are:

A. Mission Test Pilot (pilot in command) is responsible for the safe operation of the test aircraft and successful completion of the test mission.

B. Flight Test Director is responsible for all engineering and support aspects of the mission.

C. Flight Test Controller is responsible for real time coordination of ground activities with aircrews; paces the progression through the test cards as agreed to in the mission pre-briefing and defers to the pilot and flight test director for decisions as appropriate.

D. Flight Test Engineer is responsible for the technical adequacy of the test.

E. Operations Engineer is responsible for test aircraft mission preparation and range requirements scheduling.

F. Duty Pilot is responsible for advising mission test pilot from the mission control room regarding normal, work-around and emergency procedures.

PROCEDURES.

For HAZARDOUS tests, the following conditions must exist or the test will be terminated:

A. Safety-of-flight go/no-go requirements as identified in the SRB must function properly.

1. Telemetered Parameters

2. Control Room Displays

3. Aircraft Equipment

4. Ground Support Equipment

5. Etc.

B. The pilot will not proceed to the next test point unless cleared by the test director via the flight test controller.

Key elements found in AFFTC control room procedures:

A. Critical safety of flight parameters for each mission phase are monitored in the mission control room at all times; before, during and after test points and maneuvers.

B. All essential control room personnel are in place before any testing is initiated.

C. Under normal conditions, only the flight test controller communicates with the mission test pilot.

D. All mission control room personnel are in direct communication with each other and the flight test controller. All mission control room personnel monitor air-to-ground communications.

E. The mission test pilot or any member of the mission control room can terminate a test maneuver via the flight test controller.

F. All project control room personnel are qualified in their duties and checked out in test procedures, via a documented training program.

G. All safety or mission-critical steps in the flight are command/response: the flight test controller commands and the mission test pilot responds.

H. The mission test pilot calls the start and completion of each test maneuver.

An example of a command response checklist used for high angle-of-attack testing is contained in Appendix II. This checklist was recently put to test during an F-16 high angle of attack test mission. All of the procedures were followed according to the checklist, yet an improperly mated electrical connector prevented deployment of the spin chute. By using the checklist and the remaining minimizing procedure (pitch rocking maneuver), the aircraft was recovered without incident. An amendment was then filed with SES documenting an additional safety feature to prevent the electrical connector from being improperly mated.

APPENDIX I

A LIST OF TYPICAL HAZARDS

BY TEST DISCIPLINE

1. First flights of new aircraft configurations.

HAZARDS:
- a. Undesirable flight control system characteristics
- b. Inability to control the aircraft
  - Total flight control system failure
  - Hardover stabilator
  - Loss of stick control commands
  - Air data failure
  - Multiple failure of electrical system
- c. Structural overload/failure
- d. Exceeding structural limits
- e. G-induced loss of consciousness
- f. Loss of directional control during ground handling
- g. Blown tire

2. Flight envelope expansion.

HAZARDS:
- a. Undesirable flying qualities
- b. Inadequate control in one or more axes
- c. Out of control
- d. Structural overload
- e. Engine failure to start

3. Flutter testing.

HAZARDS:
- a. Undesirable flying qualities
- b. Structural failure
  - Unanticipated catastrophic flutter
  - Loads exceed structural capability

4. Rejected takeoffs at high brake energy levels.

HAZARDS:
- a. Explosive tire/wheel failure
- b. Loss of directional control on ground
- c. Structural failure of landing gear
- d. Brake/landing gear fire
- e. Blown tire
- f. Airplane departs taxiway/runway

5. Single-engine aircraft airstart envelope determination.

HAZARDS:
- a. Engine will not relight
- b. Engine overtemperature/overspeed during relight(operable)
- c. Engine flameout
- d. Engine stall/surge: recoverable/unrecoverable
- e. Pop/surge

6. High angle-of-attack, spin prevention, and out-of-control recovery tests.

HAZARDS:
- a. Departure from controlled flight
- b. Inability to achieve aerodynamic recovery following a departure
- c. Control recovery chute fails to recover the aircraft
- d. Control recovery chute recovers aircraft but will not release
- e. Failure of control recovery chute emergency jettison
- f. Loss of control after releasing spin chute
- g. CG further aft than desired
- h. Pilot disorientation
- i. Engine stagnation
- j. Accidental firing of recovery chute pyrotechnics
- k. Store impact outside spin area
- l. Control recovery chute entangles the aircraft during ground deployment
- m. Control recovery chute fails to deploy on command after being armed
- n. Control recovery chute deployment exceeds structural limits of mounting assembly
- o. After jettison, control recovery chute impacts ground in populated area

7. Helicopter height-velocity envelope determination.

HAZARDS:
a. Hard landing
b. Engine(s) failure
c. Pilot error (technique)
d. Pilot cuts two engines instead of one

8. Ground and air minimum control speed determination.

HAZARDS:
a. Loss of aircraft directional control
b. Failure of operating engine
c. Departing runway
d. Loss of control during ground roll

9. Flight tests of developmental or prototype unmanned vehicles.

HAZARDS:
a. Loss of unmanned vehicle control
b. Mid-air collision between unmanned vehicle and chase
c. Loss of track of unmanned vehicle
d. Structural failure
e. Dropped object
f. Inadvertent jettison impacts outside range boundaries
g. Jettisoned missile strikes carrier aircraft
h. Hung store
i. Deployment of stabilizing parachute during external carriage

10. Explosives tests.

HAZARDS:
a. Excessive gun gas concentrations in the gun compartment
b. Projectile ricochet
c. Attacking the wrong target

11. Tests involving high energy LASERS, MASERS, electromagnetic emitters, or hazardous (toxic, radioactive, etc.) materials.

HAZARDS:
a. Unplanned exposure to laser light
b. Laser fired at or towards personnel

12. Armament testing including tests of live explosive warheads, powered flight of developmental or prototype missiles and flight envelope clearance tests of new armament or release systems.

HAZARDS:
a. Missile exhaust gas ingestion by engine
b. Hung store
c. Inadvertent release and/or store impact outside of designated area
d. Mid-air collision with photo/safety chase, missile or other aircraft
e. External stores contact aircraft
f. Hang fire

13. Initial flights after completion of Class II modifications (AFSCR 80-33) which could affect structural integrity, aerodynamic stability, or safety of flight.

HAZARDS: See first flights of new aircraft configurations.

14. Initial investigations of stall characteristics or minimum usable flying speeds of new aircraft designs, or of modified aircraft which are predicted to have degraded stall characteristics.

HAZARDS: See first flights of new aircraft configurations and ground and air minimum control speed determination.

15. Low altitude tests of terrain avoidance/terrain following radars or systems and night low altitude tests.

HAZARDS:
a. Aircraft striking ground or obstacles
b. Bird strike on canopy
c. Radiation
d. Pilot incapacitation/loss of consciousness
e. Mid-air
f. Aircraft striking ground in weather

APPENDIX II

F-16 HIGH ANGLE-OF-ATTACK

RECOVERY PROCEDURES

HIGH AOA RECOVERY PROCEDURES

AIRCRAFT ACTION	TEST CONTROL CALLS TO PILOT	PILOT ACTION
DEPARTURE	"DEPARTURE/THROTTLE IDLE"	1. IMMEDIATELY RELEASE CONTROLS 2. THROTTLE TO IDLE
NO RECOVERY	"BEGIN PITCH ROCKING"	1. BEGIN PITCH ROCKING
ERECT SPIN (SUSTAINED YAW RATE ABOVE 50 DEG/SEC)	"DEPLOY/DEPLOY/DEPLOY"	1. DEPLOY SPIN CHUTE
INVERTED SPIN	"RUDDER LEFT/RIGHT...NEUTRAL"	1. APPLY LEFT/RIGHT RUDDER...NEUTRAL RUDDER
NO RECOVERY @ 25K OR 60 SECONDS @ O G	"DEPLOY/DEPLOY/DEPLOY"	1. DEPLOY SPIN CHUTE
AFTER RECOVERY OR NO RECOVERY	"RELEASE CHUTE"	1. RELEASE SPIN CHUTE
NO RECOVERY	"AFT FEED/ AFTERBURNER" (IF ERECT) "BEGIN PITCH ROCKING"	1. AFTFEED/AFTERBURNER (IF ERECT) 2. BEGIN PITCH ROCKING
NO RECOVERY	"DEPLOY ALTERNATE FLAPS"	1. DEPLOY ALTERNATE FLAPS
NO RECOVERY	"JETTISON STORES"	1. JETTISON STORES
NO RECOVERY @ 13K	"EJECT/EJECT/EJECT"	1. EJECT

RESULTS AND PRACTICAL EXPERIENCE USING PARAMETER IDENTIFICATION TECHNIQUES ON THE AM-X PROGRAM AT AERMACCHI

M.Lucchesini, P.Maestrelli
L.Manfriani, P.Chimetto
Technical Department
AERMACCHI S.p.A.
Via Sanvito, 80
21100 - Varese
ITALY

SUMMARY

This paper presents the results and the practical experience gained by introducing the parameter identification technique in AM-X flight test analysis at Aermacchi.

Taking into account the Company's requirement, preference was given to a well known and established program that could guarantee excellent cost effectiveness. The Iliff-Maine code used (MMLE3 program) during the flight test program is shown to be robust enough to provide useable results rapidly in almost any conditions.

Aerodynamic derivatives were evaluated using linear and parabolic models over a wide range of Mach numbers, altitudes and angles of incidence.

Both longitudinal results and an analysis of a lateral departure are presented.

1. INTRODUCTION

The AM-X program originated from a 1977 requirement of the Italian Air Force, which specified a single-seat combat aircraft optimized for battlefield interdiction, reconnaissance and close air support missions.
The convergence of the above operational requirements with those of the Brasilian Air Force led to a joint program carried out by the Italian Companies Aermacchi and Aeritalia and by the Brasilian Company Embraer.

The outcome of this process was the AM-X aircraft, developed using modern technologies such as control and stability augmentation.

As a consequence, conventional criteria for flying qualities evaluation had to be integrated with new analytical approaches.
For example, analysis and evaluation of control system design make increasing use of the direct determination of aerodynamic stability and control derivatives.
The use of parameter identification techniques from the very beginning of the flight tests was therefore envisaged.

A decisive factor in choosing the computer program to be used was that, within the work sharing among the three companies involved, Aermacchi had the task of performing an extensive analysis of flight envelopes already cleared by Aeritalia from the flight mechanics point of view. Taking into account the massive bulk of data to be processed the preference was given to a well known and established program that could guarantee excellent cost-effectiveness:
the Iliff-Maine code (known as "MMLE3" program, ref. 1) modified for the company's computer and integrated with pre- and post-processor to make it user-friendly.

"MMLE3" is a program that evaluates aerodynamic derivatives based on the method of "maximum likelyhood". The real and modelled systems are excited by inputs as recorded in flight; then, by varying the unknown derivatives, the probability of getting the same response from the two systems is maximized. This statistic approach allows one to obviate the complications due to the inevitable uncertainties of measurement and modelling.

This paper presents the results and the experience gained by introducing this technique in AM-X flight test analysis. The practical problems to be confronted were the choice of suitable manoeuvres to be performed by the pilot, the appropriate modelling of the aircraft, the data acquisition and processing system architecture and finally the analysis of the results followed by the verification of the predicted aerodynamic data.

2. FLIGHT TEST TECHNIQUES

The AM-X flight test program addressed the evaluation of the characteristics of the aircraft and its systems throughout the flight envelope.
Seven prototypes were used in the test program, of which five in Italy and two in Brazil, flying a total of over 1300 hrs. by June 1988.

The partecipation in the AM-X program allowed Aermacchi to develop its Flight Test Department significantly, by consolidating the telemetry, ground station and digital PCM acquisition systems.

2.1 Data acquisition and preprocessing

The flight test instrumentation system (see fig. 1) is made up of the onboard digital PCM/FM data acquisition system and the ground based telemetry/recording system. The signals are acquired by dedicated transducers or taken from aircraft system DATA BUSes and transmitted to the ground station by the telemetry system, together with pilot voice.

The ground-based system takes care of aircraft tracking and chooses the better of two signals transmitted through separate channels. The data are then passed to a computer for real time processing and display on CRTs in the control room.

The maximum capacity of the data acquisition system is 512 channels with a sampling frequency ranging from 1 to 256 SPS. Since each sample is represented by an 11-bit word, this corresponds to a 1.44 Mbits/s transmission rate.

Stability and control data are generally sampled at 32 SPS since this allows a good reconstruction of the signal without overloading the system. Digital filters are used where necessary.

2.2 Flight manoeuvres

The flight manoeuvres were chosen to take advantage of the small perturbation approach to stability and control analysis; this allowed data to be obtained from a single flight condition with small variations from the reference conditions. Locally linearised aerodynamic models could therefore be used, with advantages that will become evident in the subsequent discussion.

Care was taken to design a "small perturbation" manoeuvre having the highest possible Signal/Noise (S/N) ratio. High frequency noise originating from vibrations, atmospheric turbulence, sensor accuracy and resolution could be filtered, while low frequency noise close to the system's natural frequencies was more difficult to distinguish.

Test pilots were constantly involved in flight test planning. Difficult pilot tasks such as 3-2-1-1 manoeuvres were perfected during extensive training, with highly satisfactory results. Both multistep and impulse-type manoeuvres were used, as well as frequency sweeps to provide input data for the parameter identification program. Examples of manoeuvres analysed are shown in Fig. 2.

3. MODELLING TECHNIQUES

3.1 General

The standard MMLE3 mathematical models and equations of aircraft dynamics were used as these are most appropriate for the analysis of the large number of test points needed to cover the complete flight envelope.

Lateral and longitudinal dynamics were decoupled by considering the aircraft symmetry in the XZ plane ($I_{xy} = I_{yz} = 0$) and by using data obtained from flight test for the remaining coupled terms in the equations of motion (ref. 2).
The aerodynamic coefficients were linearised as follows:

LONGITUDINAL EQUATIONS IN BODY AXES:

$$\begin{cases} C_N = C_{N\phi} + C_{N\alpha}\alpha + C_{N\delta}\delta + C_{Nq}q\,\frac{c}{2V} \\ C_A = C_{A\phi} + C_{A\alpha}\alpha + C_{A\delta}\delta + C_{Aq}q\,\frac{c}{2V} \\ C_m = C_{m\phi} + C_{m\alpha}\alpha + C_{m\delta}\delta + C_{mq}q\,\frac{c}{2V} \end{cases}$$

where

$\delta = \delta_{elev}\ ,\ \delta_{stab}$

LATERAL/DIRECTIONAL EQUATIONS IN BODY AXES:

$$\begin{cases} C_y = C_{y\phi} + C_{y\beta}\beta + C_{y\delta}\delta + C_{yp}p\,\frac{b}{2V} + C_{yr}r\,\frac{b}{2V} \\ C_l = C_{l\phi} + C_{l\beta}\beta + C_{l\delta}\delta + C_{lp}p\,\frac{b}{2V} + C_{lr}r\,\frac{b}{2V} \\ C_n = C_{n\phi} + C_{n\beta}\beta + C_{n\delta}\delta + C_{np}p\,\frac{b}{2V} + C_{nr}r\,\frac{b}{2V} \end{cases}$$

where

$\delta = \delta_{ail}\ ,\ \delta_{spo}\ ,\ \delta_{rud}$

3.2 Influence of augmentation system on modelling

The schemes of the lateral and longitudinal control systems of the AM-X are given in figure 3.
The presence of feedbacks in the pitch, roll and yaw channels leads to complications when parameter identification is attempted as the closed loop gains and characteristics of the wash-out filters in the feedback loops must be determined. This implies including feedback of the 3 angular velocities in the models of the lateral and longitudinal systems.

An equally valid, and less complicated, approach is to use the concept of an equivalent aircraft developed by Koehler and Wilhelm (ref. 3) which has the advantage of avoiding the need to modify the identification program itself. This approach is possible for systems where the feedback can be considered constant, as in the case of the AM-X at a given flight condition (the wash-out filter has virtually no effect on the dynamics of the transient response).
In this case the displacements of the control surfaces can be described by an equation of the form:

$$\delta = a \begin{Bmatrix} p \\ q \\ r \end{Bmatrix} + b\ \delta_{pilot} \qquad \text{where a,b = constant}$$

and the concept of an equivalent aircraft can be introduced.
Taking the basic aircraft equations

$$\dot{\underline{x}} + [A]\ \underline{x} = [B]\ \underline{u}$$

and adding the control system feedback as

$$\underline{u} = [C]\underline{x} + [D]\underline{u}_{pilot}$$

we obtain the "equivalent aircraft equations":

$$\dot{\underline{x}} + ([A] - [B]\ [C])\underline{x} = [B]\ [D]\underline{u}_{pilot}$$

which are of the same form as those of the basic aircraft.

In the case of the longitudinal equations, for example, only the column containing pitch rate terms is modified.
C_{Lq} and C_{mq} are replaced by

$$\begin{cases} C_{Lq\ equiv} = C_{Lq} + G_q K_s C_{L\delta\ long} \\ C_{mq\ equiv} = C_{mq} + G_q K_s C_{m\delta\ long} \end{cases}$$

where

$$\begin{cases} C_{L\delta\ long} = C_{L\delta e} G_e + C_{L\delta\ stab} G_s K_s \\ C_{m\delta\ long} = C_{m\delta e} G_e + C_{m\delta\ stab} G_s K_s \end{cases}$$

Although this technique seemed particularly suited to the AM-X and was thus adopted for parameter identification of the augmented aircraft, the results obtained were disappointing. For example, the SPO manoeuvre with full FCS carried out at 5000 ft and Mach 0.5 shown in fig. 4 indicates that the curve fitting is highly approximate. A further difficulty was the identification of the stability derivatives: the program attempted to represent damping effects by varying both C_{L_α} and C_{mq}, making it necessary to fix C_{L_α} at a value previously identified using the "Bare airframe" model.

The results obtained in this case are summarised in the following table, in which the closed loop characteristics of the "bare A/C" were calculated by using nominal augmentation values given by known values of feedback gains:

MACH 0.5 H = 5000 ft A11/FLT 15/ CONFIG. CRU CLEAN				
TYPE OF IDENTIFICATION	OPEN LOOP		CLOSED LOOP	
	ω_{spo}	ζ_{spo}	ω_{spo}	ζ_{spo}
WIND TUNNEL DERIVATIVES + NOMINAL AUGMENTATION	3.40	0.290	3.65	0.690
IDENTIFIED BARE AIRCRAFT + NOMINAL AUGMENTATION	3.60	0.310	3.92	0.750
IDENTIFIED EQUIVALENT AIRCRAFT DERIVATIVES	/	/	2.91	0.410
FITTING ON "q"	3.80	0.315	3.55	0.700

TABLE 1 - Short period frequency and damping ratios obtained using different methods of identifying aerodynamic parameters.

The discrepancies are due to limited actuation rates of the stabiliser and various non-linear elements in the real system, such as hysteresis and dead bands, which are not modelled by the schemes of fig. 3.

The presence of non-linear elements makes the aircraft response particularly sensitive to pilot input. The impression gained was that the non-linearities would inhibit the identification of control system parameters even if complicated models of the system were used.

In the light of these results the most appropriate course of action seemed to be to limit parameter identification to the bare aircraft and to obtain dynamic parameters (frequencies and damping ratios) of the augmented aircraft using known values of feedback gains. These gains were available as design data and were substantiated by ground tests.

To check the validity of this approach, a fitting was made to the pitch rate signal obtained during a flight test where the same manoeuvre was repeated with FCS engaged and disengaged. The results, given in the last row of table 1, show satisfactory agreement with those of the bare aircraft model with nominal augmentation (row 2).

3.3 Parabolic model

Linear aerodynamic models were employed throughout the flight envelope and generally gave good results. The only exception was the longitudinal model at low speeds when C_L and C_m could no longer be described as linear functions of alpha. In this case attempts to fit the normal load factor (a_n) traces suggested that a quadratic term in α would be necessary (see fig. 5).
This was done, without any modification to the program, by using an α^2 input signal as an additional control input. The program was used to calculate derivatives with respect to α^2 ($C_{N\alpha}2$, $C_{A\alpha}2$, and $C_{m\alpha}2$) which were added to the linear model as follows:

$$C_N = C_{N\phi} + C_{N\alpha}\alpha + C_{N\alpha}2\,\alpha^2 + C_{N\delta}\delta + C_{Nq}q\frac{c}{2V}$$

$$C_A = C_{A\phi} + C_{A\alpha}\alpha + C_{A\alpha}2\,\alpha^2 + C_{A\delta}\delta + C_{Aq}q\frac{c}{2V}$$

$$C_m = C_{m\phi} + C_{m\alpha}\alpha + C_{m\alpha}2\,\alpha^2 + C_{m\delta}\delta + C_{mq}q\frac{c}{2V}$$

Fig. 6 shows the effect of the presence of these non-linear terms in α^2. More detailed results are given in section 4.
Parabolic models were only used to analyse lateral/directional motion in certain cases, such as roll departures at high α. Here the linear model was found to average the roll rate (p) signal excessively (see fig. 14), and a $Cl_p2\,p^2$ term was introduced in the same manner as the α^2 term in the longitudinal case, using p^2 as a fictitious control input.
These results fit the recorded traces more accurately than the linear model, as will be shown in more detail in section 5.

4. ANALYSIS OF THE LONGITUDINAL MODEL RESULTS

4.1 General

The longitudinal model results presented in this paragraph were obtained by analysing some manoeuvres performed at three different altitudes (5000, 15000 and 30000 ft) with a range of Mach numbers going from ~0.3 to ~0.82.
All coefficients refer to the "cruise" configuration (i.e. no flap and slat deflections), without external stores.

In principle, it should be possible to identify all the aerodynamic parameters relevant to the classical short period approximation, namely

$C_{L\phi}$, $C_{L\alpha}$, $C_{L\,\delta e}$, $C_{L\delta s}$, C_{Lq}

$C_{m\phi}$, $C_{m\alpha}$, $C_{m\,\delta e}$, $C_{m\delta s}$, C_{mq}, $C_{m\dot{\alpha}}$

In practice, however, this is seldom accomplished due to convergence problems of the algorithm or because one or more of the parameters have negligible influence on the aircraft behaviour in the test being considered. It is then advisable to "lock" these coefficients at convenient values (usually taken from the aerodynamic data set based on wind tunnel data).

The most important aerodynamic derivatives, i.e.:

$C_{L\alpha}$, $C_{m\alpha}$, C_{mq}, $C_{m\delta s}$,

were always identified; the others were identified whenever possible except for $C_{m\dot{\alpha}}$ which was always fixed at the predicted value.
All data are referred to the 25% m.a.c. point; Mass, c.g. position and pitch inertia were always considered as constant during a test although fairly small variations were actually observed.
Engine thrust effects were not taken into account because it was found that they were virtually negligible.

4.2 Comparison of experimental and predicted aerodynamic derivatives

The values of the most important longitudinal aerodynamic derivatives identified from flight test were compared with predicted data based on wind tunnel tests, taking into account theoretical aeroelastic effects.

Average values obtained using a linear model were first plotted as a function of Mach number at different altitudes (figs. 7-10); an attempt was then made to identify the dependancy on α at a fixed Mach number using a limited number of tests analysed with a parabolic model (figs. 11-13).

4.2.1 Longitudinal derivatives versus Mach number (linear model results)

It must be noted that the curves presented here refer to constant mass and varying Mach number, hence a varying average angle of incidence. They are therefore not representative of the variation of aerodynamic derivatives with Mach, especially in the low Mach number range:

h = 5000 ft	$0.3 \leq M \leq 0.82$	$0° \leq \alpha \leq 10°$
15000 ft	$0.3 \leq M \leq 0.78$	$2° \leq \alpha \leq 12°$
30000 ft	$0.5 \leq M \leq 0.8$	$3° \leq \alpha \leq 14°$

The main purpose of the diagrams shown in figures 7-10 is to compare flight test data with predicted (wind tunnel) data. The test points are plotted together with their uncertainty ("Cramer-Rao") bounds multiplied by what Iliff and Maine call a "Fudge Factor" equal to five. The solid line represents predicted data while the broken line represents a "weighted average" of flight test data (each data point has a weight equal to the reciprocal of its Cramer-Rao bound).

The correlation with predicted derivatives is fairly good, except for C_{m_q} and $C_{m_{\delta s}}$ at lower altitudes where the effect of uncertainties in the aeroelastic coefficients is greater. It is not surprising that large errors and uncertainties in C_{L_q} correspond to similar ones in C_{m_q} (see, for instance, the three points at 30000 ft and M = 0.8) since they are intimately connected in the ω-ζ modelling. The C_{m_ϕ} shift at low Mach confirms what was already observed during early flight tests and was explained as an error in prediction of the downwash at zero lift (ϵ_0).

4.2.2 Longitudinal coefficients versus alpha (parabolic model results)

The scope of this part of the analysis was to show how much information could be obtained from a few test points where only average values of the derivatives are available. In fact, by analysing three S.P.O. manoeuvres, characterised by a narrow α range, as well as three wide range 3-2-1-1 manoeuvres at fixed Mach number (0.5) and different altitudes (5000, 15000 and 30000 ft), quite a detailed representation of C_L vs. α and C_m vs. α could be obtained from 0° to 15°.
This was accomplished by performing a parabolic spline fit of the following functions

$$C_{M_\phi}(\alpha),\ C_{M_\alpha}(\alpha),\ C_{M_\alpha}2(\alpha);\quad C_{L_\phi}(\alpha),\ C_{L_\alpha}(\alpha),\ C_{L_\alpha}2(\alpha)$$

such that the resulting curves would optimally approximate the measured average values in the least-squares sense.
The curves were then reconstructed as:

$$C_L = C_{L_\phi}(\alpha) + C_{L_\alpha}(\alpha)\alpha + C_{L_\alpha}2(\alpha)\alpha^2$$

$$C_m = C_{m_\phi}(\alpha) + C_{m_\alpha}(\alpha)\alpha + C_{m_\alpha}2(\alpha)\alpha^2$$

This is shown in figures 11-13. If these data are plotted as C_m vs. C_L the agreement with predicted data is very good, especially in the high angle of incidence range, where the contribution of $C_{L_\alpha}2$ is most important. In the low α range the results are consistent with those obtained with the linear model as presented in the previous paragraph.

5. ANALYSIS OF THE LATERAL/DIRECTIONAL MODEL RESULTS

5.1 General

The analysis of lateral/directional data is still under way because priority was given to the longitudinal results. This was done for two reasons. The first was to allow the stability margins of the aircraft to be more closely monitored throughout the tests. The second, of a more practical nature, was that compliance with MIL specifications for the dynamic lateral models, (which are based on dutch roll frequencies and damping ratios) could be demonstrated more easily by using

fitting methods, maintaining the same quality of results as by using parameter identification.
This was possible because the damping ratio in augmented and unaugmented lateral modes never exceeded a value of 0.707, below which fitting methods have been shown to be as reliable as parameter identification in determining frequency and damping of the system.
At this stage, therefore, the use of parameter identification is limited to refining the program for the lateral modes and for verifying wind tunnel estimates of aerodynamic derivatives at particular points.

The following paragraph presents the results considered to be of most interest, namely the analysis of a roll departure which occurred at low speeds in the clean configuration.

5.2 Analysis of a lateral departure

A tendency to autorotation ($C_{l_p} > 0$) at angles of attack close to the stall was identified during rotary balance wind tunnel tests. In the course of flight tests, a roll departure occurred during a wind-up turn at low speed; it was then decided to analyse this manoeuvre as a check of the wind tunnel prediction.

As a first step a standard lateral/directional model was used. The results obtained were unsatisfactory. After a number of attempts in which the program showed no signs of converging, the results of figure 14 were obtained which show that the reconstructed roll rate (p) signal is highly averaged compared with the recorded signal.
The aerodynamic model was therefore changed by including a term in $C_{l_{p^2}}$ and fixing C_{l_p} at zero (see chapter 3).
The aim of this modification was to prevent the onset of a premature roll departure of the model due to very small fluctuations in the p signal. While this has the effect of underestimating the driving moment (C_l) in the initial phases of the departure it also leads to an overestimation once the roll is fully established.
This in turn means that the lateral control derivative opposing and eventually checking the departure is also overestimated. This effect can be seen in fig. 15 which presents the best results obtained. These are summarised in the following table:

DERIVATIVE	EXPECTED VALUE	MMLE3 ESTIMATION
$C_{y\beta}$	-.01660	-.0407 ± .0272
$C_{l\beta}$	-.00271	-.00152 ± .0065
$C_{n\beta}$	+.00142	-.00337 ± .0047
C_{l_p}	+.51200	$C_{l_p} = 0$ $C_{l_{p^2}} = .000042 \pm .0000019$
$C_{l\delta ail}$	+.00130	+.0112 ± .0046
$C_{n\delta spo}$	+.000365	-.000106 ± .000460
$C_{n\delta rud}$	-.00110	+.00473 ± .0049

TABLE 2 - Results of parameter estimation applied to a roll departure

It is worthwhile spenuing some time on the derivations of the rolling moment coefficients (C_{l_p}) from the values of $C_{l_{p^2}}$ identified.

$$C_{l_{p^2}\,ident}\; p^2 = C_{l_{\hat{p}^2}} \left(\frac{pb}{2V\;57.3}\right)^2 \quad \text{where p is in °/sec}$$

$$C_{l_{\hat{p}^2}} = C_{l_{p^2}\,ident} \left(\frac{2V\;57.3}{b}\right)^2$$

$$C_{l_p} = C_{l_p} + C_{l_{\hat{p}^2}}\, 2\hat{p}$$

where $\hat{p} = \dfrac{pb}{2V\ 57.3}$

In this case: $V = 103.67$ m/sec

$b = 8.874$ m

$0 \leq p \leq 40°/\text{sec}$

$0 \leq \hat{p} \leq 0.03$

$C_{l_{p\,av}} = 1.071 \qquad (\hat{p}_{av} = 0.0071)$

The accuracies of the estimated derivatives given in table 2 show that the motion is almost exclusively influenced by the C_{l_p} and $C_{l_{\delta ail}}$ derivatives.
$C_{l_{\delta spo}}$ was not identified because of an unrealistic interaction between the estimated values of $C_{l_{\delta ail}}$ and $C_{l_{\delta spo}}$.

An attempt was made to use the estimated values of $C_{l_{p^2}}$ to reconstruct the C_l curves obtained using the rotary balance within the range of roll rates encountered in flight.
The results are shown in fig. 16.
The amount by which the control derivative $C_{l_{\delta ail}}$ is overestimated can be quantified as follows (see fig. 16):

$$\int_{\hat{p}=0}^{\hat{p}_{max}=0.03} C_{l\ \text{IDENT.}}\, d\hat{p} = 0.00068$$

$$\int_{\hat{p}=0}^{\hat{p}_{max}=0.03} C_{l\ \text{ROTARY BAL.}}\, d\hat{p} = 0.00023$$

$$K = \frac{0.00068}{0.00023} = 2.957 \sim 3.0$$

$$C_{L_{\delta ail}\,\text{EFFECTIVE}} = \frac{C_{L_{\delta ail}\,\text{ESTIMATED}}}{K} = \frac{0.0112 \pm 0.0046}{3} = 0.0037 \pm 0.0015$$

This is much closer to the expected value.

6. CONCLUSIONS AND FUTURE DEVELOPMENTS

The MMLE3 parameter identification code was used during the AM-X flight test program to evaluate aerodynamic derivatives over a wide range of Mach numbers, altitudes and angles of incidence.

The bulk of the results presented refers to the longitudinal case and shows good agreement with wind tunnel results in most conditions. In particular, the use of a parabolic model is shown to give a good representation of the aerodynamic coefficients in the high α region, even when only a few data points are available.

A lateral roll departure is also analysed to demonstrate the applicability of the code to manoeuvres close to stall conditions.

The program is shown to be robust enough to provide useable results rapidly in almost any conditions; in addition, a lot of additional information can be extracted when the program output is interpreted with deeper insight and creativity.

The ongoing AM-X test program at high angle of incidence will provide the opportunity of fully exploiting the capabilities of the MMLE code, using complete six-degree-of-freedom modelling to identify aerodynamic parameters during extreme manoeuvres.

REFERENCES

1. Maine, Richard E.; and Iliff, Kenneth W.: User's Manual for MMLE3, a General FORTRAN Program for Maximum Likelyhood Parameter Estimation. NASA TP 1563, November 1980.

2. Maine, Richard E.; and Iliff, Kenneth W.: Application of Parameter Estimation to A/C Stability and Control. The Output-error Approach. NASA RP 1168, June 1986.

3. Koehler, R.; Wilhelm, K.: Closed Loop Aspects of Aircraft Identification. AGARD LS 104, November 1979.

4. Maine, Richard E.: Programmer's Manual for MMLE3, a General FORTRAN Program for Maximum Likelihood Parameter Estimation. NASA TP 1690, June 1980.

5. Maine, Richard E.; and Iliff, Kenneth W.: Identification of Dynamic Systems AGARD-AG-300, September 1983

6. Marchetto, B.; and Dolce G.A.: Esperienze di Identificazione di Parametri da Prove di volo AIDAA-083-85, Settembre 1985

ACKNOLEDGEMENTS

The authors wish to thank Mr. B.Bufacchi of Aermacchi Technical Department, for his help with this work.

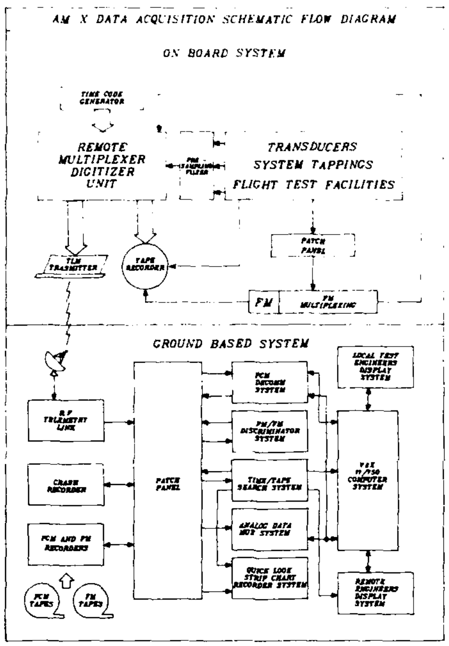


FIGURE 1 - AM-X DATA ACQUISITION SCHEMATIC FLOW DIAGRAM

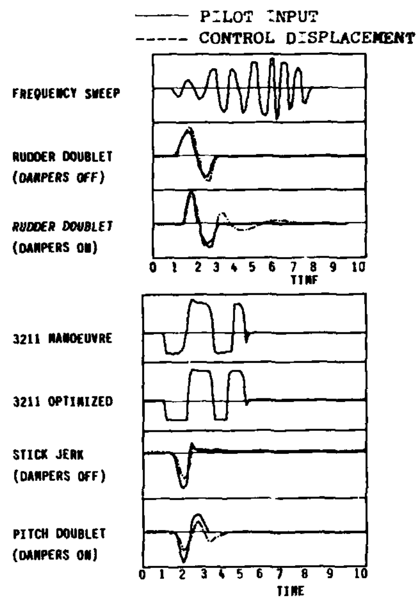


FIGURE 2 - EXAMPLES OF ANALYSED FLIGHT TEST MANOEUVRES USED FOR PARAMETER IDENTIFICATION ACTIVITY

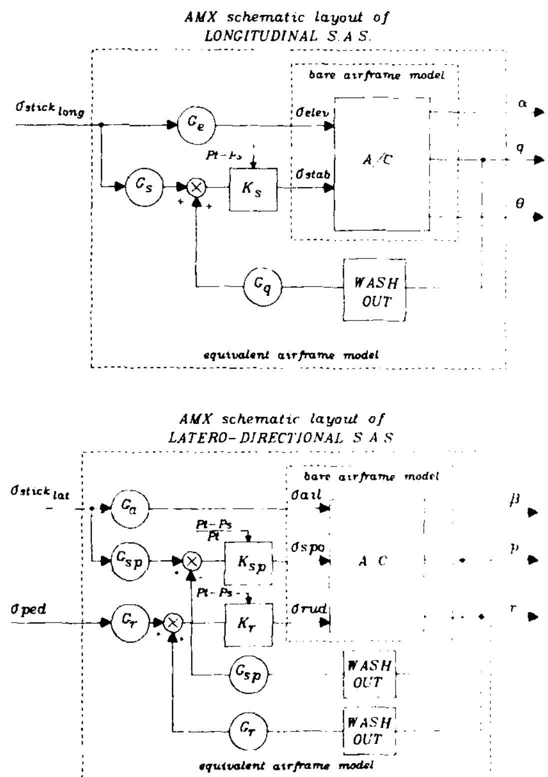


FIGURE 3 - AM-X SCHEMATIC LAYOUT OF LONGITUDINAL AND LATERO-DIRECTIONAL S.A.S.

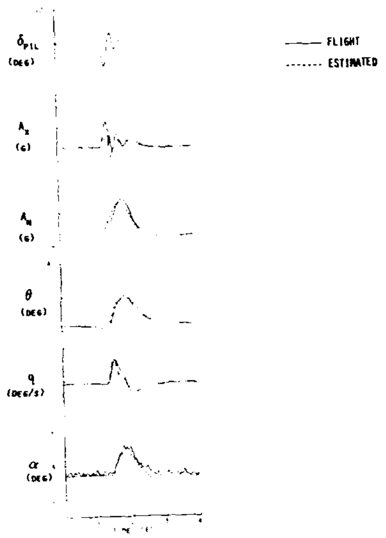


FIGURE 4 - COMPARISON OF A MEASURED AND COMPUTED LONGITUDINAL MANOEUVRE OF AM-X BY USING THE "EQUIVALENT A/C" CONCEPT

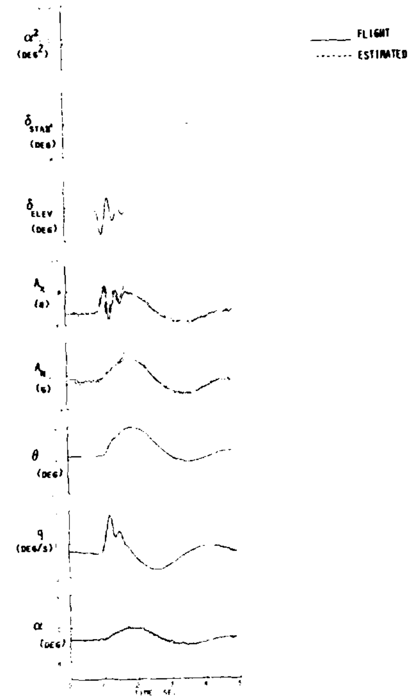


FIGURE 6 - *COMPARISON OF A MEASURED AND* COMPUTED LONGITUDINAL MANOEUVRE OF AM-X WHERE α^2 TERMS ARE PRESENT

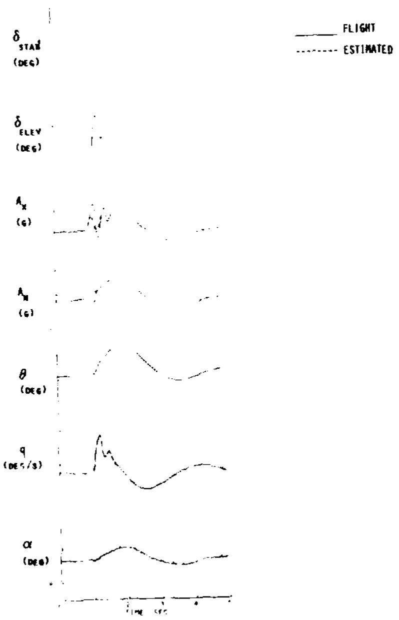


FIGURE 5 - COMPARISON OF A MEASURED AND COMPUTED LONGITUDINAL MANOEUVRE OF AM-X WHERE THE NEED OF α^2 TERMS IS APPARENT

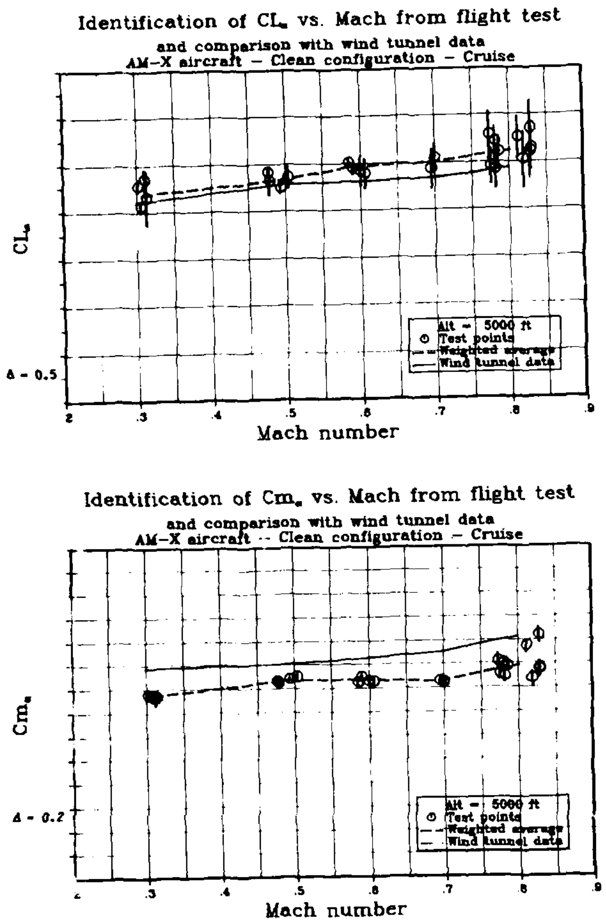


FIGURE 7

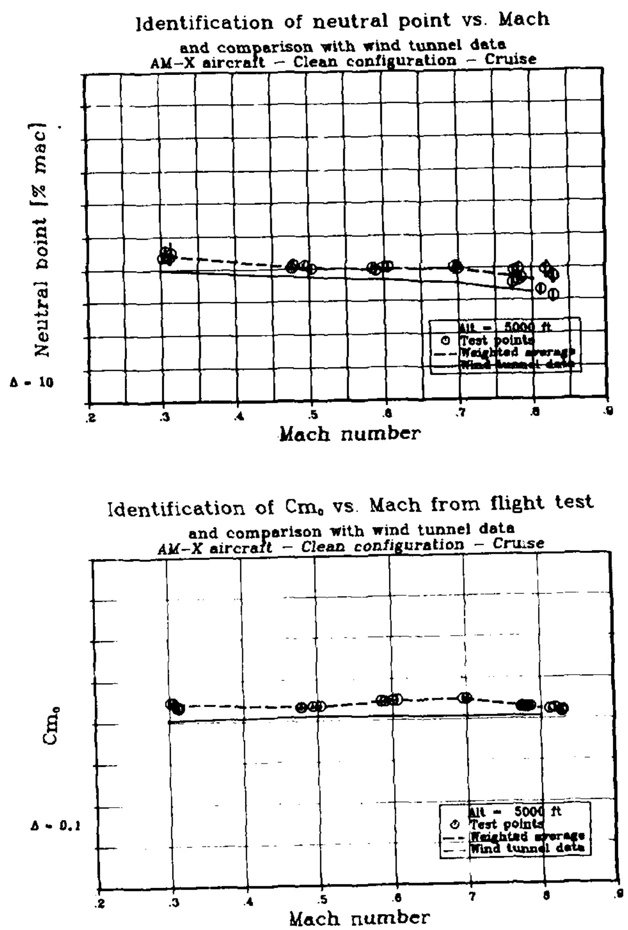


FIGURE 8

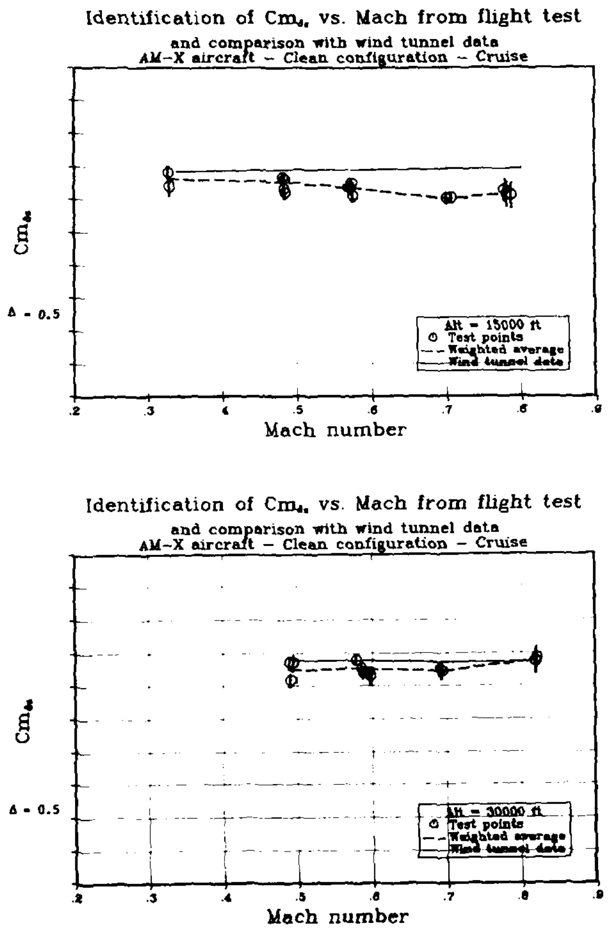


FIGURE 9

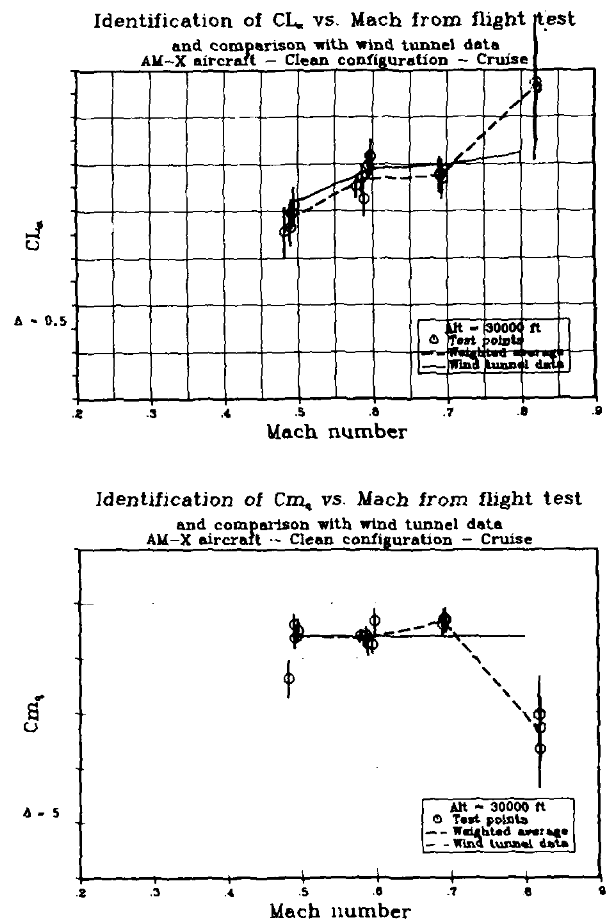


FIGURE 10

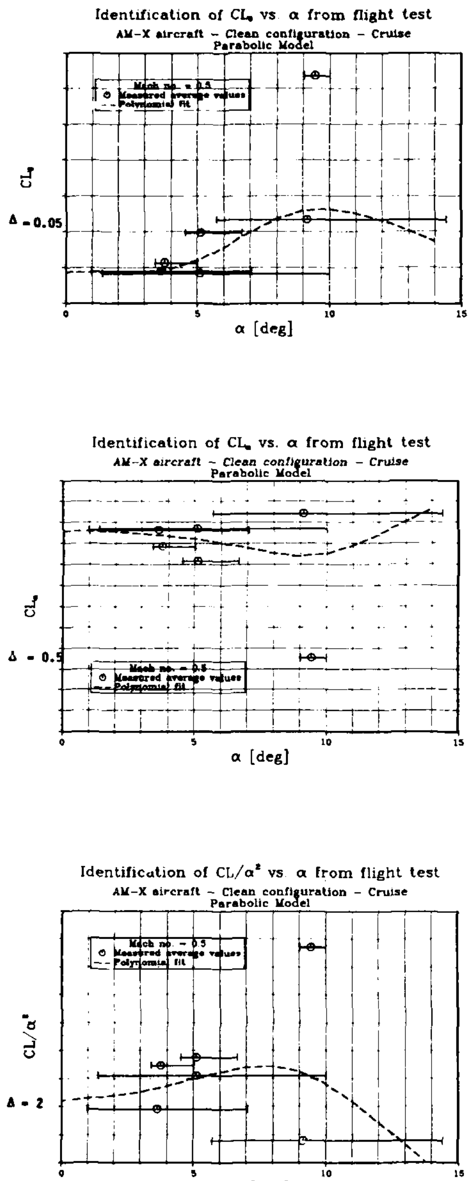


FIGURE 11

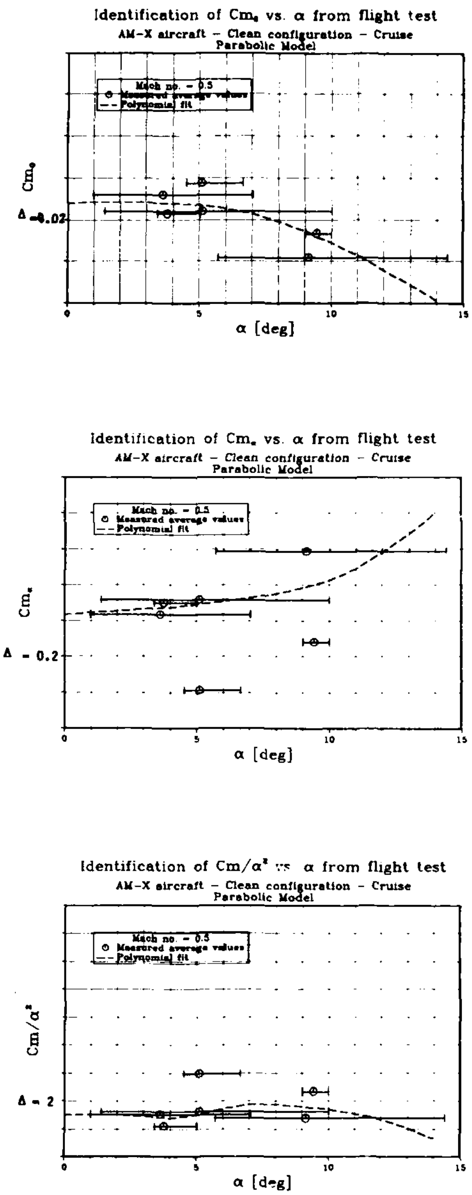


FIGURE 12

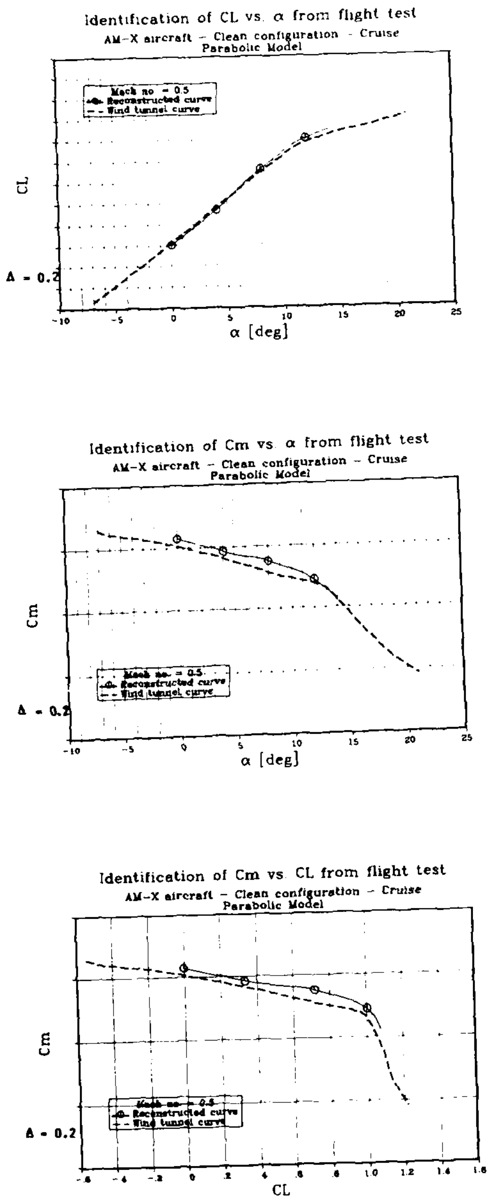


FIGURE 13

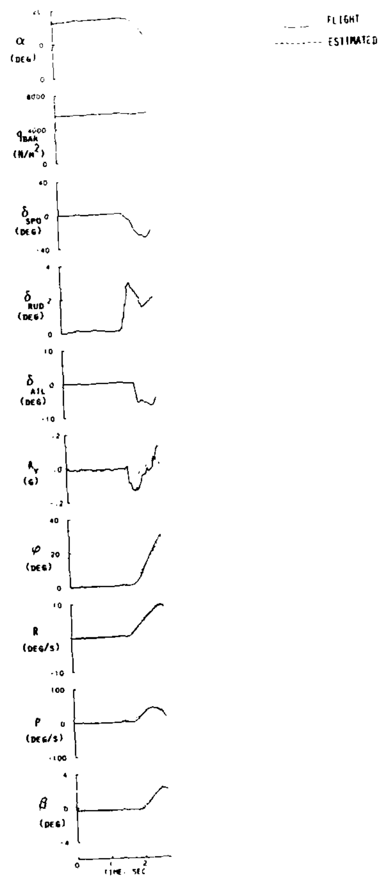
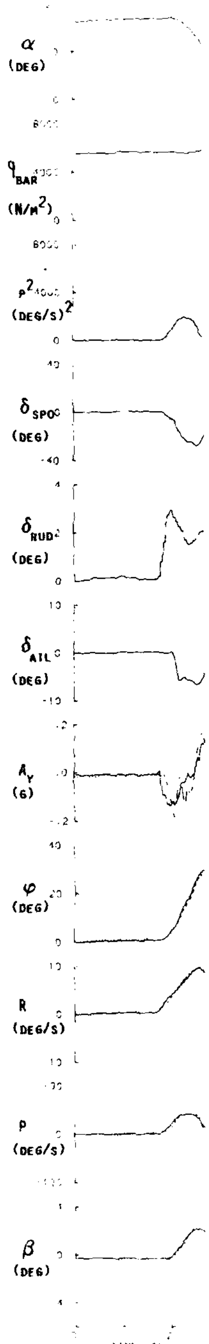


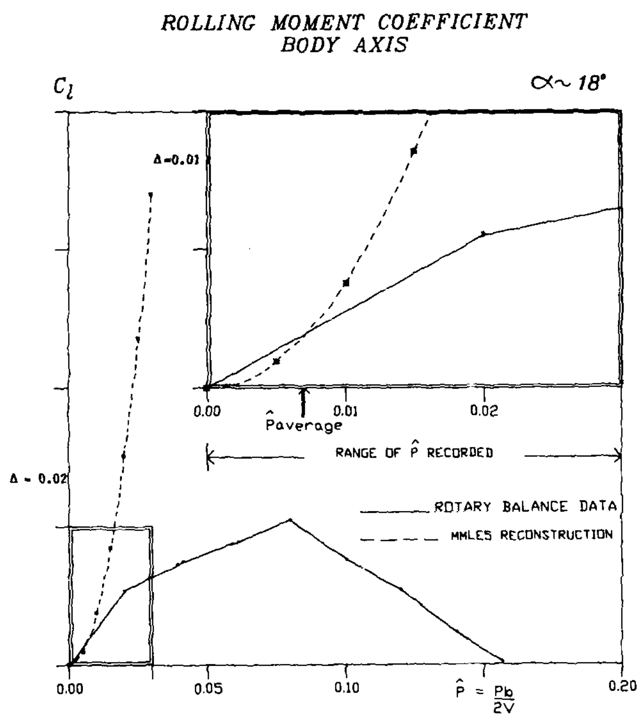
FIGURE 14 - COMPARISON OF A MEASURED AND COMPUTED LATERAL DEPARTURE OF AM-X WHERE THE NEED OF THE p^2 IS APPARENT



———— FLIGHT
-------- ESTIMATED

<u>FIGURE 15</u> - COMPARISON OF A MEASURED AND COMPUTED LATERAL DEPARTURE OF AM-X WHERE THE p^2 TERM IS PRESENT

<u>FIGURE 16</u> - ROLLING MOMENT RECONSTRUCTION



LES ENSEIGNEMENTS DE L'INTEGRATION DES SYSTEMES EMBARQUES

par JACQUES TRESSET
Ingénieur Navigant d'Essais
CENTRE D'ESSAIS EN VOL - 91220 BRETIGNY-SUR-ORGE (FRANCE)

RESUME

Ce document se propose de présenter un ensemble de réflexions qui ont été menées au Centre d'Essais en Vol afin de proposer des solutions visant à minimiser les coûts globaux et les délais des essais en vol liés à la mise au point des systèmes d'armes aéronautiques fortement intégrés.

1. DEFINITION D'UN SYSTEME INTEGRE SUR AVION DE COMBAT

Pour un avion d'armes, un système intégré est l'ensemble des équipements dialoguant par l'intermédiaire d'un ou de plusieurs bus numériques gérés par un ou plusieurs calculateurs et dont l'objectif est de satisfaire un ensemble de fonctions opérationnelles permettant de répondre aux besoins des utilisateurs exprimés dans des spécifications opérationnelles ou dans une fiche programme.

L'ensemble de ces équipements n'est pas limité et croît régulièrement au fil des années proportionnellement aux exigences des utilisateurs en terme de fonctions opérationnelles. La complexité qui en résulte se traduit par des augmentations spectaculaires des coûts d'étude, de développement et d'essais en vol.

Aujourd'hui la notion de système n'est plus limitée aux équipements du système d'armes proprement dit, mais bien à la quasi totalité des équipements de l'avion comme les commandes de vol, le système propulsif, les interfaces hommes-machine mais aussi à l'ensemble des moyens opérationnels permettant de réaliser et d'exploiter la mission.

La planche 1 montre un exemple de système d'armes fortement intégré, de structure classique comportant des capteurs, des armes, des moyens de communication et d'identification, les commandes de vol et le moteur, l'interface homme-machine, un ensemble de gestion et de calcul qui peut être redondant permettant de sécuriser certaines fonctions critiques, etc...

2. ANALYSE DES ESSAIS D'INTEGRATION DES SYSTEMES EMBARQUES

2.1 - Analyse de l'évolution des systèmes

Ce paragraphe a pour but d'examiner l'évolution de la complexité des systèmes d'armes des avions de combat en s'appuyant sur l'exemple des principaux programmes français des quinze dernières années.

Un critère possible pour évaluer la complexité d'un système est celui consistant à dénombrer le nombre d'équipements utilisés, ainsi que le nombre de fonctions opérationnelles réalisées par ce système (on entend par fonction opérationnelle une fonction correspondant à l'expression d'un besoin opérationnel précis; exemple de FO : conduite de tir d'un missile donné). D'autres critères pourraient bien sûr être retenus, comme les volumes d'échanges de données et les charges des calculateurs mais il nous a semblé plus réaliste de retenir le premier critère car il correspond mieux à l'évaluation de la charge d'essais en vol qu'il sera nécessaire de réaliser.

L'examen de la planche 2 fait apparaître l'évolution dans le temps du nombre d'équipements à essayer en vol ainsi que du nombre de fonctions opérationnelles. On peut constater que c'est au niveau des fonctions opérationnelles que l'évolution est la plus importante.

La planche 3 cherche à représenter l'évolution des coûts rapportés à l'heure de vol pour un programme de développement d'un avion de combat récent. L'examen de cette planche montre que le coût de l'heure de vol d'un prototype est sans commune mesure avec le coût équivalent pour un avion de servitude d'un type d'usage courant dans les forces ou dans l'aviation générale.

2.2 - Analyse de la méthodologie utilisée précédemment

Les moyens aériens utilisés en France pour les essais en vol des programmes d'avions de combat sont typiquement de deux sortes : d'une part les avions prototypes et d'autre part les avions de servitude.

En se référant aux programmes précédents, ce paragraphe a pour but de faire la critique objective de l'utilisation de ces différents moyens.

En ce qui concerne les avions prototypes, on peut constater que si ces avions en début de programme sont bien utilisés à la mise au point de la plate-forme elle-même ainsi qu'à l'optimisation de l'interface homme-machine, ils sont trop souvent utilisés comme banc d'essais des équipements, alors que ces derniers n'en sont qu'à une phase de mise au point. Cette utilisation des avions prototypes doit être reconsidérée dans le cadre des programmes futurs car les coûts induits sont prohibitifs et peuvent être réduits de façon significative par l'utilisation d'autres moyens. En conséquence, ces moyens très coûteux doivent être réservés à la mise au point de la plate-forme et à la validation finale des fonctions opérationnelles, comme par exemple les conduites de tir d'armements.

Parallèlement, l'expérience passée montre que l'utilisation et la définition des avions de servitude n'étaient pas optimales. En effet, d'une façon générale, la conception de ces avions était telle qu'elle favorisait la mise au point de l'équipement lui-même, mais n'abordait que de façon incomplète la mise au point des échanges que cet équipement devait assurer au sein de l'ensemble du système. Dans le cas du programme MIRAGE 2000, une expérience a été menée qui a consisté à utiliser des avions de servitude (MYSTERE XX FALCON) dont l'architecture était le reflet de celle de l'avion final (aux armements près). L'utilisation de ces avions dotés d'équipements prototypes a été bénéfique pour la mise au point des fonctions mais a posé des problèmes de disponibilité d'équipements au début du programme.

En ce qui concerne les moyens sol, trois types de moyens ont été utilisés par les Centre d'essais en vol lors des programmes précédents :

- les moyens de simulations pilotées. Ces moyens ont été plus spécialement utilisés pour préciser les caractéristiques des interfaces hommes-machine.

- les bancs d'équipements dont les fonctions étaient limitées à la mise au point du minimum d'échanges dont avait besoin l'équipement sur avion de servitude.

- les bancs d'intégration globale dont le rôle était de valider l'ensemble des échanges du système d'armes avant de le valider et de l'évaluer en vol. Ces bancs étaient également stimulables par des enregistrements effectués en vol.

3. ENSEIGNEMENTS DES ESSAIS D'INTEGRATION

Ce paragraphe a pour objectif d'essayer de tirer les leçons de l'expérience des essais d'intégration des programmes précédents, afin de proposer dans le cadre de nouveaux programmes, une méthodologie qui vise à minimiser les essais d'intégration sur avions prototypes, au profit d'essais au sol et sur avions de servitude.

Le premier objectif pour les services chargés des essais, est à partir des spécifications initiales et par un travail d'analyse fonctionnelle de traduire le besoin opérationnel en fonctions que doit remplir le système. Les moyens de simulations pilotés doivent être utilisés de façon intensive pour ce travail et leur emploi ne doit pas être limité à la seule mise au point de l'interface homme-machine.

Le second objectif est de reconsidérer la méthode de mise au point des principaux équipements du système d'armes comme le radar, les contre-mesures, l'optronique, etc... En effet il conviendra de se donner tous les moyens d'essais au sol et en vol qui éviteront d'accumuler les vols longs et coûteux sur avions prototypes. Parmi ces moyens on peut citer :

- Très en amont de l'essai de l'équipement lui-même, le principe de la stimulation d'un modèle informatique de l'equipement réalisé à partir d'enregistrements effectués en vol est d'ores et déjà un moyen qui a fait ses preuves dans des études préparatoires (radar par exemple).

- Lorsque les premiers équipements prototypes sont disponibles, un moyen très puissant de mise au point des fonctions et des traitements internes à l'équipement est la stimulation du matériel lui-même, réalisée là aussi à partir d'enregistrements effectués en vol sur avions de servitude.

Ces principes de stimulation nécessiteront des systèmes d'enregistrements très performants à bord des avions de servitude. Ce point sera évoqué plus loin.

- L'expérience des programmes précédents a mis en évidence un autre problème en ce qui concerne les grands équipements, qui est qu'il existait un découplage trop important entre la mise au point et l'intégration de cet équipement au système d'armes. Ce point très important se traduisait par le fait que lorsque le matériel était livré par l'équipementier à l'avionneur chargé de l'intégration, bien souvent la mise au point des échanges que devait assurer cet équipement au sein du système avait été négligée au profit de la mise au point du fonctionnement interne du matériel. Afin de remédier à ce problème, la réalisation par les équipementiers de moyens de simulation d'environnement, conformes à une définition approuvée par l'avionneur chargé de l'intégration est certainement une solution pour les programmes à venir.

- Le dernier point en ce qui concerne les équipements est la méthodologie à retenir pour effectuer la mise au point sur avion de servitude. Les essais en vol devront être menés de telle façon que le soucis permanent des responsables d'essais soit de procéder à la mise au point des échanges en plus de la mise au point du fonctionnement interne du matériel. Ce soucis aura bien sûr des implications importantes en ce qui concerne la définition des avions de servitude.

Enfin le troisième objectif qu'il conviendra de se fixer se rapporte à la mise au point initiale des fonctions opérationnelles sur les avions de servitude. En effet l'expérience prouve que plus de 50 % des vols effectués sur les prototypes et les avions affectés au développement concernent la mise au point de ces fonctions. Compte tenu du coût de mise en oeuvre de ces avions, il est indispensable de prévoir pour l'avenir de faire le maximum d'essais sur des moyens aériens moins coûteux.

4. MOYENS A METTRE EN OEUVRE (EXEMPLE DU PROGRAMME ACT RAFALE)

Ce paragraphe se propose d'examiner les différents moyens qui permettraient de se conformer à la méthodologie décrite précédemment. Cette présentation des moyens d'essais s'appuie en partie sur un ensemble de réflexions menées dans les services d'essais en vol français dans le cadre du programme d'avion de combat RAFALE D.

4.1 - Moyens sol

Un des premiers travaux des équipes chargées des essais en vol va être, au vu des spécifications initiales de l'utilisateur, de préciser grâce à un travail d'analyse fonctionnelle, le besoin opérationnel. Un des outils le plus rentable pour effectuer ce travail est sans nul doute la simulation pilotée.

En France, la tendance est de faire coexister deux types de moyens de simulations pilotées, à savoir : des moyens dits "légers" d'aide à la spécification et par opposition des moyens dits "lourds" plutôt dédiés aux travaux d'évaluation.

Pour des moyens légers l'environnement du pilote est limité aux visualisations et aux commandes. Ces moyens très utiles doivent être mis en place chez les industriels qu'il s'agisse de l'avionneur ou des équipementiers. Ces moyens permettront aux industriels d'affiner la définition de leur matériel en présentant à l'avionneur ainsi qu'aux utilisateurs les concepts qu'ils proposent. Une autre utilisation de ces moyens sera de mettre au point les modules informatiques correspondant à la modèlisation des différents équipements qui seront intégrés ultérieurement dans les moyens "lourds". Ce point nécessitera d'assurer la compatibilité informatique des différents moyens afin de pouvoir effectuer les transferts des modèles entre ces moyens.

Les fonctionnalités d'un moyen dit "lourd" seront les suivantes :

- Il doit assurer au pilote un environnement plus représentatif.
- Il doit être conçu de façon à pouvoir intégrer les différents modèles en provenance des moyens "légers".
- Il doit permettre d'exploiter au mieux les essais qui s'y déroule grâce à un ensemble de moyens analogues à ceux utilisés pour les vols.
- Il doit éventuellement pouvoir être couplé aux bancs sol afin de pouvoir stimuler de façon cohérente les composantes du système.

En ce qui concerne les capteurs comme le radar, l'optronique, etc... et en amont de la réalisation des équipements eux-mêmes, un moyen très puissant de mise au point des traitements internes est la stimulation de modèles informatiques. Cette méthode nécessite bien sûr de disposer d'enregistrements en vol. Cette exigence impose de se doter de bases de données obtenues en vol à l'aide de maquettes de capteurs développées suffisamment tôt pour ce besoin. Cette exigence impose également de disposer de moyens d'enregistrements à bord, autorisant des débits très importants pouvant atteindre plusieurs dizaines de Mégabits par seconde.

Les autres moyens sur lesquels il convient de faire des efforts importants sont les bancs au sol. Ces moyens concernent un grand nombre des équipements constituant le système, ainsi que l'ensemble du système lui-même. En ce qui concerne les équipements, et plus particulièrement les capteurs, les bancs de mise au point au sol devront répondre à un certain nombre de critères selon une architecture qui pourrait s'inspirer de celle qui est présentée sur la planche 4. Leurs principales caractéristiques seront :

- Possibilité d'être stimulable, c'est-à-dire possibilité de pouvoir injecter le plus en amont possible les données enregistrées à bord, afin de pouvoir répéter autant que nécessaire les passes réalisées en vol et optimiser par conséquent les traitements internes à l'équipement.

- Architecture permettant de simuler l'environnement avec lequel l'équipement devra dialoguer, afin de valider avant les phases d'intégration les échanges que cet équipement devra assurer.

En ce qui concerne les bancs globaux dits d'intégration, leur architecture devra, en s'inspirant de celle des moyens existants, inclure en plus des capacités de simulations, afin de pouvoir explorer les différentes configurations qu'il est possible de rencontrer en vol. Il sera sans doute souhaitable de prévoir également le couplage de ces moyens à ceux présentés précédemment.

Pour en rester dans le domaine des moyens sol, il est indispensable de parler ici des moyens d'exploitation des essais en vol. Deux grandes familles de moyens sont envisagées selon que l'on s'attache à suivre et à exploiter les essais en temps réel ou à exploiter en temps différé.

Dans le premier cas, qui correspond surtout aux essais d'évaluation et d'intégration sur avions prototypes, les moyens de traitement en temps réel sont bien identifiés et existent aujourd'hui. Ils font appel à des moyens de transmission par télémesure et à de puissants calculateurs temps réel.

En ce qui concerne les exploitations en temps différé, correspondant plustôt aux essais de mise au point d'équipements ou de partie de système sur avions de servitude, les moyens d'exploitation devront répondre à un certain nombre de critères.

Ces critères sont les suivants :

- Capacité de réaliser dans des délais brefs des fichiers de données fusionnées, à partir de données issues de plusieurs moyens de mesure selon des formats différents, comme par exemple les différents messages enregistrés à bord du ou des avions participant à l'essai, ainsi que les messages issus des moyens de trajectographie, etc...
- Possibilité de fournir très rapidement à l'ingénieur d'essais un historique du vol, afin de lui permettre de porter un jugement qualitatif sur la réalisation de l'essai et de programmer les vols suivants.
- Possibilité pour tous les intervenants d'avoir accès dans les meilleurs délais aux fichiers de données qui les concernent en en garantissant la confidentialité.

Le Centre d'Essais en Vol, s'est mis en mesure de satisfaire ces objectifs sur ses différentes bases, en faisant appel à un système décentralisé par réseau et qui devrait être opérationnel en 1989 (projet MATISSE). Ce projet est présenté sur la planche 5.

Enfin pour en terminer avec les moyens d'essais au sol, il convient de ne pas oublier un certain nombre de moyens importants, indispensables à la mise au point des différentes composantes du système d'armes. Sans vouloir être exhaustif on peut citer les moyens suivants qui dans le cadre du programme ACT RAFALE D sont en développement :

- Les moyens de trajectographie multicibles couplés aux systèmes de suivi des essais en temps réel.
- Les moyens de mesure des signatures électromagnétiques et infra-rouge des avions en vol.
- Les moyens de mise au point des contre-mesures.
- Les moyens d'étude du comportement physiologique du pilote lorsqu'il est soumis aux accélérations brutales.
- ...

4.2 - Moyens aériens

Ce paragraphe a pour but de parler des moyens aériens et plus particulièrement de l'architecture de l'installation d'essais dont ils pourraient disposer afin d'améliorer leur productivité, ceci dans le cadre des réflexions exprimées précédemment.

Dans le cadre du programme ACT RAFALE D, le Centre d'Essais en Vol en accord avec les Services Techniques Officiels et les Industriels concernés a proposé d'utiliser un certain nombre d'avions de servitude du type MYSTERE XX FALCON et des MIRAGE 2000. Dans le soucis de réduire les coûts et de respecter les délais, il a semblé indispensable de repenser complètement la philosophie d'utilisation et l'architecture des avions de servitude, par rapport à ce qui se faisait pour les programmes antérieurs.

En effet, compte tenu du nombre limité d'avions prototypes et de leurs coûts d'exploitation, la méthodologie proposée consiste à effectuer sur les avions de servitude, en plus des essais de mise au point des équipements, des essais de mise au point d'éléments critiques de fonctions opérationnelles. Pour atteindre ces objectifs, l'architecture de l'installation d'essais de ces avions, dont le principe est présenté sur la planche 6, répond aux caractéristiques suivantes (cas des MYSTERE XX FALCON par exemple) :

- En début de programme, le système avionique de servitude fait appel à des équipements de base éprouvés, d'une bonne fiabilité et d'une bonne disponibilité (au standard MIRAGE 2000 par exemple) car il est bien connu que pendant les phases de développement les équipements prototypes sont rares et chers.

- L'équipement en essais est couplé à ces équipements par son propre système de dialogue (1553 B, DIGIBUS, STANAG 3910, etc...) et dans le cas où le dialogue ne se fait pas selon le même standard, un calculateur spécifique assure une fonction de "passerelle" entre les deux systèmes. Plus tard, au cours du développement lorsque les équipements prototypes sont qualifiés, ils peuvent remplacer leurs homologues.

- Pour les mêmes raisons et afin de ne pas s'imposer la présence physique de tous les équipements ou armements participant à une fonction ou une sous-fonction étudiée, un calculateur particulier assure la fonction de simulations de ces équipements.

- L'installation de mesure est dimensionnée pour satisfaire les besoins des équipementiers et de l'avionneur, en particulier en ce qui concerne les enregistrements des données à grand débit, ainsi que pour assurer en temps réel le suivi de l'essai à bord par l'ingénieur d'essais.

- L'avion est aménagé de telle façon que la place du co-pilote de droite puisse recevoir l'ensemble des équipements de dialogue homme-machine. Cette caractéristique permet très en amont de faire participer les pilotes aux travaux de mise au point, ceci en coordination avec les travaux effectués sur simulateurs.

Pour les avions prototypes, la règle sera d'essayer de minimiser les essais d'intégration, c'est-à-dire les essais de mise au point de fonctions opérationnelles. Ces avions devront être réservés aux essais qui ne peuvent pas être menés sur d'autres moyens comme par exemple les validations finales : des conduites de tir, des fonctions de navigations, d'efficacité des contre-mesures, etc.. et les évaluations de l'ensemble du système d'armes.

5. CONCLUSION

En concl[illegible]ion, ce document a pour but de convaincre de l'impérative nécessité de repenser les méthodes d'essais en vol des systèmes d'armes fortement intégrés, afin de réduire les coûts correspondants qui prennent de plus en plus d'importance dans le financement total d'un programme.

Les méthodes pr[illegible]es à l'heure actuelle en France, pour les essais en vol des systèmes des avions de combat, comme [illegible]gramme RAFALE, s'inspirent de celles qui ont été présentées, en particulier en ce qui concerne le[illegible] sol et la philosophie d'utilisation des avions de servitude.

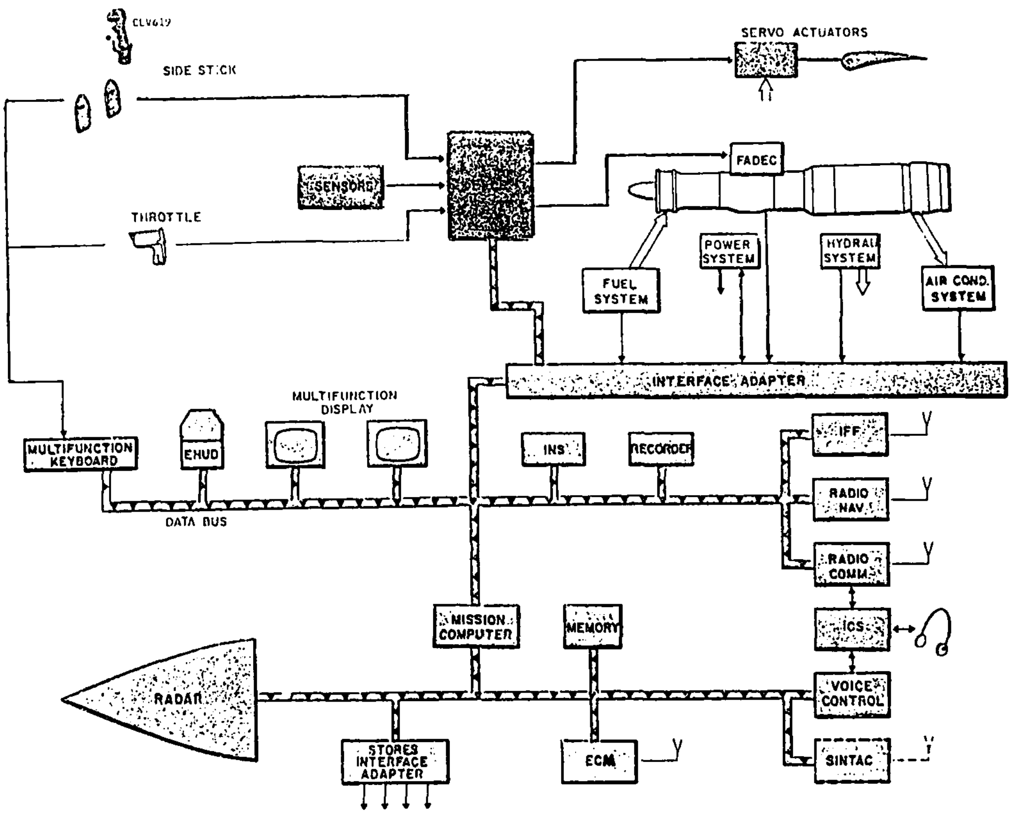

SIDE STICK
SERVO ACTUATORS
FADEC
SENSORS
THROTTLE
POWER SYSTEM
HYDRAU SYSTEM
FUEL SYSTEM
AIR COND. SYSTEM
INTERFACE ADAPTER
MULTIFUNCTION DISPLAY
MULTIFUNCTION KEYBOARD
EHUD
INS
RECORDER
IFF
RADIO NAV
RADIO COMM
DATA BUS
ICS
MISSION COMPUTER
MEMORY
RADAR
VOICE CONTROL
STORES INTERFACE ADAPTER
ECM
SINTAC

PLANCHE 1

COMPLEXITE DES SYSTEMES

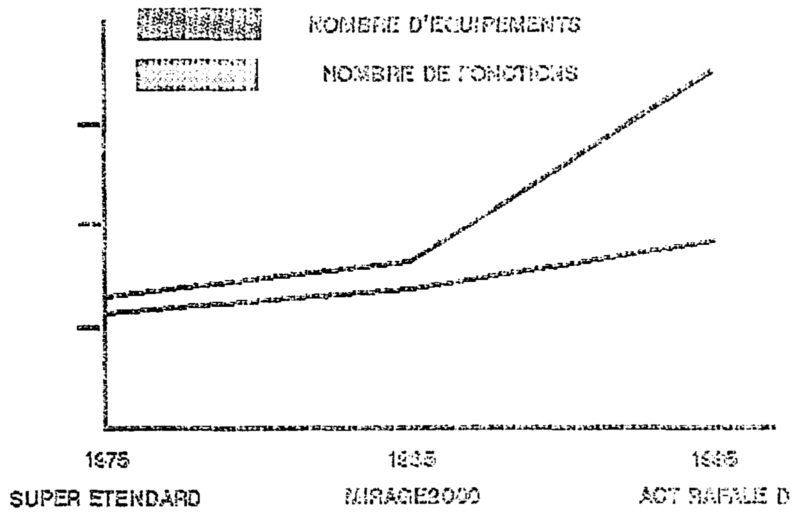


PLANCHE 2

EVOLUTION DES COUTS DES ESSAIS EN VOL

(EN MF 1987 PAR AN ET PAR AVION)

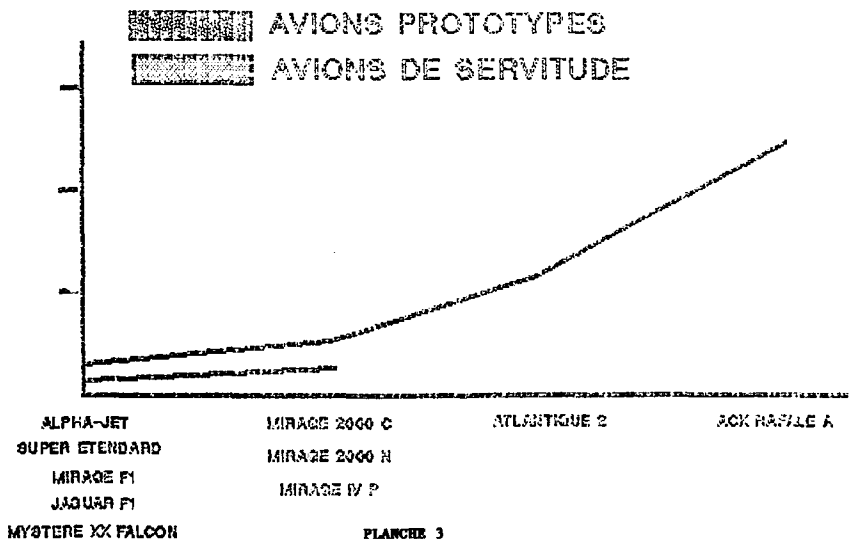


PLANCHE 3

BANCS SOL

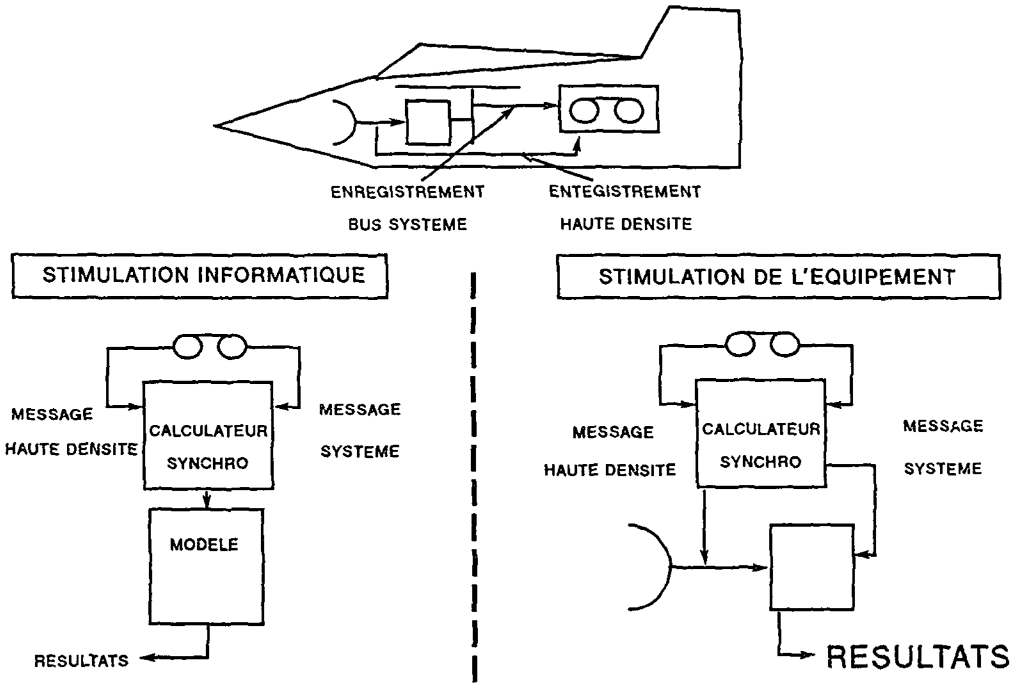
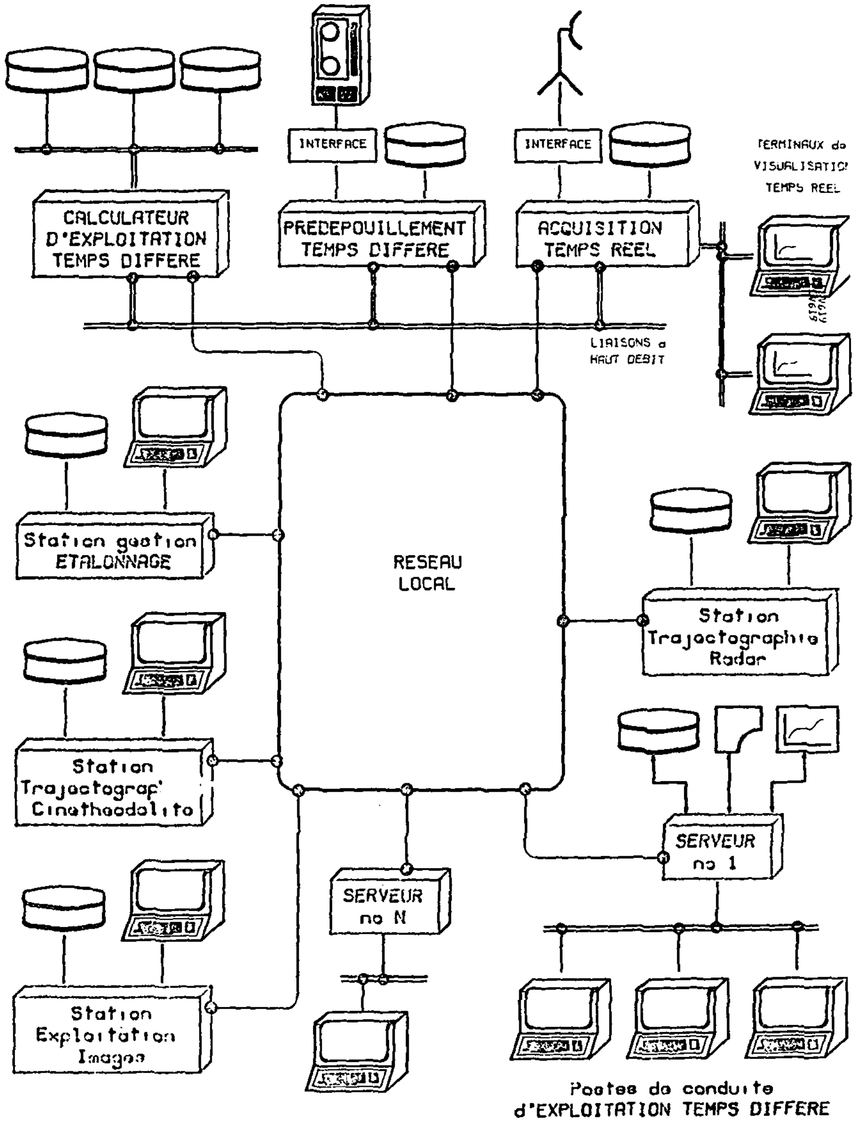


PLANCHE 4



SYSTEME D'INFORMATIQUE
DISTRIBUEE ET INTERACTIVE

PLANCHE 5

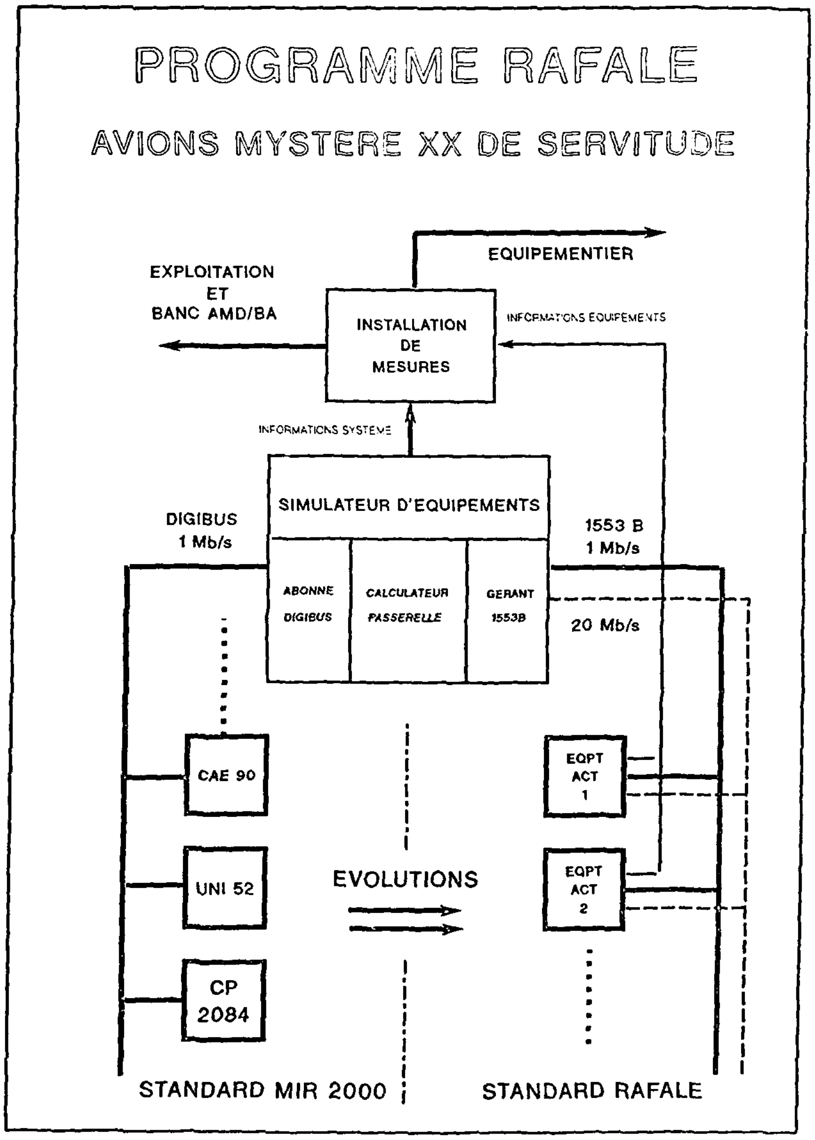

PROGRAMME RAFALE
AVIONS MYSTERE XX DE SERVITUDE
EQUIPEMENTIER
EXPLOITATION ET BANC AMD/BA
INSTALLATION DE MESURES
INFORMATIONS EQUIPEMENTS
INFORMATIONS SYSTEME
SIMULATEUR D'EQUIPEMENTS
DIGIBUS 1 Mb/s
1553 B 1 Mb/s
ABONNE DIGIBUS
CALCULATEUR PASSERELLE
GERANT 1553B
20 Mb/s
CAE 90
UNI 52
CP 2084
EVOLUTIONS
EQPT ACT 1
EQPT ACT 2
STANDARD MIR 2000
STANDARD RAFALE

PLANCHE 6

FLIGHT TESTING OF THE TORNADO TERRAIN FOLLOWING SYSTEM

Thomas Fleck

Flight Test
Messerschmitt-Bölkow-Blohm GmbH
Military Aircraft Division
P.O. Box 801160
D-8000 München 80, West Germany

SUMMARY

This paper reviews the flight test activities on the assessment of the Tornado Terrain Following System.

Extensive hardware and pilot in the loop simulation, a stepwise approach to the lowest height and proceeding from VMC to IMC kept the risk at a minimum level.

For a demonstration of system performance a method of comparing simulation had been applied that considers avionic parameters, aerodynamic parameters and flight control data.

Special emphasis was placed on system integrity and flight safety,including human factors of the aircrew.

Today, after having absolved more than 200 successful testflights and rectified a considerable number of problems, the system is adopted by the services and is a substantial part of the low level missions.

INTRODUCTION:

The Tornado aircraft, a trinational development, is a fighterbomber with variable wing geometry. It' s development was the answer on the military requirements in the late '60's' and early '70's'.
One of these requirements was lowest penetration and attack height at high speeds in all weather, day and night and over all types of European terrain.
The only way to satisfy this demand was the implementation of an automatic Terrain Following System.
The demonstration of system performance and integrity required the development of evaluation methods that especially considered the dynamic nature of the terrain following mode.

This paper gives an overview of the flight test activities on this subject and adresses the selected methods used to determine the system performance.

SYSTEM DESCRIPTION:

Terrain Following is one of the AFDS-modes to be selected at the AFDS-control panel.
The control loop consists of the TF-radar, the radar processor, the Autopilot and Flight Director Computer and the Command and Stability Augmentation System (CSAS).
The TF-radar scans the terrain ahead in elevation and azimuth.
The TF processor calculates climb/dive commands that are linked to the AFDS in form of vertical acceleration demands.

Commands are derived from a zero command line shaped like a ski toe depending upon the following factors:

* selected clearance height
* aircraft speed
* selected ride mode

In manual mode the commands are displayed on the Head Up Display (ADI as a backup) to be followed by the pilot manually, or in automatic mode the AFDS processes rate demands to the CSAS. (Fig. 1)

Additional sensors providing information for flight path computation are:

* Radar Altimeter
* Inertial Navigation System (IN)
* Secondary Attitude and Heading Reference System (SAHRS)
* Doppler
* Main Computer

The basic pnnciple of the command generation is the comparison of target returns with the zero command line. Targets penetrating the skie toe will cause a climb command, obstacles below will result in a push over.

AUTOPILOT AND FLIGHT DIRECTOR SYSTEM

The AFDS consists of two digital computers that are identical in software and both computing the control laws and the mode switching and failure logic.

The hardware is slightly different depending on the distinct application of each computer, hence they are not interchangeable.
Computer No. 1 converts the output pitch and roll rate demands to analogue signals. The output hardware is triplex and linked to the CSAS. (Fig. 2)

Monitoring of the AFDS output forms part of the CSAS.
Computer No. 2 drives the Flight Director. Both Computers monitor each other by permanent comparison of the demand computation path.
A difference exceeding a certain threshold causes the autopilot to disconnet.

Part of the control laws are variable g-limits, rate limits and bank anlge limits, depending on speed, wingsweep and altitude.
An emergency fly up is desinged for safe recovery in TF-mode, when the autopilot is cut off by the failure logic.

DISPLAYS:

Monitoring of the control loop is essential for pilots confidence in the system, particularly in IMC.

Therefore a socailed e-scope provides a display of radar returns over the skie toe relative to the range ahead measured in miles. (Fig. 3)

An additional Clearance Range Ahead Monitor (CRAM) line above the zero command represents the locus of a constant g pull up to clear obstacles at set clearance height.

The CRAM computation is independent from the command generation and forms part of the TF failure logic.

Once the CRAM line is penetrated by a target, the TF system transmits a fail event to the autopilot which then reacts with an emergency fly up.

Mapping radar videos can be displayed in the repeater mode.
It must be stressed, that e-scope videos are only used as a monitor.
Flying by interpretation of the video will essentially degrade the TF-performance.

INTEGRITY:

Due to the extreme requirements in terms of flight safety for low level flying, the system was designed to be fail safe. Any system failure that could have an influence on the flight path will result in a fly up, either closed loop or open loop, depending on the type of failure.
Redundance is guaranteed by duplex autopilot and triplex CSAS. The TF command path is extensively monitored by independent sensors, providing warnings if absolute limits are exceeded (low height, unobeyed command, excessive turn rates and bank angle). (Fig. 4)

PHILOSOPHY OF TESTING

Testing of such a highly integrated system with outstanding impact on flight safety required a very careful approach and the development of test methods that could cope with the various aspects of safety and performance.

Thus, the Tornado TF/AFDS flight test program was seperated into four main items:

* System integration
* Performance testing
* Failure testing
* Simulation

In 1973 the TF system was installed in two Buccaneer aircraft at British Aerospace, to allow realistic testing at a prototype stage.

Main tasks of the trials were:

* Integration of TF-radar, Main Computer and Sensors
* Assisting the development with flight test results at an early stage prior to Tornado flight tests.
* A first assessment of the AFDS control loop by Flight Director verification.

Later on these trials were still supporting Tornado flight tests.

When Tornado flight test began, the TF-system was already developed to a status, at which all basic functions had proven to be adequate. The Tornado TF-trials started in 1976. This was the first time, the system was flown fully automatic.
A special flight test program was devoted to TF/AFDS system integration. Until 1978 test results were used to assist further development and rectification of system shortcomings. Then, since 1978, tests concentrated on performance assessment, first with pre-series and later on with production equipment.

150 flights were allocated to demonstrate adequate performance and operational effectiveness in:

* Manual TF
* Automatic TF
* TF in combination with lateral AFDS modes (Heading and Track acquisition) and Auto Throttle.

FLIGHT TEST FACILITIES

Flight test instrumentation

The Tornado testaircraft are equipped with a PCM recording system, covering both, analogue and digital parameters. Sample rates were up to 32 sps in the beginning of flight test.
Problems occured, when the evaluation required a high resulution to assess TF and AFDS processor data.
Methods had to be developed that provided essentially higher sample rates. This was achieved by the direct recording of the data transmission lines (DTL) on separate tracks of the recorder tape. At present sampling rates of maximum 2000 Hz are available.

Telemetry

During a test flight all parameters and events sampled by the FTI are transmitted to the ground station via telemetry and recorded on tape, used as a back up in case the onboard recorder fails.
A limited number of parameter or events can be displayed on monitors or multi channel plotters.
One monitor is reserved for a synthetic graphic of the head up display, AFDS control panel and central warning panel.
Being permanently computed by oncoming telemetry data, the test engineer has a realistic picture of selected AFDS modes, warnings, and a complete and dynamic HUD-indication. (Fig. 5)

Ground station facilities

The MBB Flight Test Center is equipped with a comprehensive computer system which includes the flight telemetry monitor system, complex data aquisition and online/offline data evaluation. (Fig. 6)

Trajectory tracking can be performed by:

* Cinetheodolites (stationary and mobile)
* Precision Radar tracking (AN/MPS 36)
* Laser Precision Aircraft tracking System (PATS)

All stations are syncronized, so that hand over from one system to the other is possible.
Data are recorded, processed and automatically analyzed.

PERFORMANCE ASSESSMENT:

TF routes:

When the flight test program was established, a precondition was, that tests had to be performed over defined test routes to guarantee a high degree of reproducability.

These routes had to represent all but alp-type European terrains including different types of vegetation, land-sea crossings and man made obstacles.

A coverage of all routes by Cinetheodolites ano tracking radar was considered impracticable due to the extensive effort that would be involved.
Other more effective methods to determine the flight path over the terrain had to be established.

Only for calibration purpose, a number of simulated runs over the airfield were tracked by Cinetheodolites.

Terrain Profile Reconstruction

In the early stage of system development a two dimensional terrain profile picked up from a map of a TF-route was the basis for the flight path simulation. The idea was to use exactly these tracks for flight test.
At TF routes without significant land marks or obstacles, beacons were installed to improve the track accuracy.
However, during the Tornado performance assessment phase this system was not suitable, since the navigation accuracy necessitates extensive effort to achieve adequate results.

The following analysis method has been adopted to assess the system performance:

1. Reconstruction of the terrain profile by flight test data
2. Computation of the flight path by flight test data and assessment of the peak/vally performance
3. Comparison with a simulated flight path over identical terrain
4. Evaluation of the difference between actual and simulated flight path

A good approach was to compute the terrain profile by flight test data.
Basically a terrain profile can be determined by subtracting the radar height from the barometric height.
For sufficient accuracy, correcting factors must be taken into account, such as:

- Air pressure
- Air temperature
- Height of test range
- Lags in Radar Altimeter and ADC measurements
- Radar altimeter beam width

The terrain thus obtained is two dimensional and will be used to display terrain profiles with the aircraft flight path above, both actual and simulated. (Fig. 7)

Together with relevant flight test data of aircraft and system parameters, the terrain profile is transcribed on a tape No 1. Then the terrain profile is fed into the flight path simulator from a tape N° 2 and simulated parameters of the TF/AFDS loop are recorded on a tape N° 3.

The information on tape N° 1 and N° 3 are now compared by a computer program with the following possibilities:

- Comparison of system parameters in relation to the overflown terrain
- Comparison of aircraft flight path with a simulated flight path
- statistical evaluation of the difference between flight test and simulation

The computer program compares simulation and test results along the flight path at 50 m intervals.
Hence, a TF route of 20 km length yields 400 samples to be compared. (Fig. 8)

Tests have shown, that the error band of this comparing system increases with the terrain roughness.

Some factors that contribute to this phenomenon are:

- Tolerances within the FTI system, calibration errors of parameters
- The terrain reproduction has a certain error band due to the radar altimeter beam width. Within the antenna beam, the shortest distance to ground is measured.
- By the two dimensional terrain reproduction, obstacles off the track but within the radar azimuth scan angle are not considered in the terrain profile computation.
  Hence, the TF antenna is detecting targets not directly in line with the flight path.
- Errors within the AFDS feedback loop:
  The AFDS triplex output contains a most nose up logic that may cause a slight fly high effect.

To present the radar detection performance a computer program was developed that combines the flight path with radar range and scan angle data.
These plots are helpful in analysing the detection of small sized obstacles or to verify the detection of non-existing targets caused by beta-spikes. (Fig. 9, 10)

Assessment of the Peak/Valley Performance:

The height over peaks ideally should be equivalent to the set clearance height. Hence peak values can be derived from Radar Altimeter data, which however might be difficult for small sized obstacles.

The height above valleys is depending on speed, selected ride mode and the surrounding terrain. A performance assessment requires the comparing simulation.

FAILURE TESTING

Prior starting IMC trials, pilots had to build up confidence in the system. In IMC the only possibility to monitor the system response is to rely upon cockpit instruments and warnings (Fig. 11), considering the E-Scope as the primary instrument.
Therefore an extensive failure testing was a prerequisite:

* System response on failures
* effectiveness of cockpit warnings
* mishandling
* failure injections
* cross software tests
* simulation of worst cases
* EMC testing

System Response on Failures

An essential part of flight testing was the assessment of the open loop pull up, a function that takes place when the autopilot disconnects due to a system or sensor failure.

Being designed to initiate an automatic fly up with g-loads between 3 - 4 g depending on speed and wingsweep, flight test results have shown, that under certain circumstances i.e. high aircraft mass and aft c.g. at corner points of flight envelope, the aircraft alpha-limits or max. permissable g's can be exceeded. (Fig. 12)

The flight envelope cleared for TF flying had to be reconsidered for each possible store configuration. mainly done by closed-loop simulation.

Cross Software Testing

Cross software testing was especially developed for the digital Autopilot and Flight Director System. Although a redundant system, errors in the identical software could cause hazardous failures.

To minimize the risk to an acceptable level, following method was adopted:

An AFDS software model is programmed in a high order language by an independent software team on a separate computer system.

Then an automatic program stimulates both, the software model and AFDS computers within and beyond the range of system limits. By comparing the outputs, a failure will be detected with a high degree of probability. (Fig. 13)

EMC-Testing

EMC can have an influence on fly-by-wire control systems. For example, in Germany there are several powerful broadcasting stations and when flying in the vicinity of these transmitters, the aircraft can be exposed to quite significant field strength levels.

Therefore, EMC testing is of vital importance for the flight safety of the aircraft.

Assessment of the susceptibility level of the flight control system was carried out in three ways:

- Bulk current injection (BCI) with stepwise increasing Rf power into the aircraft wiring.
- Illumination tests where the aircraft was exposed to a defined RF-field on ground. (Fig. 14)
- RF-station fly by.

The FCS was modified such that although flying mechanical mode, a full fly by wire mode could be selected and monitored.
Analysis of data gave an impression of the susceptibility of the flight control system.

Results of ground tests and flight tests urged the need of EMC-hardening measures, mainly in the analogue CSAS and the analogue AFDS in- and outputs.

PROBLEM AREAS

The beta-spike problem

During the course of flight test, unexplained pull ups occured when approaching solid targets that were far below the set clearance height.
Especially for IMC-flight this situation was unacceptable and had to be solved. Analysis revealed, that the terrain following radar sensed erroneous echos above the ski toe or Cram line and interpreted then as real obstacles. Harsh 3 to 4 g climb commands were generated, followed by 0 g pushovers.

The cause of these spikes was finally traced to a reflection problem from the aircraft pitot nose boom and pitot static tubes within the radome, causing a radar detection via the antenna side lobes.

To overcome the problem, these parts were covered with radar absorbing material.
Together with a sensitivity adjustment of the TF receiver, beta-spikes were eliminated by more than 95%.

Unexplained Autopilot Disconnect

Frequent Autopilot disconnects occured when flying TF, although no significant system - or sensor failure was detected. On ground, the problem could no be reproduced.

Only when the sample rate of the instrumentation was increased such that processor cycles could be displayed, the problem was traced to the autopilot failure logic.

Fail events in the range of one program cycle were summarized and had a combined effect thus tripping the disconnect logic. By changes in the logic structure the problem was solved.

TESTING THE SYSTEM IN MOUNTAINOUS TERRAIN AND BAD WEATHER

In 1983, after having achieved the clearance for flying the TF-system at all speeds and heights in VMC, tests started under IMC-condition.
The testgoal was progressively approached by starting over flat terrain with runs against single cloud formations of different appearance, aiming at an assessment whether the system would identify clouds as an obstacle.
In this context two selectable weather modes had to be tested, one with reduced receiver sensitivity, the other with a limited radar range.

IMC was defined as an inflight visibility of less than 1,5 km. This, of course, required tests in all possible kinds of weather such as fog, rain, snow, drizzle and showers.

To test the worst conditions, a test route over mountainous terrain with a maximum peak to valley height difference of 4000 ft and an overall length of 60 NM was selected.

At this stage, where the system performance as such had already been demonstrated, human factors of the crew like mental or physiological loads were highlighted.
The pilot, having no visual contact to the outside world, spends most of the time for E-scope monitoring.
As a parallel task, he has to watch the instruments for engine-readings, speed, radar altiude, angle of attack, g-load and to select modes at the AFDS control panel. Any unexpected event like unpredicted manoeuvres or warnings will result in an increase of the physiological load. Changes in the cockpit lay out and automatic warning philosophy were indispensable to reduce pilot workload.

Main items were:

- All important indications and autopilot controls to be arranged as far as possible in the head up field of view.
- Optimization of TF-related warnings.
- Aircraft motions must be predictable for the aircrew (nuisance disconnects, beta-spikes)

Flight Director mode, being a simplex system so far, is not cleared for IMC operation.
A presently introduced AFDS upgrade will provide a duplex, self monitoring Flight Director that will be extensively tested in 1989.

IMPROVEMENTS AND MODIFICATIONS

Of course, the majority of modifications had been introduced before starting tests with production equipment. However, with growing experience by aircrews and engineers,modifications in form of system upgrades were introduced to cover technical and operational aspects.

Important changes being subject of present flight testing are:

- Substitution of the head down AFDS control panel by a head up status and selector panel.
  This was a strong aircrew requirement out of the IMC trials with respect to flight safety.
- Introduction of a split axis control mode (SAC)
  This modifiation is part of an AFDS upgrade and allows the pilot to fly the aircraft laterally by stick inputs while in pitch the aircraft is still automatically controlled.
- Implementation of a duplex Flight Director System.
- EMC hardening measures.

CONCLUSION

Flight testing of the Tornado TF-System has shown, that computer aided analysis yields independence from special test ranges with sophisticated ground equipment and time consuming preparation of tests.

Numerous problems had to be solved before the system was up to an adequate standard fully developed to be offerd to the services.

Performance has been demonstrated under "worst condition" with continous operation in excess of one hour under:

- full IMC
- rough mountainous terrain
- lowest set clearance height
- 0.9 Mach, 66^{0} wingsweep

The system is successfully used by the services of three nations since seven years and has proven to be reliable and safe.

LIST OF ABBREVIATIONS

ADI	Attitude Director Indicator
ADC	Air Data Computer
AFDS	Autopilot and Flight Director System
CSAS	Command and Stability Augmentation System
CP	Control Panel
CRAM	Clearance Range Ahead Monitor
CWP	Central Warning Panel
EMC	Electro Magnitic Compatibilty
E-Scope	Elevation Scope
FTI	Flight Test Instrumentation
G,g	Gz-loads
IN, INS	Inertial Navigation System
IMC	Instrument Meteorological Conditions
LHW	Low Height Warning
PCM	Pulse code Modulation
SAHRS	Secondary Attidtude and Heading Reference System
SAC	Split Axis Control
UCM	Unobeyed Command Monitor
VMC	Visual Meteorological Conditions

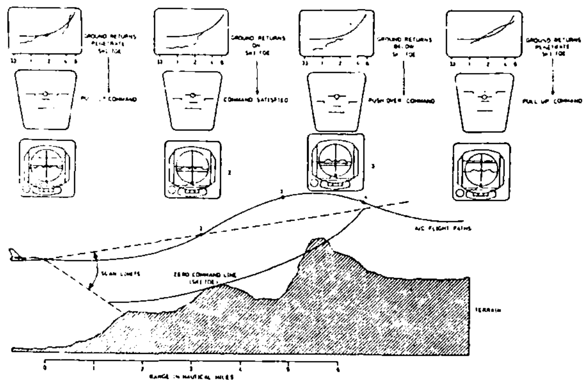


Fig. 1 Flight Profile and Indications

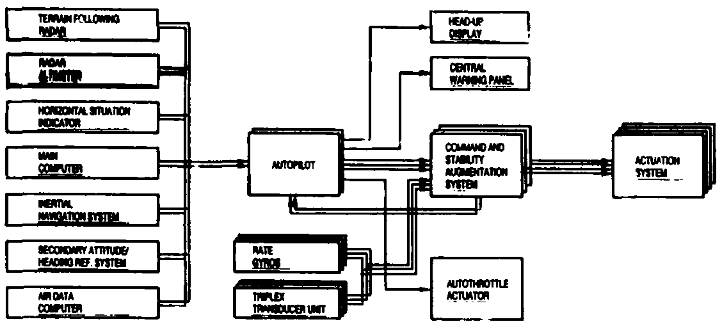


Fig. 2 Block Schematic of Tornado Flight Control System

The ZCL is changing as a function of:

- **Ground Speed V_G**
- **Selected Clearance Height H_0**
- **Aircraft Flight Vector Angle γ**
- **Ride Control**

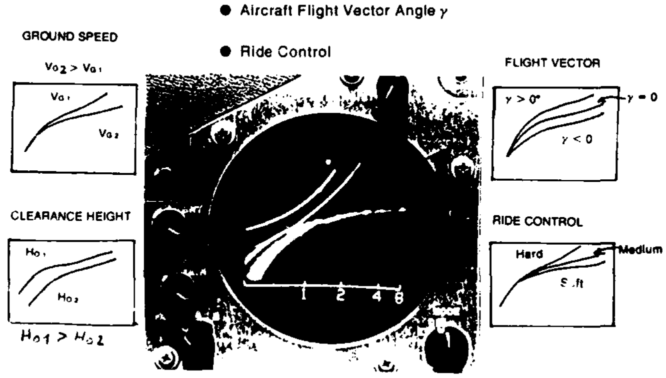


Fig. 3 Dynamic of Zero g-Command Locus (ZCL) on the E-scope

TYPE	COCKPIT FRONT	COCKPIT REAR	VISUAL	AUDIO	REASON
TF FAIL	X	X	RED TFR ON CWP	LYREBIRD	TF DATA GOOD REMOVED
TF MONITOR	X		AMBER TF MON ON CWP	NONE	SOURCE FAULT OR DEVIATED
MANOEUVRE MONITOR	X	X (ONLY AUDIO)	AMBER LIGHT ON COAMING	600 Hz INTER-RUPTED TONE	- UCM - LHW - EXCESSIVE TURN RATE
HT FAIL	X		AMBER LIGHT ON TF CP (WITH TF MON)	NONE	IN SIGNAL TO TFR FAILED
TURN FAIL	X		AMBER LIGHT ON TFR CP (WITH TF MON OR MAN MON)	DEPENDENT ON TF OR MAN MON	TURN RATE OR DRIFT MON

Fig. 4 TF related Cockpit Warnings

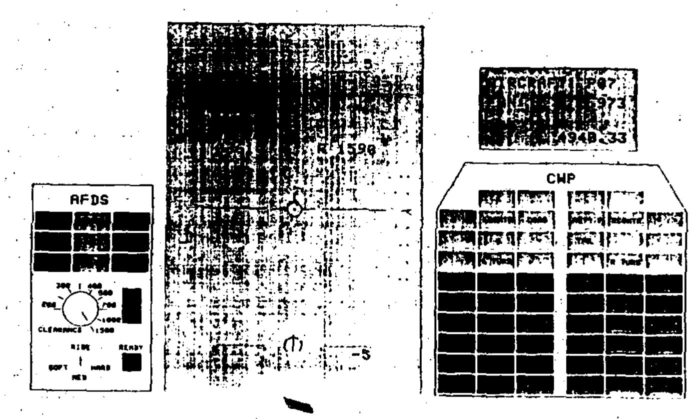


Fig. 5 Graphic Display of HUD, AFDS-Control Panel and Central Warning Panel

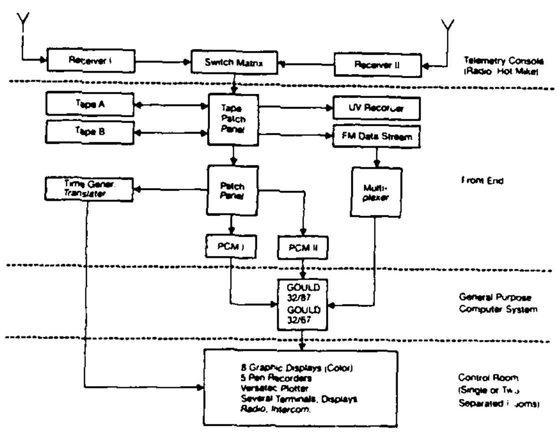


Fig. 6 Schematic of FTI Ground Station

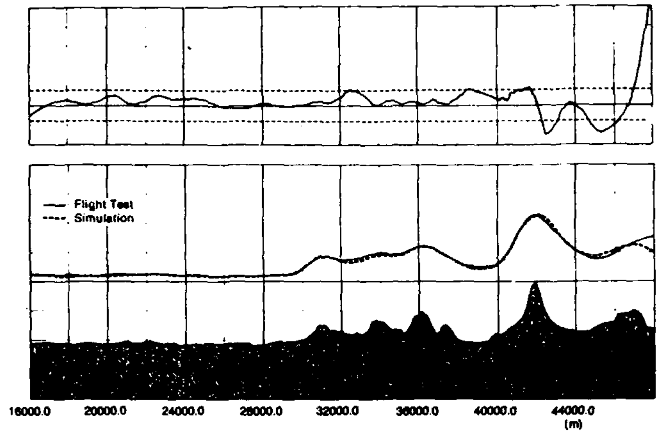


Fig. 7 Comparison of FT and Simulation Results

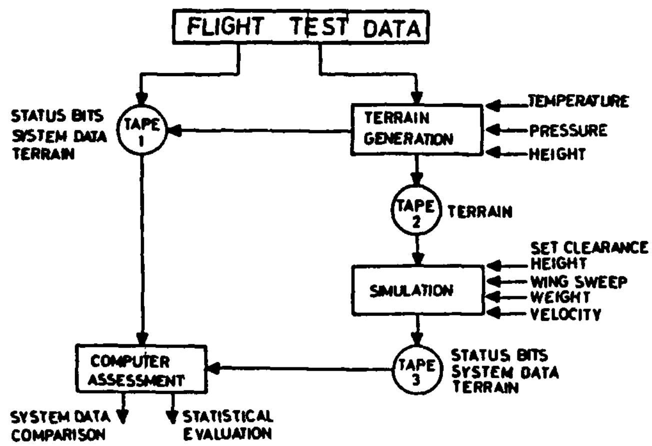


Fig. 8 Data Handling for Result comparison

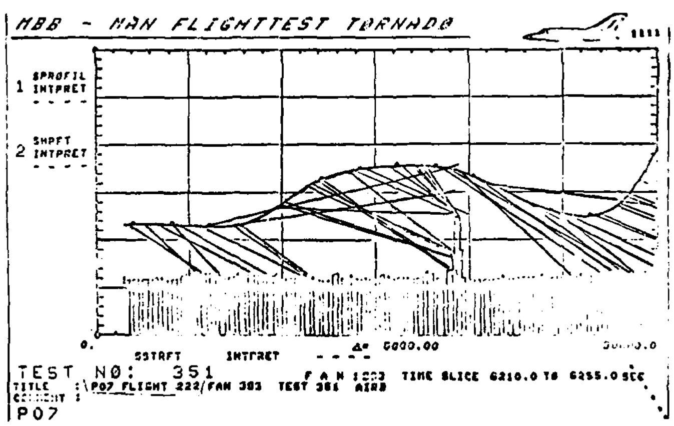


Fig. 9 TF-Radar tracking of Manmade Obstacles

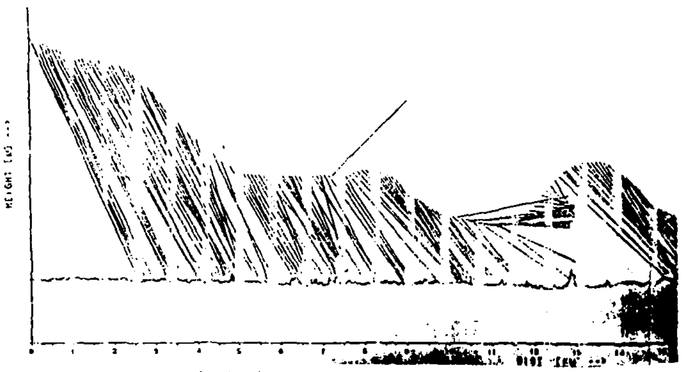


Fig. 10 Radar tracking during stepwise let down Beta-Spike and small sized Obstacle.

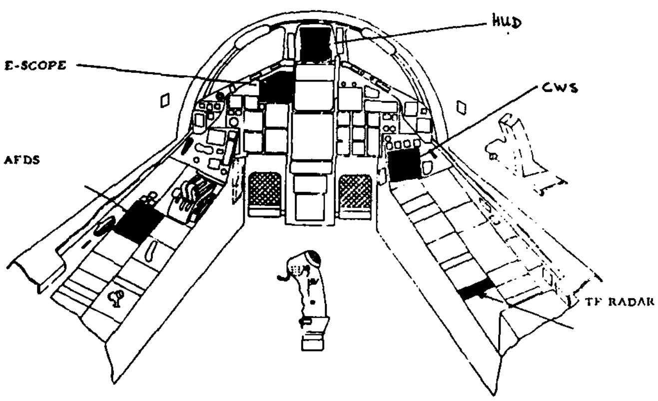


Fig. 11 TF related Indications, Control Panels and Warning Panel

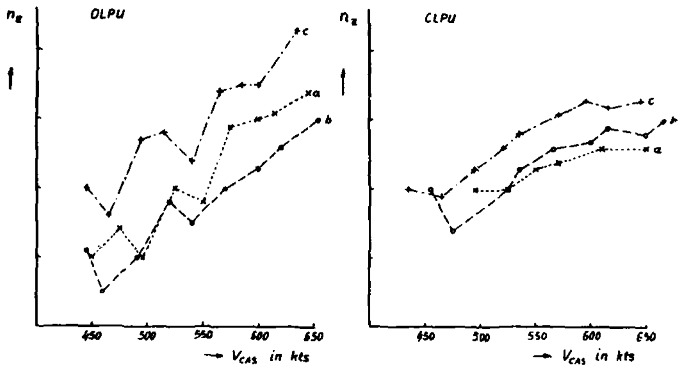


Fig. 12 Peak Values of Normal Acceleration during Open- and Closed-Loop Pull-Up, at 67° Wing Sweep and different c.g.'s.

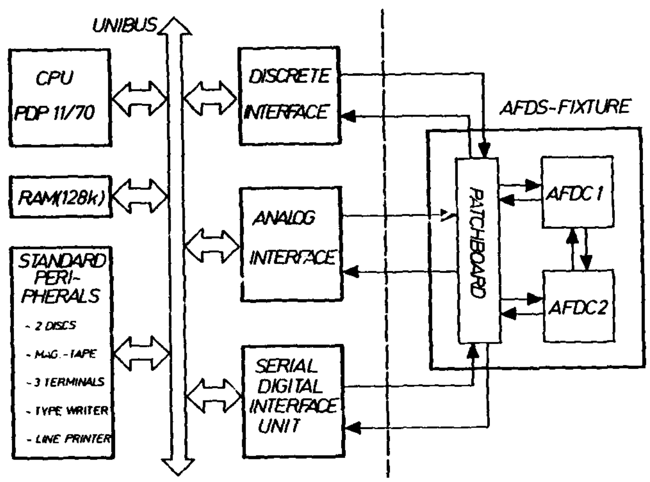


Fig. 13 Block Schematic of AFDS-Cross Software Test

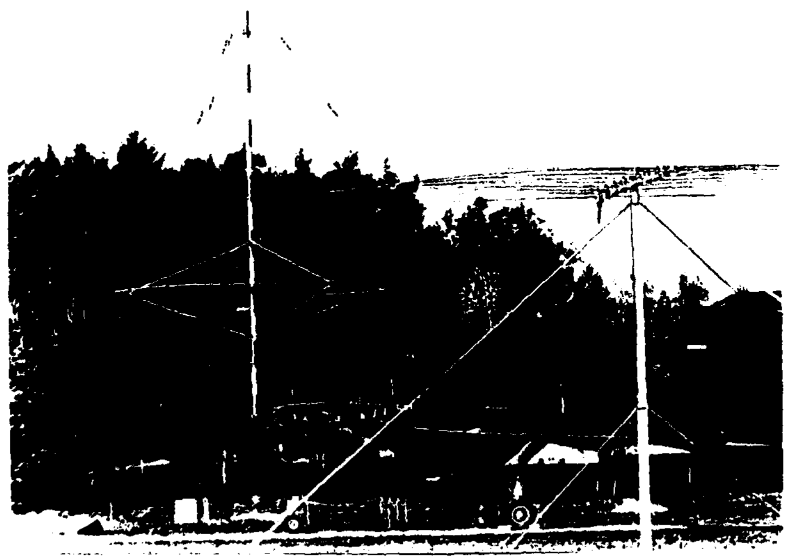


Fig. 14 RF-Illumination Test

EVALUATION OF STORE CARRIAGE (AND RELEASE) ENVIRONMENT USING FLIGHT INSTRUMENTED STORES

by

H.A.Torode and P.J.Driffill
Royal Aerospace Establishment
Farnborough, Hants, GU14 6TD
UK

and

D.Charles
Cranfield Institute of Technology
Cranfield, Bedford, MK43 0AL
UK

SUMMARY

This paper describes the approach adopted by Attack Weapons Department, Royal Aerospace Establishment, Farnborough, in conjunction with Cranfield Institute of Technology for obtaining experimental data pertinent to the carriage and release of conventional free fall weapons. To this end a series of instrumented, flight cleared stores have been developed for various data gathering exercises, and results from several of the flight experiments completed to date, are discussed. In addition some future developments are presented.

1 INTRODUCTION

With ever increasing demands on aircraft performance and the current preoccupation with external weapons carriage it continued to be increasingly important to understand fully the carriage and release environment experienced by the weapon to be understood. Two significant areas, flight carriage loads and vibration, are the subject of continuous detailed investigation by RAE, particularly with respect to the implications on store design and airframe integration. In addition to this research aspect RAE maintains a continuous monitoring brief in support of in-service problems on various weapons and carriers. A close relationship is also maintained with the relevant sectors of operational staff and industry via consultative committees.

To aid this work a range of self-contained instrumented stores were constructed during the mid-seventies. These were capable of sensing and recording aerodynamic loads, both steady state and manoeuvres (up to 3g), and vibration experienced by the weapon during carriage flight. This work previously presented to the AGARD FMP in Reference 1. These instrumented stores have since been developed over a number of years by RAE in collaboration with the Flight Systems and Measurement Laboratories at the Cranfield Institute of Technology (CIT). The RAE held responsibility for the fundamental design concepts of the stores and the overall management of the flight trials while CIT advised on instrumentation aspects and held responsibility for the primary data reduction and analysis of the vibration data.

Section 2 of this paper gives a brief summary of the store status and developments since the presentation in Reference 1. In Section 3 a selection of flight trials results in respect of flight carriage loads is discussed. Results for two specific trials using the load measuring stores are presented. In section 4 vibration data gathering and analysis are discussed and the position in general aspects, (which relate to all rather than to specific aircraft trials) is reviewed. Our future plans for developments in store design and instrumentation are presented in section 5.

2 STORES REVIEW

2.1 VIBRATION STORES

The two original vibration stores V02 and V03, (described in reference 1), both based on the UK 1000lb bomb casing, are still in limited use, but are restricted due to the limitation of recording only six channels of data. These stores use piezo-electric accelerometers for measurement and charge amplifiers for signal conditioning with the results recorded on an internal eight channel tape recorder. An additional store, again using the same shell, has been produced utilising accelerometers which have integral electronics to overcome difficulties experienced with the original instrumentation such as cable noise sensitivity and instrumentation bulk. A multiplexer has been added to increase the flexibility of the recording system and enables additional transducers to be added when required for more detailed investigations such as bomb tail fatigue aspects. The vibration data is recorded, on the stores internal tape recorder, using $^1/_2$ inch tape, over a bandwidth from steady state to 3 kHz.

The vibration data recorded by these stores has been gathered with two basic aims:

a) To provide detailed information leading to the understanding of, and the solution to, in-service problems which occur as a result of changing operating conditions from the original design requirements.
b) To provide data to allow more realistic methods and spectra for ground testing to be devised and, incidentally, to isolate the various parameters affecting store vibration levels. Subsequently guidelines for developing more efficient designs have been formulated.

2.2 AERODYNAMIC LOAD STORES

Two loads stores designated A01 and A02 have been designed around a Patushkin Aerodynamic Load Balance which measures the relative deflections between two plates by means of six strain gauge links, Figure 1. One plate is attached to the aircraft, via the weapon suspension equipment, and the other is attached to the store skin. The aerodynamic loads and moments are derived from the strain gauge outputs, by means of a calibration matrix, from the links through which the plates are coupled. It should be noted that roll is derived from the differences between the two forward elements of the vertical load links. One advantage of this design is that the store skin is attached to the balance plate by two rings, this enables the skin to be light to minimise inertia effects. This attachment system permits these balances to be readily embodied into a variety of weapon shapes on the present UK inventory. Currently body shapes are available for the in-service 1000lb with the No 114 Tail and BL755 weapons.

These stores are totally selfcontained as they have internal battery power supplies and tape recorder. The only aircraft interface is the recorder operation, usually controlled from the cockpit. This enables them to be fitted on most store locations on all service aircraft.

The 'handed' sign convention shown in Figure 2 has been adopted so that the axes frame used corresponds to that of the aircraft. This has the further advantage that the recorded loads correspond on both sides of the aircraft centreline.The moment reference centre of the store is on the centreline at the mid-point of suspension. This is also necessary for comparison purposes between various weapon shapes which have differing Centres of Gravity.

The 'aero' stores are carried to measure the aerodynamic loads, resulting from the aerodynamic flow field around the aircraft/weapon combination, to provide data which may be used in the following areas:-

a) Detailed investigation of the aerodynamic loads and moments experienced by the weapon during carriage flight and the implications on release disturbance for current in-service aircraft.
b) The generation of a UK data base of weapon release parameters. This data base is currently in use notably providing information for the validation of current prediction methods for release characteristics of weapons.

2.3 USAGE

A large number of assignments have been completed undertaking a wide variety of tasks and investigations. Amongst those which have been completed most recently are the systematic investigations of the fatigue environment encountered by a store during carriage on current in-service aircraft. In support of this aerodynamic loads have been measured during low level flight and a variety of manoeuvres. Two tasks, typical of such studies are now described in detail.

3 AERODYNAMIC STORE LOADS STUDIES

3.1 HARRIER - TWIN STORE CARRIER

A series of flight trials have been conducted on the Harrier aircraft to investigate store carriage loads on Twin Store Carriers(TSC). Mathematical modelling had predicted, Reference 2, that increased store separation could be expected to alleviate the large forces and moments contributing to the release disturbance environment on this aircraft when carrying two large stores on the inboard wing pylon. To validate these predictions a trial was mounted using an instrumented stores to gather results from the conventional inservice TSC and a new twin store carrier, designated YDC, modified to increase the separation between the stores, Reference 3. The physical differences between the carriers used are shown in Figure 3.

Figures 4a-e show the results for 1000lb bomb shape carried on both inner and outer positions on the conventional TSC and YDC. In the TSC configuration the axial and normal force increase with increasing speed while side force appears to remain steady and is less sensitive to changes in speed. The pitching moment, nose down, tends to increase linearly with speed and does not exhibit any dependence on store position on the carrier. Noteably the yawing moments increase with speed and are of opposite sign depending on store position indicating that the tails are coming together with the possibility of the tails colliding during the release of the first store.

The results from the angled carrier (YDC), again for both store positions, demonstrate the same trends as the conventional TSC, but with a significant reduction both in nose down pitching moment and the mutual yawing attraction of the tails.

This trial validated the new aerodynamic design showing that increased separation of the stores on the carrier has a beneficial effect on the carriage environment and in particular the reduction of both pitching and yawing moments. In our experience release disturbance is related to the magnitude of the pitching and yawing moments experienced. Subsequent trials have demonstrated that the release of the inner store (first release, and therefore in the presence of the outer) has represented a clear improvement over the earlier TSC. As a result of the trials evidence the YDC carrier has been put into production and has entered service as the CBTS 30.

3.2 TORNADO - RELEASE DISTURBANCE DATA BASE

The RAE is currently performing a series of research flight trials using its Tornado test aircraft, Figure 5a, to provide full scale aerodynamic data for use in validation of the wide variety of predictive techniques currently available in carriage and release modelling. Heretofore such predictions have generally been compared, one against the other, to demonstrate their validity, in the absence of comprehensive and reliable flight data. At RAE, both Aerodynamics and Weapons Departments concluded that it was important that such techniques should be tested against "real world" data as the ultimate test. These trials were planned to make available a comprehensive data set using all current carriage and release data gathering techniques including the Aero stores as just described to measure carriage loads, a special to type instrumented Ejector Release Unit (ERU) to define the ejection impulse and conventional cameras to identify post release trajectories.

This program is being conducted in collaboration with UK industry who will be carrying out the comparative studies under MoD contracts. Differing Computational Fluid Dynamics (CFD) techniques will be compared by British Aerospace sites (at Brough, Kingston and Warton), and at RAE and Hunting Engineering. Tunnel techniques will also be compared with specific interest in the Accelerated Model Rig (AMR) at BAe Brough and the Twin String Rig (TSR), Figure 5b, operated by the Aircraft Research Association (ARA), both of which represent state of the art techniques in store trajectory simulation. The majority of these approaches were discussed in UK contributions to Reference 4.

3.2.1 Instrumentation to Define Release Event

The practice of instrumenting release events is, of course, as old as flight testing itself; and in principle, any clearance trial such as those to gather aeroballistics data could be used for validation purposes. Unfortunately, it has been proven, time and again, that these routine clearance trial events are usually inconsistent or inadequately instrumented and documented and rarely of adequate quality, particularly when near field behaviour is being considered. The reasons for such variability are several, but the following are considered to be of most significance:

1) Imprecise or incomplete knowledge of the detailed conditions of flight before and at release event. This leads to two further issues: (a) An incomplete understanding of the initial conditions for dynamic modelling. (b) Inability to estimate store load at release.
2) Variability of crutching preloads and/or store compliance.
3) Lack of actual knowledge of the ERU performance and therefore the impulse applied to the store.

This programme is attempting to eliminate all these limitations by use of a comprehensive instrumentation fit and thorough ground test and preparation. The RAE Tornado is fitted with a MODAS (Modular Data Acquisition System), Reference 5, and a comprehensive sensor fit capable of meeting all our requirements for release work.

In respect of crutching and preload imprecision we are fortunate that the Tornado is equipped with the MACE system of Minimum Area Crutchless Ejectors. This system was conceived primarily to avoid crutch arms to reduce carriage drag; these arms being replaced by snubbing wedges which bear on the top surface of the special bomb suspension lugs.

To evaluate the store compliance on the pylon a special rig was constructed to enable a range of loads and torques to be applied to the mounted store and its compliance measured. The results permitted us to conclude that the store position would remain essentially constant in flight. We also consider that the store loading variations using the automatic MACE suspension system would be small and repeatable.

In order to define the release impulse applied to the store, a production Light Duty Ejector Release Unit(LDERU) has been modified so that the gas pressures within the ejector ram chamber and its position can be monitored during ejection of the store. The instrumentation is fitted to the forward and aft ejector rams. As the ram chamber walls are of insufficient thickness to allow a pressure transducer to be directly mounted, a suitable mounting had to be designed. This mounting permitted the installation of a piezo-electric pressure transducer, 1000 bar full scale, to monitor pressure at the ram crown. The ram position instrumentation, Figure 6, comprises a control rod which attaches to ram foot, by means of the existing gas exhaust mechanism, so that the control rod mirrors the ram extension. A thin multi-stranded steel wire is attached to the top end of the control rod, and runs over a pulley system and is attached to a rotary potentiometer, which is a spring biassed against the direction of rotation. The pulley attached to the potentiometer shaft is sized such that one rotation of the wiper shaft equates to full travel of the ram. The potentiometers are mounted externally on the pylon owing to the lack of internal space. The outputs from the ERU instrumentation are recorded on the aircraft MODAS system. We believe that this instrumentation of the release impulse is novel, Figure 7, and results compare extremely favourably with previous attempts to measure ejection force directly, which is invariably susceptable to noise and 'ringing'.

Two camera pods are carried one on each outboard wing pylon using a modified Carrier Bomb Light Series (CBLS) 200's. These carriers are cleared on virtually all stations of current UK aircraft and therefore offer a flexible package which can be used fleet wide. Each pod is equipped to carry three Photosonic 1P 16mm cine cameras which can be aligned in pitch and yaw to film the release over approximately the first 3 metres. An additional camera is mounted in the modified rear fairing of the Laser Range Finder Pod such that it is looking rearwards and across the fuselage at the release store to improve definition of the store yaw during release. The film records are analyzed at A&AEE Boscomb Down on a VISTA system, as developed by BAe Warton, Reference 6.

3.2.2 Flight Test Programme

For the purposes of these validation experiments the BL755 cluster weapon has been chosen as the release store. This is a relatively light store which could be easily disturbed particularly if released with the folding fin flip mechanism disabled making it more sensitive to the prevailing flow field conditions. The stores are to be released from the instrumented LDERU mounted in the port rear pylon position in various store configurations. The configurations for carriage and release, Figure 8, range from four stores to a single store to maximise the use of AO1 and AO2. All drops will be carried out at 5000ft over a range of airspeeds from 300kts to 550kts. Within operational limits the aircraft weight will be varied to alter the flight incidence during release, which together with the maximum achievable sideslip, is designed to provide the widest possible variation in release conditions.

Store carriage loads at all stations will be measured over a wide range of flight conditions, Figures 9a & 9b, for each store configurations, as an integral part of the program to provide a database for those interested in carriage aerodynamics. Aerodynamic induced loads have been recorded over a range of IAS, Mach Number, Altitude, Steady Heading Side Slips and wind-up turns up to limit incidence. The data will be acquired using both instrumented stores A01 and A02. Typical Aero Store results from a recent flight are shown in Figure 10.

To date (September 1988) all flights to record carriage loads have been completed for all store carriage configurations. Release flights at 300 kts, for maximum and minimum aircraft weights, have recently been completed for the four and single store configurations.

4 STORE FLIGHT CARRIAGE VIBRATION STUDIES

4.1 VIBRATION FLIGHT TEST PROGRAMME

During the early 1970's, it became apparent when compiling project design and test specifications, that existing UK MoD requirement documents were deficient for several store environmental conditions. One such deficient area was that associated with carriage flight vibration on high performance aircraft.

The RAE Vibration Flight Test Programme(FTP) was designed to gather data from a number of aircraft types, including a wide range of aircraft configurations and manoeuvres, using the vibration instrumented stores described above. The data gathered was to form the basis of a revision of MoD requirement documents.

This FTP continues to this day in its original role of gathering data from new aircraft and/or sortie profiles as they appear and, additionally, in the role of trouble-shooting.

This section describes the how data from the vibration measuring stores are processed and analyzed to help quantify the flight carriage vibration environment.

4.2 DATA PROCESSING

4.2.1 Data Bank

Over a period of eighteen years the UK MOD Data Analysis Facility at the Cranfield Institute of Technology (CIT), has been processing and archiving data obtained from a wide variety of trials. This has resulted in a data bank holding vibration and shock data arising from a comprehensive catalogue of store environments. The data bank relating to store flight carriage, including data from the RAE FTP, is believed to be the largest in Europe of flight vibration data relating to the external carriage of stores. The scope of the data bank at CIT is illustrated in Figure 11.

The data in the form of acceleration spectra, both average and peak hold has been archived onto the data bank. Analysis parameters, viz: acquisition rate, filter data, transform size, record duration, etc, together with additional information relating to flight conditions, aircraft configurations, etc, are also stored in the data bank.

The data bank at CIT has formed the basis of major parametric studies of store flight carriage vibration, Reference 7. These studies are currently being used to support the on-going revisions of DEF.STAN.0035 and DEF.STAN.00970.

4.2.2 Hardware

Data processing is carried out using the Facility's mini-computer based versatile signal analyzer. A diagram of the system is presented in Figure 12. Using this system, signals from replay tape decks are presented to the computer via 16 input channels of switched gain amplifiers, anti-aliasing filters and analogue to digital converters (ADC's), all operating under software control. The anti-aliasing filters may be either Butterworth or Bessel types (48 dB/octave) and were designed and built by CIT. Filters are matched to one another within $^{+}/_{-}$ 0.1% and $^{+}/_{-}$ 2 degrees in terms of gain and phase. The front end of the computer consists of an array processor capable of fast matrix arithmetic (17 million floating point operations per second). This capability, together with the full double-buffering techniques, permits real time analysis of 16 channels of data at rates up to 60000 samples per second, per channel, without any loss of data. All processed digital data are stored on disc, together with the analysis parameters, including calibration factors, acquisition rates and filter details.

4.2.3 Techniques

Data processing initially involves a data characterization phase to establish the most appropriate formats and processing parameters, eg: upper frequency limit, digital acquisition rate, resolution bandwidth, windowing, etc. This typically involves processing a limited quantity of data, probably that pertaining to the most severe condition identified from quick-look records.This phase is used to examine data in terms of its stationarity by processing data in g_{rms} versus time format. The second phase of data processing comprises the processing of the bulk of the gathered data using the formats and processing parameters established in phase one.

Data formats for store responses relating to carriage on high performance aircraft routinely include Acceleration Spectral Density; g_{rms} versus time, for checking the stationarity of data; amplitude probability density, for checking the normality of data.

To support subsequent assessments, it is usual to reduce the primary data into special formats to facilitate the recognition of data trends. Environment descriptions are derived from these formats.

4.3 ASSESSMENT

4.3.1 Environment Description

An environment description based on trial results represents in summary form the output of the data processing phase. A flow diagram illustrating the production process of an environment description from field data is presented in Figure 13. An environment description comprises two elements, ie:

a) Measured data defining the actual environment experienced during the trial.
b) An identification of parameters and trends governing that environment.

An examination of trends attempts to identify the major parameters effecting the severity of an environment. Such information can be used to extrapolate, either quantitatively or qualitatively, from the particular trials data to the more general situations of inservice use. Particular examples which have been studied using the instrumented stores are discussed below.

4.3.2 Vibration Response Versus Speed (G_{rms} v Dynamic Pressure)

Vibration as influenced by altitude and airspeed has been examined using a regression analysis of overall rms vibration on flight dynamic pressure. In this analysis, a relationship of the following form has proved to be appropriate:

$$g_{rms} = A^{*}(Q)^{n}$$

where A is a constant and Q is Dynamic Pressure

The data is then transformed into a linear equation of the form:

$$\mathrm{Log}(g_{rms}) = \mathrm{Log}\,A + n * \mathrm{Log}(Q)$$

By this means, the rate of increase of vibration, ie: the slope of the computed regression line, has been estimated.

This analysis has also been used to compute a vibration level for a reference dynamic pressure of 1000 psf. This parameter is useful when comparing the relative severity of aircraft configurations. An advantage of this means of comparing data is the improved statistical confidence in the reference value compared with that of a single estimate. A typical example of this analysis is shown in Figure 14.

The correlation coefficient is used to describe the quality of curve fit to the measured data. It has been found that there exists very high correlation between vibration and dynamic pressure, with coefficients of 0.99 being typical. Consequently, this type of analysis has been found to be a valuable aid in a general assessment of flight vibration records, because rogue data is easily recognized in this format.

Results of this regression analysis indicate that the slopes of the computed regression lines are seldom unity, but lie generally in the range 0.7 to 1.5, according to both aircraft and store types: specification documents, such as MIL-STD-810D, assume a slope of unity, which may be adequate for general use but may not be appropriate in particular cases.

4.3.3 Reference Spectra

The above process has been taken a stage further to the computation of 'Reference Spectra'. Such a spectrum represents the best estimate of vibration response at a reference flight dynamic pressure of 1000 psf. Reference Spectra are computed by carrying out a regression analysis for each spectral line, of which there may typically be 1024. Smoother estimates are usually obtained by a coarser bandwidth, for example, one-third octave. This type of analysis also provides evidence regarding the sources of excitation of stores during flight carriage. Typically, the correlation coefficient is good above around 200 Hz, indicating that aerodynamic excitation is dominant in this domain. It is believed that vibration mechanically transmitted from the aircraft to the store influences the low frequency store response, which corresponds to relatively poor correlation below 200 Hz. These trends may be seen in Figure 15.

4.3.4 Store Fins and Carriage Equipment

A special series of trials was instigated to examine any effects on store vibration due to the presence of a store's fins, and also to compare store vibration arising from MACE and Sway Brace carriage equipment.

It was seen from the trials that for tail vibration, an average reduction of around 50%, compared to the response of the finned store on Sway Brace, could be achieved by either removing the fins or by switching to MACE. No further decrease was seen by both removing fins and using MACE. These effects are apparent in Figure 16. It is believed that the more turbulent flow over the store when using Sway Brace equipment, together with a store whose ability to extract vibration energy from the airstream is enhanced by the presence of fins, leads to the increased vibration of the store.

4.3.5 Store Carriage Stations

Data presented in Figure 17 relates to four carriage stations on a Buccaneer aircraft, ie: inboard and outboard wing pylons and a bomb bay station with doors open and closed. It may be seen that the store response on either wing station is similar and somewhat more severe than the fuselage station. Also, as expected, carriage in an enclosed bomb bay is markedly less severe than for external carriage, in this instance by a factor of about 5 on overall g_{rms}.

The regression analysis shows the rate of increase of vibration with respect to flight dynamic pressure to be similar for all four carriage configurations.

4.3.6 Influence of Adjacent Stores

The influence of adjacent stores upon a store's vibration response was one of the first aspects to be addressed by the FTP and is extensively reported in Reference 8.

It is well known that the installed drag of multiple stores carried in close proximity to one another can exceed by a considerable margin the sum of drag components of each store carried in isolation. Furthermore, it is to be expected that the intensity of the turbulent field surrounding a store will be a function of the installed drag.

An analysis has been made comparing the effect of adjacent stores upon an instrumented round carried on a Phantom aircraft equipped with a triple-carrier. The analysis involved normalizing the g_{rms} values of each transducer channel obtained at 550 Kts. by those pertaining to the three store configuration, and averaging the results. These data are presented in Figure 18 from which it is inferred that:

a) Vibration severity is mainly a function of the local store configuration, rather than the position of the carrier on the aircraft.
b) Compared with single store carriage, a 30% increase in g_{rms} is shown when two stores are carried abreast in close proximity, and a further 10% increase when the store is surrounded by two stores in close proximity.

Similar effects to those seen on the Phantom are also evident in data relating to the Tornado equipped with a twin store carrier. Data comparing single and twin store carriage are presented in Figure 19 in the form of a regression analysis of vibration on flight dynamic pressure. This analysis indicates the twin store configuration to be generally some 50% more severe, in terms of overall g_{rms}, than single store carriage.

5 FUTURE DEVELOPMENTS - ULDAS

5.1 REQUIREMENT

Resulting from the operational experiences with A01 & A02 it was recognized that a new store was required to extend the research initiated by RAE and also to take advantage of the technological advances in instrumentation and electronic design. The new store is to be known as the Universal Loads Data Acquisition Store(ULDAS).

The areas addressed by the new store are:-

1) To provide an aerodynamic load balance where the roll component is measured directly rather than derived by computation.
2) The recording of linear and rotational accelerations for the separation of inertia and aerodynamic load components.
3) The measurement of the store surface pressure distribution.

5.2 MECHANICAL DESIGN

The mechanical construction of the ULDAS load balance is shown in Figure 20, the balance comprises two cylinders, the inner being attached to the aircraft suspension, and the outer forming the store skin. The two cylinders are connected, as with A01 or A02, by strain gauged links which measure relative displacements. Two links measure lateral loads, two for the vertical and one for axial loads. The sixth link is attached to a gimbal ring detects the rolling moment. Six accelerometers are mounted within the inner tube; three linear transducers sense accelerations along the three principle axes and three rotary transducers sensing rate of acceleration about the principle axes. Pressure tappings, a total of 208, are provided over the skin surface which are piped to miniature, electronically scanned pressure sensors, which are capable of accommodating 32 channels. Owing to cost and complexity considerations only 4 such devices, serving 128 channels, are provided at present. Therefore, it is recognized that several flights will be required to survey the entire store surface. Two reference pressures for the pressure sensors' self calibration, are also provided. All the transducers and associated instrumentation are mounted within the inner cylinder.

5.3 INSTRUMENTATION

The Universal Loads and Data Acquisition Store (ULDAS) contains a flight measurement and recording system designed and built, by CIT, to accept and condition analogue transducer signals, digitize and store the data in a solid state memory, Figure 21. Data channels to be measured are preprogrammed prior to flight by a portable microcomputer, which constitutes the Ground Station Equipment (GSE). ULDAS is controlled by a TMS32020 16-bit microprocessor. A signal conditioning unit (SCU) is provided to accept the load cell, accelerometer and reference pressure transducer signals and these signals are then routed to a 32-channel multiplexor. The output from this device is passed to an ADC which has a 12-bit resolution. Data is stored directly in the unit's main memory which comprises 256 K words of dynamic ram (DRAM) which is under the control of a second TMS32020 microprocessor. After a flight trial, the contents of the ULDAS main memory are downloaded to the GSE, either directly or via a Data Transfer Unit (DTU). The GSE stores data on floppy disc for subsequent off-line analysis.

In its present form, ULDAS will accept the following transducer inputs:

6	Load Sensors
6	Accelerometers
128	Pressure Sensors
2	Reference Pressures

The 128 pressure channels are presented to ULDAS as 4 sub-multiplexed data channels. Therefore, ULDAS is presented with a total of 18 analogue inputs. Using a sample rate of 20 sps per channel for all channels a recording duration of approximately 20 minutes can be achieved. Alternatively the system can be programmed, by the GSE, to provide various permutations of channel, sample rate and recording duration.

5.4 CALIBRATION

At present the store is mechanically complete and the instrumentation is in the process of commissioning. On completion of the instrumentation commissioning the store will be mounted in a wind tunnel balance calibrator. This will apply a series of loads and moments to the ULDAS balance so that a transformation matrix can be derived. This matrix will be used to resolve the individual strain gauge outputs into loads and moments at the store reference centre. The pressure measuring instrumentation will be assessed by mounting the store in the RAE 5M Wind Tunnel.

6 FUTURE FLIGHT TRIALS

Our current commitments are to the Tornado release disturbance data base,(Section 3.2). These trials are in hand at the present time. Commissioning of ULDAS is expected to be completed towards the end of that programme.

It is then proposed to use ULDAS to assess store loads and pressure distributions on both single and multiple carriage conditions in future support of that data base.

Further a series of non-circular shells are being manufactured as part of a programme to investigate the implications of variations from the circular cross-section, on carriage loads and vibration, to full square section in several stages, Figure 22 and Reference 9.

We are also currently considering the opportunities made available by cheap and/or ruggedised instrumentation to equip launchable stores for the analysis of the near field trajectory phase.

7 CONCLUSION

The case studies presented here have been chosen from a wide variety of trials carried out over a considerable period. They have demonstrated the utility of a maintained base of expertise for tackling problems of stores integration with flexible tools that are applicable on a fleet-wide basis. The data gathered has been to and found supportive of a wide variety of issues including

a) Validation of modern predictive techniques.
b) Inputs to revision of national (and international) standards.
c) Proof of concepts for new interface equipments.
d) Quick response to analysis of problems arising in service.

Our future plans are currently centred on the ULDAS store which will provide a significant increase in our data gathering capability and our flexibility of response.

REFERENCES

1.	F.J.Bigg N.Tait D.A.Williams	Weapon Testing Techniques AGARD CP No 223
2.	J.W.Peto P.G.Pugh	An Investigation by Mathematical Modelling Into a Problem of Store Release. Unpublished RAE Tech Memo Jan 1981
3.	P.J.Bateman	In Flight Loads On Stores On Twin Store Carrier On Harrier Unpublished RAE Tech Memo Jan 1981
4.	Various	Stores Airframe Aerodynamics AGARD CP 389 Athens Oct 1985 NATO RESTRICTED
5.	J.Woods	An Introduction To High Density MODAS RAE FS(B)3 Working Paper 1983
6.		VISTA Description
7.	D.Charles	Vibration Parametric Study of Air Armament Stores on Helicopters CIT, COA-DA-1373, 1988
8.	D.A.Williams	Assessment of Results Obtained From the Initial Vibration Store Flights CIT, COA-DA-133, 1977
9.	D.Daniel L.E. Lijewski G.Zollars	Experimental Aerodynamic Studies of Missiles With Square Cross Sections. AGARD CP 336 NATO RESTRICTED Sept 1982

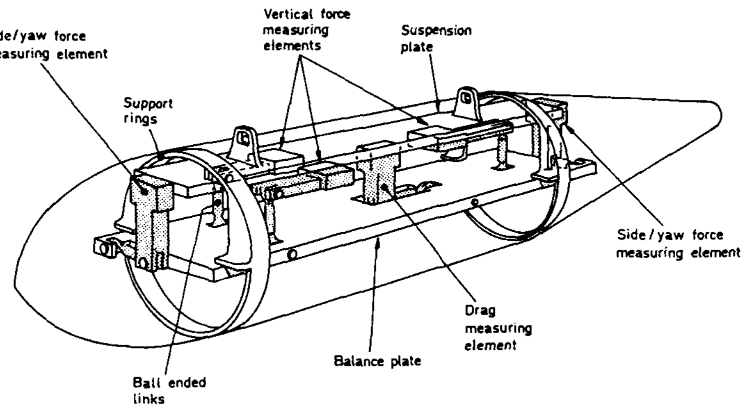


Fig 1 Aerodynamic store

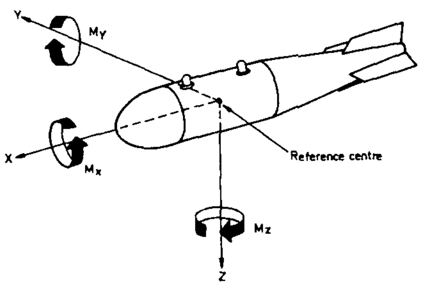


Fig 2 Positive forces & moments

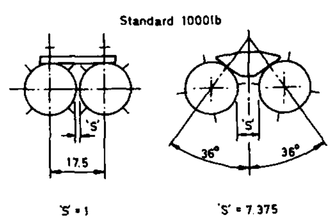


Fig 3 Types of twin store carrier used in Harrier trials

HARRIER TSC & YDC WITH 2 STORES

Fig 4a Normal Force v IAS

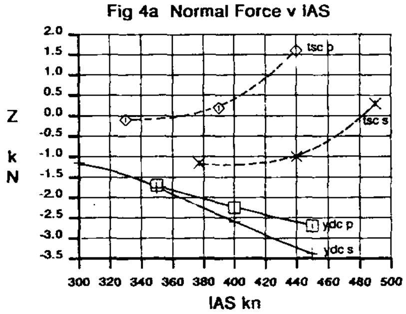


Fig 4b Pitching Moment v IAS

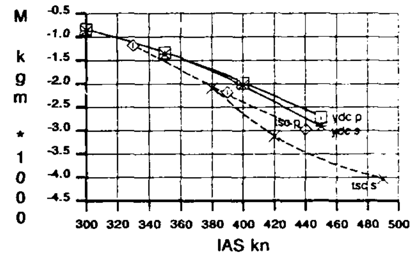


Fig 4c Side Force v IAS

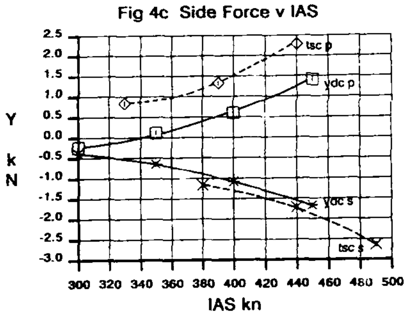


Fig 4d Yawing Moment v IAS

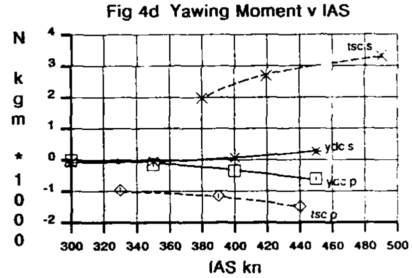
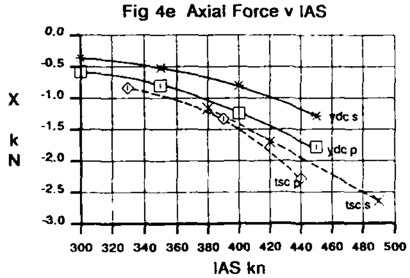


Fig 4e Axial Force v IAS



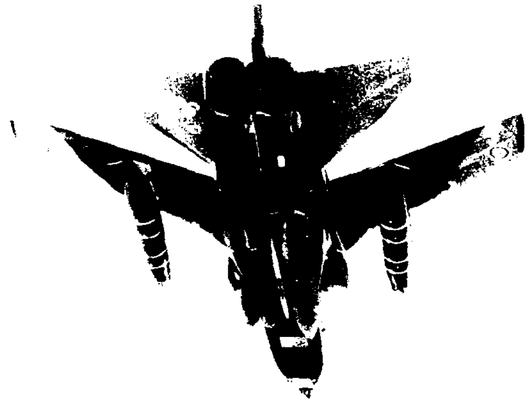


FIG 5a RAE TORNADO RESEARCH AIRCRAFT

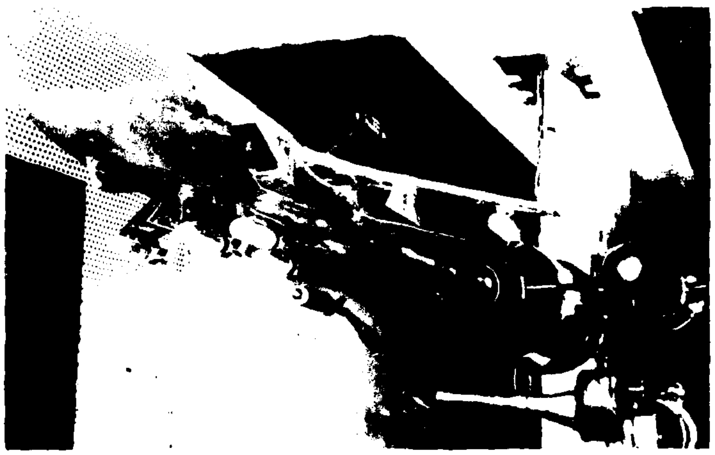


FIG 5b RELEASE DISTURBANCE TEST CONFIGURATION ON
ARA TWO STING RIG

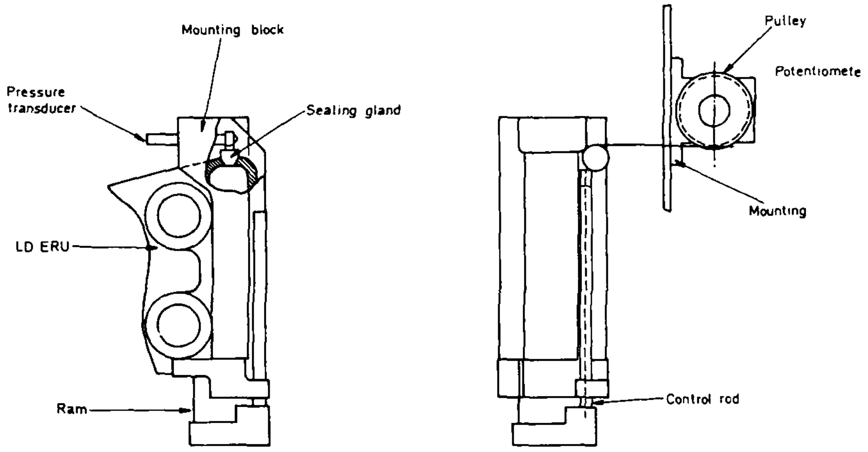


Fig 6 LD ERU instrumentation for Tornado

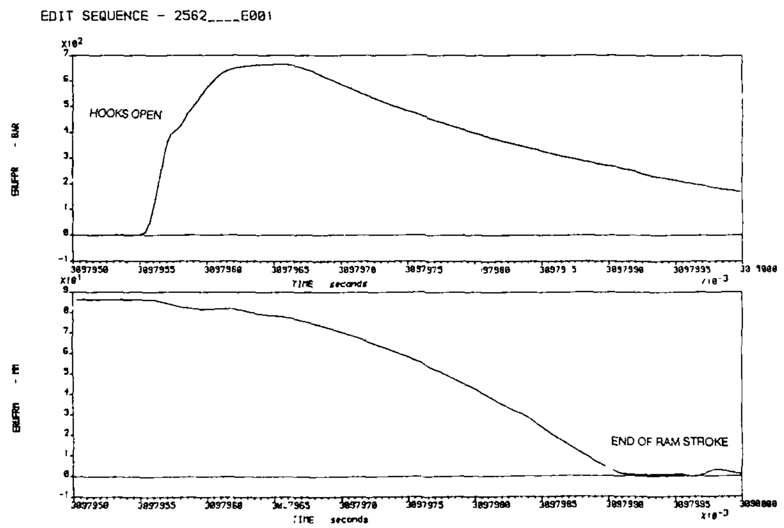


FIG 7 ERU INSTRUMENTATION OUTPUT FOR FORWARD RAM

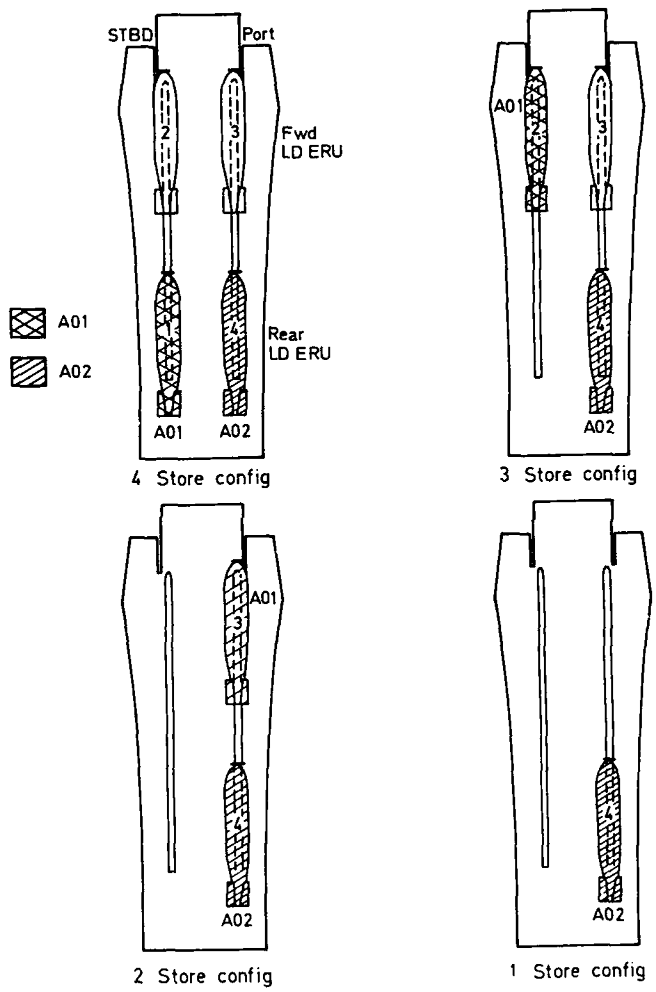


Fig 8 Tornado store configurations

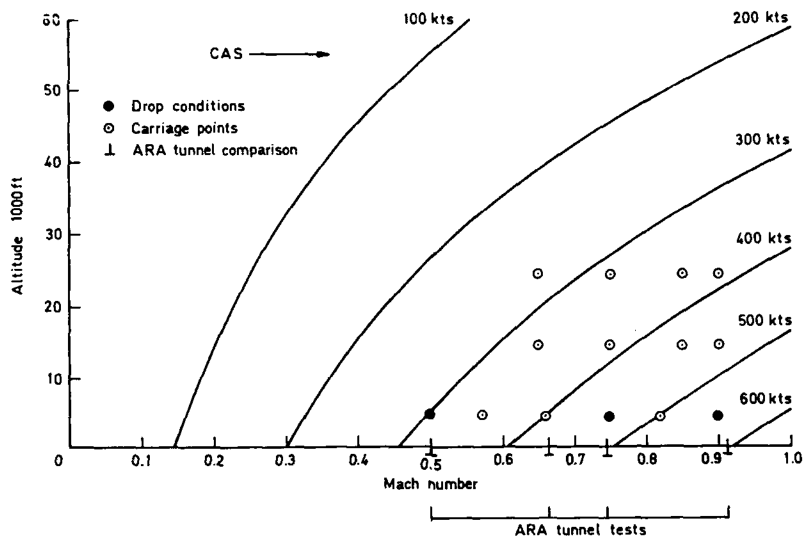


Fig 9a Revision of Tornado flight trials conditions

Height ft	IAS Kts	Mach No	Test Conditions
2000	300		Straight and level Steady heading side slip Wind-up turn to max g/alpha
	350		All test conditions repeated
	400		All test conditions repeated
	450		All test conditions repeated
	500		All test conditions repeated
	550		All test conditions repeated
5000	300		All test conditions repeated
	350		All test conditions repeated
	400		All test conditions repeated
	450		All test conditions repeated
	500		All test conditions repeated
	550		All test conditions repeated
15000		0.65	All test conditions repeated
		0.75	All test conditions repeated
		0.85	All test conditions repeated
		0.90	All test conditions repeated
25000		0.65	All test conditions repeated
		0.75	All test conditions repeated
		0.85	All test conditions repeated
		0.90	All test conditions repeated

Figure 9b Flight Test Conditions

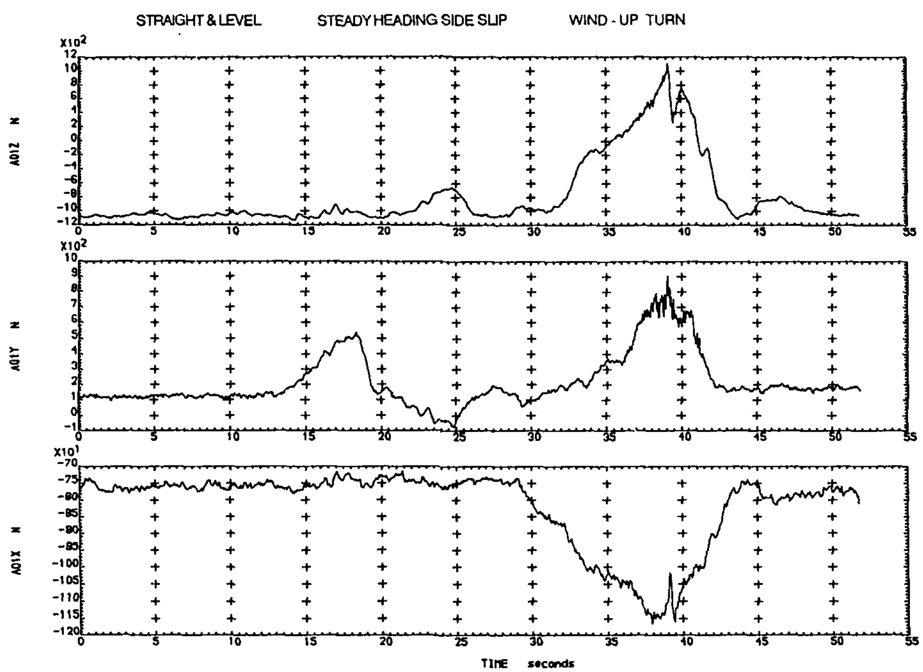


FIG 10 TYPICAL AERO STORE RESULTS

VEHICLES		ENVIRONMENT		
		Store Carriage	Transportation	Installed Equipment
Fixed Wing Aircraft	Jet	✓	✓	✓
	Propeller	✓	✓	✓
Helicopters		✓	✓	✓
Land Vehicles	Wheeled		✓	✓
	Tracked		✓	✓
	Rail		✓	
Ships			✓	

Figure 11: Range of the Data Bank

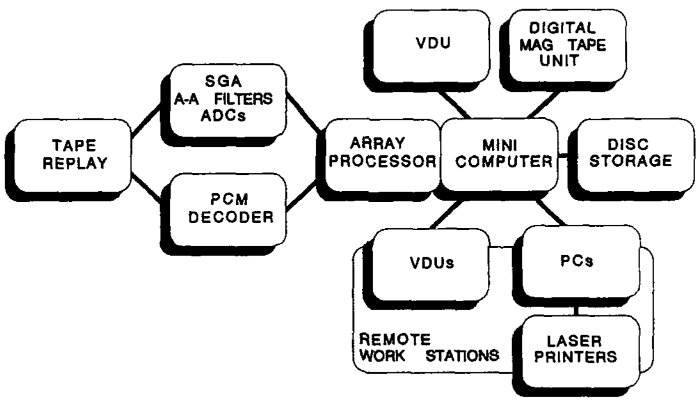


Figure 12: Data Processing Hardware

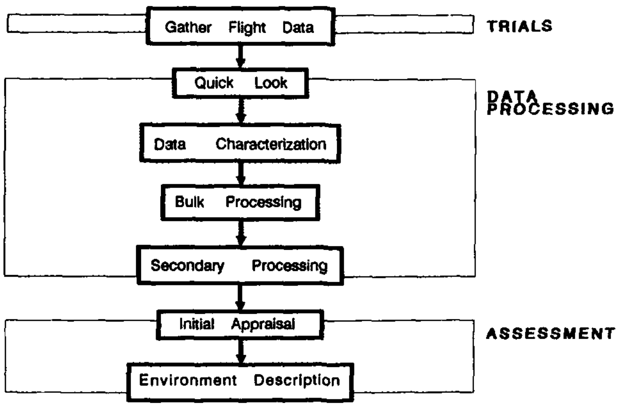


Figure 13: Derivation of an Environment Description

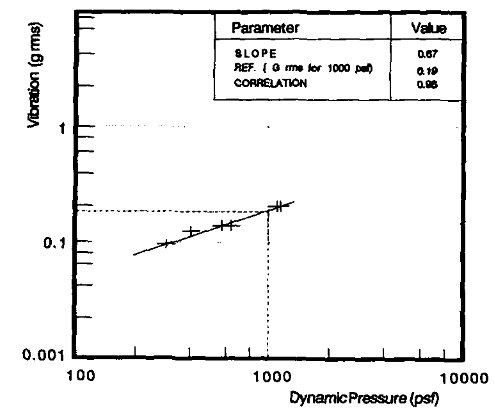


Figure 14: Regression Analysis of g rms on Flight Dynamic Pressure

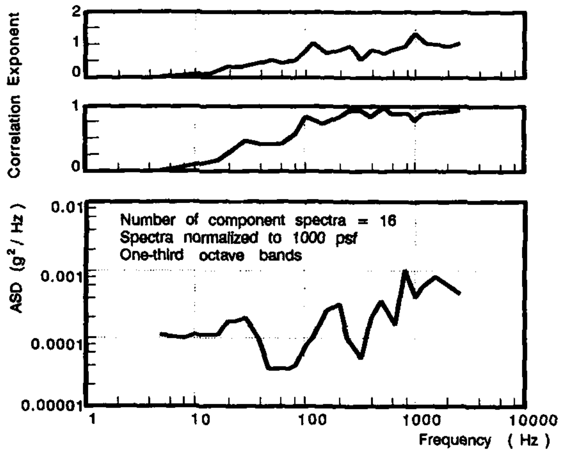


Figure 15: Typical Reference Spectrum

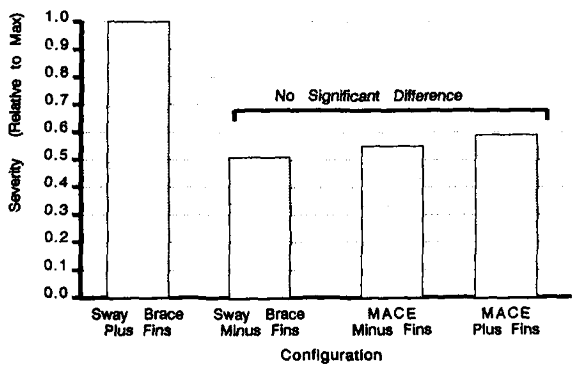


Figure 16: Effects of Store Fins and Carriage Equipment

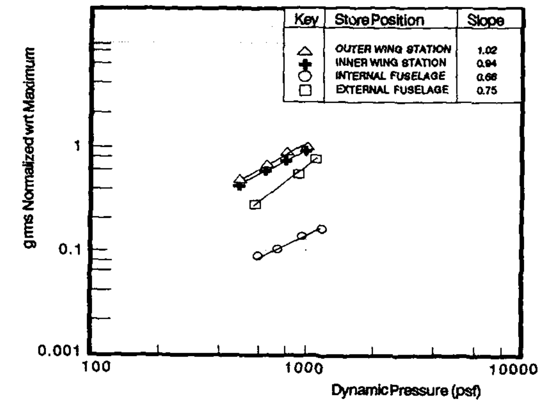


Figure 17: Relative Severity of Store Carriage Stations on Buccaneer

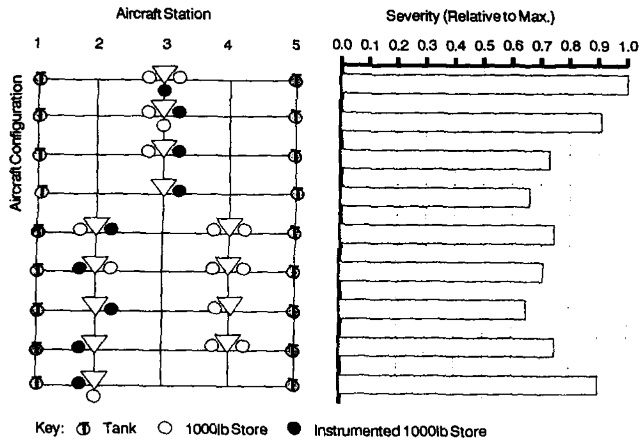


Figure 18: Relative Severity of Store Carriage Configurations on Phantom

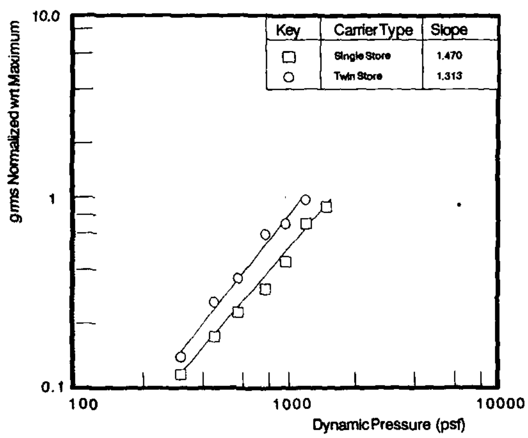


Figure 19: Effects of Store Carrier Types on Tornado

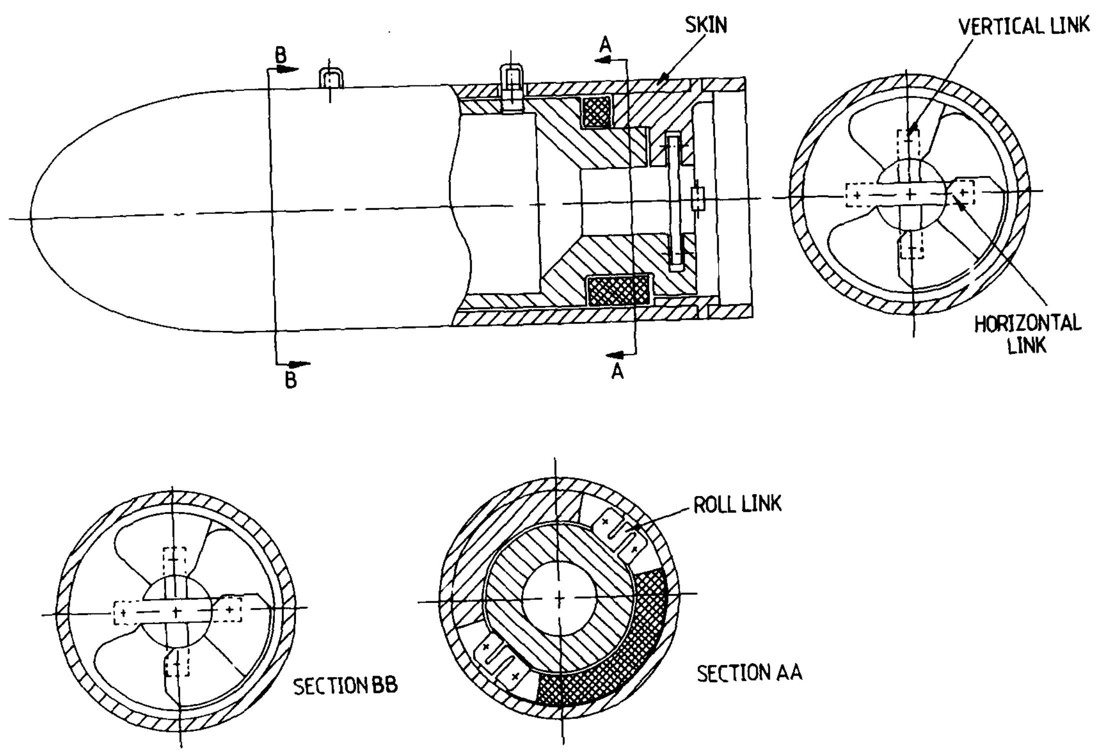


FIG20 SCHEMATIC DIAGRAM OF ULDAS

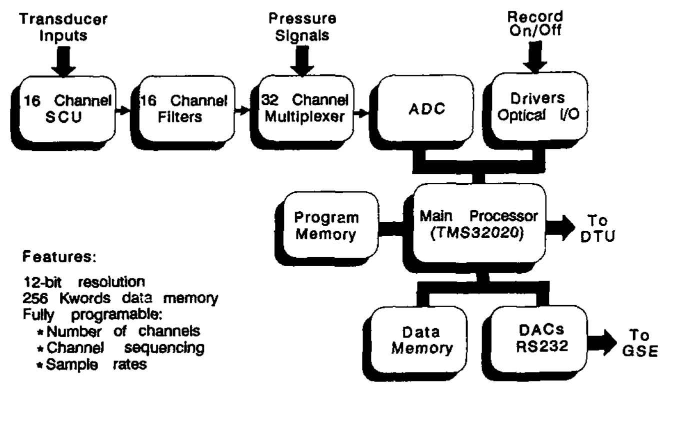


Figure 21: ULDAS Data Acquisition and Recording System

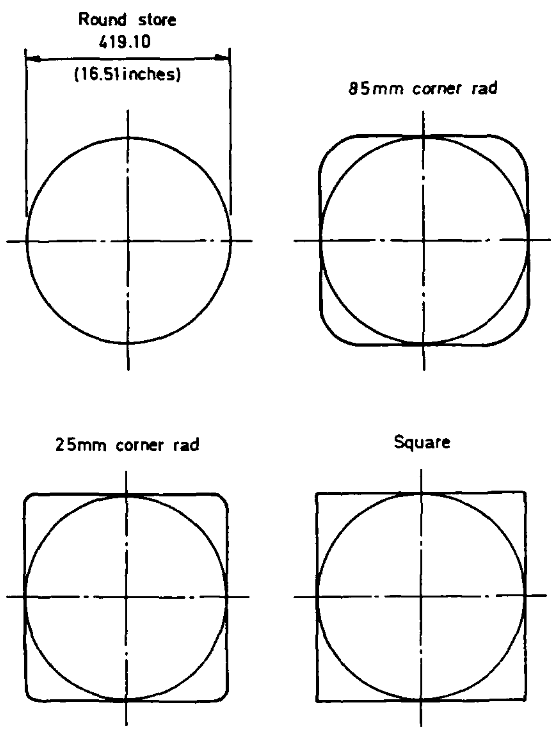


Fig 22 Cross—section of aerostore shapes (centre sections)

CF 18 480 GALLON EXTERNAL FUEL TANK STORES CLEARANCE PROGRAM

by

Major Douglas N Kobayashi, CD
Captain Mirko B.Zgela, CD
Aerospace Engineering Test Establishment
Canadian Forces Base Cold Lake
Medley, Alberta, T0A 2M0
Canada

ABSTRACT

The Canadian Government has embarked on a program with Canadian industry to manufacture a composite 480 gallon external fuel tank (EFT) designed for the CF-18 aircraft by McDonnell Aircraft Company (McAir). Prior to commencing the production phase of the program, the Aerospace Engineering Test Establishment (AETE) of the Canadian Forces (CF) was tasked to support McAir in the flight test certification of the 480 EFT on the inboard wing stations of the CF-18. The main objective of this program was to provide a proof of concept flight demonstration as well as establish an operational flight envelope for the carriage and jettison of the 480 EFT with and without adjacent stores. The certification process involved a progressive series of analyses, laboratory tests, wind tunnel tests, ground tests and flight tests. As the CF flight test authority, AETE was responsible for conducting all flight testing activities including flutter, active oscillation control (AOC), structural mode interactions (SMI), stability and control, structural carriage loads, separation/jettison, and dynamic response testing. This paper presents an overview of the joint CF-18 480 EFT stores clearance program and its main emphasis will be in describing the analyses and tests performed during the ground vibration testing, flutter, AOC, SMI and carriage loads phases. The aircraft instrumentation and the qualification/ground tests performed on the tanks prior to the beginning of flight test activities are briefly discussed. Test results and the technical problems encountered during the program are also presented.

1.0 INTRODUCTION

1.1 Background. In order to attain a sufficient war stock of external fuel tanks for the CF-18 aircraft, the Government of Canada established a follow-on fuel tank acquisition program. The options included buying more of the existing 330 United States Gallon External Fuel Tanks (EFT) or buying a new advanced composite material 480 EFT developed by McDonnell Aircraft Company (McAir) for the CF-18. The latter option was selected primarily because of the technological benefits which could be accrued from transferring filament wound composite technology to Canadian industry. As a secondary benefit, the 480 EFT had the potential of providing increased range performance and payload capacity. The 480 EFT certification program on the CF-18 was divided into two distinct phases. Phase I tasked the Aerospace Engineering Test Establishment (AETE) of the Canadian Forces (CF), with engineering support from McAir, to conduct a limited safe carriage and jettison flight test program (proof of concept demonstration program) so that a wartime clearance could be issued for the carriage of 480 EFTs on the CF-18 inboard wing pylon stations. In addition, a Royal Australian Air Force (RAAF) requirement to clear the 480 EFT on the centreline station up to current 330 EFT limits was also integrated into the program. McAir was assigned the overall responsibility of recommending certification of the 480 EFT on the CF-18. Upon successful completion of Phase I, the CF and Canadair Incorporate will conduct Phase II which will establish a full clearance envelope for the release/firing of stores and asymmetric carriage of stores in the presence of the 480 EFT.

1.2 Tank Description. The prototype 480 EFTs are presently manufactured by the Brunswick Corporation, under contract from the designer McAir. The tank is a light weight, survivable structure fabricated from two graphite filament wound shells with foam filled honeycomb core between them. Glass cloth laminate core inserts are used to provide frames for attaching graphite strongbacks, access doors and aircraft interface features. The tank does not contain baffles, and has been optimized for low manufacturing cost and ease of maintenance. The general layout of the 480 EFT is shown in Figure 1 while Figure 2 compares its basic geometry to the 330 EFT presently in service. The tank cannot be jettisoned with the aircraft trailing edge flaps fully deflected as the aft cone of the tank would impact the flaps. Carriage on the centreline pylon requires the use of a pylon interface adapter to allow sufficient clearance for safe operation of the landing gear. The benefits of the 480 over the 330 EFT include increased reliability, improved maintainability and valve performance, weight reduction, and lower unit and life cycle cost. To expedite the certification process, the prototype tanks were equipped with the same fuel system components as the 330 tanks, however, the production tanks will be fitted with a modular fuel system of improved performance which is presently under development.

2.0 480 EFT CERTIFICATION PROCESS

2.1 Methodology. The external stores clearance plan used for the 480 EFT program was based on a logical progression of laboratory tests, engineering analyses, ground tests, and flight tests. The methodology was developed and used successfully by McAir during the F-15 and F/A-18 Full-Scale Development (FSD) programs. The underlying objective was to clear the 480 EFT with the minimum number of test points. A block diagram of the basic clearance process is shown in Figure 3 and each of the major areas will be briefly discussed below.

2.2 Qualification Tests. To conform to the procurement specification, the 480 EFT was subjected to a qualification testing program performed by McAir and the supplier. The program consisted of a series of laboratory tests to ensure that the tank design could satisfactorily operate under all possible service conditions. The tank qualification program started in late December 1986 and included tests in the following areas: maintainability, lightning strike, catapult loads, slosh and vibration, ground ejection, burst pressure, flame engulfment, explosion containment, environmental testing, fragment impact, and fit and function checks.

2.3 Wind Tunnel Tests. All wind tunnel tests were funded and performed by McAir to gather the aerodynamic coefficients and derivatives required for the analyses. Transonic performance and stability and control data were obtained from a 6% scale model test conducted in the CALSPAN 8-foot wind tunnel. Low speed, power approach and power approach with half flap configuration stability and control characteristics were evaluated using the 12% model in the McAir Low Speed Wind Tunnel (LSWT). Flutter data to support the flutter analysis program was obtained using a 17.5% scale model in the McAir LSWT. Aerodynamic loads data were obtained from 6% scale model tests conducted in the McAir Polysonic Wind Tunnel in 1984. Separation and jettison trajectories were investigated in the Naval Ship Research and Development Center (NSRDC) wind tunnel. Testing included several combinations of Mach number and angle-of-attack

2.4 *Engineering Analyses* Several store configurations were analyzed however only the most critical configurations were selected for flight test evaluation. Analyses were performed based on computer derived studies and various test results, to determine the most critical configurations and to establish the initial flight test envelopes. The most significant analyses performed by McAir included: determination of the stability and control characteristics of the CF-18 aircraft with various store configurations; a six-degree-of-freedom separation analysis of the 480 EFT, fuselage mounted AIM-7, and specified stores; flutter analyses including level, climb, and dive tank attitudes for symmetric and antisymmetric fuselage boundary conditions; and carriage and dynamic load analyses for the pylon to aircraft interface loads and the wing dynamic response during weapons release.

2.5 Ground Tests. The ground tests performed prior to flight testing included ground vibration tests (GVTs), pylon load calibrations, centreline pylon adapter proof test, ground fit and function test, and electromagnetic compatibility safety of flight checks. The GVTs were performed to determine the aircraft dynamic characteristics for correlation with corresponding analytical predictions and validation of the dynamic math model prior to flutter flight testing. A summary of the critical GVT configurations is shown in Figure 4. The actual GVTs performed are described in Section 3.0. Two AETE CF-18 instrumented wing pylons were delivered to McAir where additional strain gauge bridges were installed. The pylons were installed in a loading fixture and a series of loads including load combinations were applied. The resulting gauge outputs were run through a matrix regression solution by McAir to generate the pylon to wing interface load equations. A proof load test of the centreline pylon adapter was performed by McAir to 115% of design limit load to gather reference strain levels at critical locations using two known flight loading conditions. As the centreline pylon adapter was not instrumented with load bridges, the strain levels resulting from the applied loads were not to be exceeded during flight testing. The ground fit and function test verified 480 EFT loading procedures, clearances to adjacent structure, and fuel transfer capability.

2.6 Flight Tests. The flight test program included testing in flutter, active oscillation control (AOC), structural mode interaction (SMI), carriage loads, dynamic response, stability and control, separation/jettison, and limited performance testing for drag index measurements. The first four activities are described in Section 4.0.

3.0 GROUND VIBRATION TESTING

3.1 Cantilevered Pylon GVT. This GVT was performed by McAir to determine the liquid fuel correction factors as a function of tank fuel level. The correction factors were used to develop a dynamic model of the 480 EFT and wing pylon that accurately defines its modal characteristics. This model was used to identify the configurations to be tested in the full GVT on the production aircraft. Based on the results of the full aircraft GVT, the dynamic model was then modified slightly to improve its correlation with the measured data from both the cantilevered pylon GVT and aircraft GVT results. The final correlated analytical model was then used for all subsequent flutter studies. The test setup consisted of a 480 EFT mounted to a standard F/A-18 wing pylon which was attached to a rigid test fixture in the same manner as normally installed on the aircraft wing. Five different tank fuel levels (empty to full) were tested. Prior to tank installation, mass properties of the empty tank were measured. The weight of the tank was measured by suspending the tank from two load cells, and the center of gravity location was determined from the measured reactions at the load cells. The mass moments of inertia were obtained by the torsional pendulum method.

3.2 Production Aircraft GVT. The full aircraft GVT was conducted by McAir on a single seat production aircraft in St. Louis, Missouri. This GVT was conducted to provide vibration data to validate the aircraft and 480 EFT analytical dynamic model used to perform flutter analyses. It also provided baseline data for comparison with vibration mode frequencies and damping coefficients that were obtained during flutter flight testing. The three GVT configurations tested (see Figure 4) were selected based on preliminary flutter analyses conducted with liquid correction factors correlated to the cantilevered pylon GVT results. The aircraft was supported on a soft jack system designed to dynamically uncouple the aircraft from the ground (providing aircraft rigid body modes at frequencies less than 2 Hz). The aircraft was supplied with hydraulic and electrical power and the canopy and all access doors were closed. To ensure all control surfaces were in a neutral position the Control Augmentation System (CAS) was deactivated by selecting the flight control system in the "RIG" mode. To ensure wing symmetry for flutter testing, the pylons stores were rigged to minimize freeplay. This consisted of tightening the pylon sway brace pads, pre-loading the tank aft tie, torquing the pre-load post and shimming the pylon aft attachment (see Figure 5) to ensure that maximum mechanical energy was transmitted through the interfaces. The symmetry of the store rigging was verified by comparing the store pitch, yaw and roll mode resonant frequencies on both sides of the aircraft using dwell excitation. The values obtained were different and adjustments were made to obtain acceptable limits of dynamic symmetry.

Two electrodynamic exciters were used throughout the testing to excite all wing, fuselage and store modes of interest. Symmetric and antisymmetric frequency response surveys were conducted to obtain transfer function plots (see Figure 6) for various locations. A sine sweep excitation ranging logarithmically from 1 to 25 Hz at constant excitation force was used to obtain the plots. Modal frequencies were identified from these plots and then each mode was manually tuned and force input linearities were measured for all the modes. Modal damping values were obtained using the log-decrement method on single mode decay time histories captured on the HP 5451C Analyzer. The mode shapes were then mapped using the multi-mode sinusoidal excitation technique. Different frequency oscillators were set to the modal frequency of interest and the exciting signals summed. This combined signal was amplified to drive the shakers. Using this technique up to three modes were simultaneously mapped, and the structure responded in the discrete modes corresponding to the input modal frequency. Approximately 150 mapping stations were used during mode mapping.

3.3 SMI GVT. The SMI GVT check was conducted in conjunction with the production aircraft GVT to verify that low frequency tank modes do not couple with the flight control system to produce an unacceptable dynamic response in the aircraft. The SMI setup consisted of exciters attached laterally on the tanks with accelerometers attached to the ailerons, wings, stores, and fuselage at the stick position and the flight control feedback accelerometer package. The centreline carriage configuration SMI was performed with the aircraft landing gear extended on soft tires (tire pressure reduced to 50% of nominal). The test procedure consisted of sinusoidal sweeps through the tank mode frequency ranges (2.5 to 4.0 Hz for wing carriage configuration and 1.0 to 6.0 Hz for centreline carriage) using maximum force lateral excitation on the tanks, followed by a dwell at the antisymmetric roll frequency. During the dwell an operator in the cockpit positioned and held the control stick in each of the four stick quadrants while data were recorded. For some of these conditions the input excitation force was suddenly removed and the decay time trace was recorded to investigate any sustained oscillation of the control surfaces. SMI was investigated in all possible flap positions.

3.4 **Rigging Check GVT** Rigging check GVTs were performed at AETE to ensure proper installation and rigging of aircraft stores prior to flutter testing. The checks were limited to the two most critical flutter configurations and only to those modes having a significant contribution to the flutter mechanism, as predicted by analysis. The test setup and procedure was similar to that used in the production aircraft GVT. Since AETE did not have a soft jack suspension system, the GVT was carried out with landing gears extended on soft tires. Although the suspension systems for tests were different, previous testing had shown that both systems yield virtually identical results. As with previous GVTs, freeplay was minimized to achieve dynamic similarity on both sides of the aircraft. Linearity checks were also performed for the most significant modes. Transfer function plots were gathered at selected locations on the aircraft and finally the modes of interest were partially mapped by manually recording response amplitude and phase relative to a reference location on the structure.

4.0 FLIGHT TESTING

4.1 **Aircraft Instrumentation**. AETE's two instrumented CF-188 aircraft, CF-188701 (single-seat) and CF-188907 (two-seat) were used during the flight test program. These aircraft have identical data acquisition systems, capable of acquiring data from the avionics multiplex (mux) buses and from other sources (see Figure 7). The current system provides a 64-channel analogue data acquisition capability. Data from the analogue signal conditioners along with selected data from direct analog and digital inputs, mux buses via the Data Bus Interface Unit (DBIU), time code generator and the Flutter Exciter Control Unit (FECU) are encoded into a pulse code modulation (PCM) format by a programmmable Data Acquisition System and stored on the onboard MARS 2000 tape recorder. IRIG-B format time code, pilot voice and direct analogue data are also recorded. An L-band, frequency modulated transmitter radiating four watts total power from a pair of antennae is used to telemeter PCM data to the Flight Test Control Room (FTCR) for recording and real-time monitoring (see Figure 8). To enhance flutter testing part of the standard instrumentation consists of wing stain gauges which were installed by McAir during production assembly. These gauges are sensitive to either wing bending or wing torsion modes. The gauges are installed at three different spanwise locations to allow identification of the overall wing motion by comparing gauge output magnitudes. Pylon/store motion is determined by using lateral and vertical accelerometer signals.

Additional measurands required for specific flight test activities on CF-188907 included the following instrumentation: forward and aft wing tip and wing fold accelerometers; center of gravity vertical accelerometer; pilot seat lateral accelerometer; radar bulkhead vertical and lateral accelerometers; nine strain gauge channels for the two loads calibrated pylons; the flutter exciter control unit (FECU), and the vertical and lateral accelerometers mounted on inboard and outboard stores on both sides of the aircraft. Additional instrumentation required for flutter testing consisted of the added accelerometers and a series of strain gauges bonded internally to the wing structure and pylon. Aileron motion sensors were also used to provide the wing excitation input signal. The calibrated pylon strain gauge installation enabled real-time measurements of pylon aft attach vertical and side loads, pylon post preload, pylon hook vertical load, pylon post roll moment, and aft tie fuse load.

The instrumentation on CF-188701 was less extensive than on CF-188907. The aircraft was configured with an aircraft centre of gravity accelerometer and approximately 30 strain gauges for in-flight strain monitoring at designated critical locations in the centreline pylon, centreline pylon adapter, the left-hand outboard upper longeron, and the centreline pylon aft attachment fuse. Figure 9 shows a typical strain gauge installation on the centreline pylon critical area.

The FECU can be installed in either aircraft in place of the left hand digital display indicator (see Figure 10). It sends signals to the flight control computers which then send corresponding commands to the ailerons. Various modes of aileron excitation are available including sinusoidal sweep (from one frequency to another), dwell (at one frequency for a given time) and random (pseudo random noise within a selected frequency band). The FECU also has built-in safety features which allow shut down of the control surface excitation from the panel "run about" switch, control column paddle switch or anytime the aircraft exceeds a roll rate greater than 20 deg sec, or a normal acceleration greater than 2.0g or smaller than 0g. The FECU can be pre-programmed using up to 15 different set-ups. The FECU control display can be reproduced on a monitor in the ground station allowing verification of program entry during testing.

4.2 **Flight Test Control Room (FTCR)** For flight safety most of the flight test missions were monitored in real-time in the FTCR at AETE. The FTCR processed raw telemetered PCM data from the aircraft to produce real-time data displays, including two large television screens for parameter display (using up to five different page formats), five six-channel strip chart recorders, a ten module engineering display of critical flight parameters, a six parameter status alarm display, a binary light display, IRIG-B time display, and a flutter analysis workstation comprising a fast fourier analyser for near real-time spectral analysis, four lissajous scopes, and a display for monitoring the aircraft FECU parameters. The FTCR also has the capability to perform post-flight data tape editing and reformatting of aircraft data tapes to compatible computer tapes. The FTCR has a 32-channel digital communication system which provides each operator with capability to transmit/receive on any air-ground-air communication with the test vehicle via one of two UHF transceivers. All operator positions have a selectable "Hot Mike" capability, as well is a momentary "Master Override" which disables all communications on every intercom except the individual who has selected the override.

4.3 **Risk Assessment** AETE's flight testing policy is to ensure safety of flight during all phases of the flight test program while conducting flight test activities in the most efficient manner. To ensure adherence to this policy AETE has an internal mechanism by which flight testing is carefully reviewed and risk factors assigned accordingly. Risk assessment factors consist of both the probability of an occurrence and the consequent damage. The probability of flight safety occurrences is expressed in terms of no, low, medium or high risk while the consequent damage is classified in categories from "A" to "D". The first category being the loss of an aircraft and the latter being damage that can be repaired within AETE resources. Figure 11 shows the risk assessment factors as assigned by AETE for the different 480 EFT flight test program activities.

4.4 **Critical Test Configurations** Since it would be impractical to flight test all store configurations, efficient flight testing dictates that a judicious selection has to be made. Based on the analyses and different laboratory test results a selection can be made and generally flight testing is limited to one or two critical configurations for each of the flight test activities. Figure 12 shows the configurations flight tested during this program. These configurations were determined to be the most critical during the GVT and analysis phases of the store clearance program.

4.5 Flutter Flight Testing. The aim of this testing was to verify through extrapolation of flight test data and correlation with analyses that the allowable carriage envelope of the CF-18 for the critical configurations is flutter free up to 1.15 times limit speed, and as long as there is unequivocal indication of positive damping. Flutter flight testing basically consisted of monitoring modal damping trends and frequency coalescence of the different modes involved in the flutter mechanism, with increasing dynamic pressure. Classical flutter will occur when two mode frequencies coalesce as dynamic pressure is increased and when at least one of these two modes becomes unstable (damping decreases to zero) as dynamic pressure is increased. Flutter onset will occur at the speed where these two conditions are satisfied. Because of the destructive nature of flutter, it is necessary that flight test data be acquired at subcritical speeds.

Flutter flight testing was conducted over a range of altitudes and airspeeds using the FECU system to provide inflight aileron excitation. Stick raps were employed when the FECU could not be used because of normal acceleration limits. Sinusoidal sweeps and single frequency dwells were used at each test condition. In general, the FECU was initially set in the sweep mode to determine resonant frequencies. Using the results from the sine sweep spectral analysis, sine dwells at and near these frequencies were then performed to obtain damping values. Figure 13 shows both types of excitation and corresponding wing responses. The frequency and damping values were subsequently used to predict flutter margins. After establishing the damping data and flutter margin, flight tests were performed using a build-up technique (increasing dynamic pressures) to verify the predicted flutter free flight. The test points were divided into distinct dynamic pressure groups as shown in Figure 14 which is a typical test matrix flown during this phase. Each successive group represents a higher dynamic pressure zone; therefore, to ensure flight safety, data was analyzed between each group to verify that flutter would not be encountered in the next group.

Depending upon the configuration and the flutter mechanism involved, symmetric or antisymmetric excitations were used at different tank fuel states and aircraft attitudes to excite the modes of interest. In addition, different wing loading conditions were conducted to determine the sensitivity of the flutter mechanism to wing loading. The last test points consisted of a series of dives performed at maximum velocity from 30K to 5K mean sea level (MSL) with one second dwell excitations at selected altitudes to demonstrate flutter free operations. Each flutter flight followed the same basic procedures. Prior to take-off the test pilot would verify onboard data system operations and FECU programming. Shortly after take-off, emergency FECU shut down procedures were verified at safe flight conditions. Following clearance from the flight test controller the test pilot proceeded to the required test points.

Real-time monitoring of key parameters was performed on strip charts and lissajous displays. Review of near real-time transmissibility plots (T-plots) was performed as sine sweeps were completed and review of the decay trace was performed during dwell excitations. Post-flight activities included review of strip charts, generation of T-plots, power spectral density plots and flight condition data, and tabulation of modal frequency and damping flutter parameters.

4.6 Active Oscillation Control (AOC) Testing. The F/A-18 aircraft encounters a limit cycle 5.6 Hz oscillation during high speed, low altitude flights when configured with heavy outboard stores with high pitch inertias and wing tip missiles on. Unlike flutter, the limit cycle oscillation (LCO) does not go unstable but results in severe lateral acceleration levels in the cockpit that hamper pilot effectiveness during weapons delivery. This phenomenon is because of a complex interaction that excites the antisymmetric outboard store pitch mode. It is characterized by wing bending and torsional motion which couples with the fuselage to produce lateral fuselage bending, and is aggravated slightly by the carriage of heavy inboard stores, with the current worst being the carriage of 330 EFTs. The LCO is presently suppressed to an acceptable level by the AOC system that is implemented in the flight control system. The AOC system is automatically activated when the aircraft is flying above 0.82 Mach or below 9K feet MSL when carrying heavy outboard stores and AIM-9 wing tip missiles. As this oscillation is not predicted analytically, flight testing was conducted with 480 EFTs to verify the AOC system adequately controls the oscillation. AOC testing was conducted for the configuration shown in Figure 12 with full, half, and empty tank fuel states. Flight testing was also performed with the AOC system deactivated under similar flight conditions so that a direct system effectiveness assessment could be made. A minor modification to the flight control computer wiring was required to disable the AOC system inflight.

The AOC test matrix is presented in Figure 15. The test technique basically consisted of flying symmetric manoeuvres under increasing normal acceleration and Mach number while simultaneously exciting the structure with lateral stick raps. The lateral acceleration levels at the pilot seat were monitored and a value of 0.15 g was established as a soft limit above which testing would be stopped. On completion of the test matrix, simulated weapon delivery manoeuvres using 20 to 35 degree dive angles at maximum velocity were performed to demonstrate the AOC system effectiveness. The AOC flight testing procedures followed the same format as for flutter except that the FECU was not used and real-time data processing was limited to strip chart monitoring.

4.7 Structural Mode Interaction (SMI) Testing. Although previous flight test experiences with similar store configurations have shown that no SMI coupling was likely to occur during flight, demonstration of an SMI free aircraft was still required. SMI tests consisted of both taxi and flight tests, with the high speed taxi runs commencing first. Taxi runs were conducted at speeds approaching the take-off speed. Flight testing was carried out during take-off and climb-out at a number of low speed and altitude conditions. Other flight test points were integrated within the flutter test matrix. Aircraft flutter configuration 1 was flight tested as it was the most critical for SMI.

Taxi runs were performed with full 480 EFTs and half flaps selected. The first taxi run was performed to a speed of 125 knots on a smooth runway while the second was performed to speeds up to 100 knots on a rough runway in attempt to induce structural mode coupling with the flight control system. During these runs the control stick was rigidly held in the aft right and forward left quadrants respectively for 5 to 10 seconds to see if an oscillation would build-up. Flight tests consisted of exciting the aircraft structure with lateral and longitudinal stick raps while monitoring the aircraft and flight control surface response. The stick raps were induced at regular intervals during take-off and climb-out to 20K feet in military power and during steady-state wing level conditions at 10K feet MSL, Mach 0.60 and 20K feet MSL, Mach 0.80.

4.8 Loads Flight Testing. The overall objective of this testing was to acquire flight test data to identify the safe carriage envelope of selected CF-18 480 EFT configurations. To expedite the 480 EFT stores clearance program, flight testing was conducted using both CF-188701 and CF-188907. The aircraft configurations selected for testing is shown in Figure 12. In general, the test flights consisted of performing standard manoeuvres which were known from previous flight testing and analyses to induce critical loading of the pylon to aircraft attachments. These manoeuvres, as defined in Reference 1, included steady state pull-ups (SSPUs), wind-up turns (WUTs), steady state push downs (SSPDs), 1 g 360 degree rolls, -1 g 180 degree rolls, rudder kicks (RKs), and rolling pull-outs (RPOs).

For flight safety considerations, loads and stability and control (S&C) test points on CF-188701 were integrated into one test matrix because some S&C test points were considered load critical and similarly some loads test points, for example the RPOs needed S&C clearance prior to execution. All testing on CF-188701 was conducted with empty wing fuel tanks as this was the only configuration cleared at the time of testing. Two methods, trajectory analysis and calibrated strain gauges, were used to determine the centreline loads on CF-188701. Trajectory analysis involved using measured aircraft flight path parameters and previously derived wind tunnel data in a computer model which calculated inertial and aerodynamic forces, and then solved for the pylon attachment loads. The prediction process is shown schematically in Figure 16. This technique required a considerable amount of post-flight data processing due to the large number of time slices within a manoeuvre. After reviewing the data and predicting the loads after each mission, clearance was given to proceed to the next test point. The second method involved using centreline adapter strain gauges which were installed and calibrated in a test rig by McAir using the maximum loads that were anticipated with a 330 EFT during worst case manoeuvres. After calibrating the gauges, plots of strain versus load were provided. Confidence using this method was relatively low because of the limited instrumentation used in the correlation.

The loads test matrix for CF-188701, illustrated in Figure 17, used a build-up technique based on both progression in dynamic pressure and criticality of the manoeuvre. Testing usually started with SSPU manoeuvres leading to high normal accelerations (Nz) with no sideslip and finished with RPO manoeuvres generating high Nz with large amounts of sideslip. Testing included a number of different tank fuel states from empty to full, however, some manoeuvres were restricted in terms of tank fuel quantity and pilot control input, because the design load envelope, according to initial analysis, would have been exceeded. Strain gauges at designated instrumented locations on the centreline pylon were also measured and monitored in real-time to ensure design limits as predicted by McAir were not exceeded.

With the special instrumentation described in Section 4.1, CF-188907 was used to measure real-time pylon to wing interface loads with its fully instrumented, calibrated wing pylon. Symmetric and asymmetric manoeuvres that were considered non-departure critical were performed first. Asymmetric manoeuvres in critical departure regions of the flight envelope were conducted after S&C flight test clearance. The test matrix for CF-188907 testing was similar to that of CF-188701, however, testing was not limited to the initial flight test envelope as determined by pre-flight analyses. Since real-time monitoring of pylon loads was available a more practical approach was used which allowed envelope expansion during testing.

For example, when the aircraft normal acceleration (g) was limited to 4.0g for a given manoeuvre, flight condition, and fuel tank state, the first test point would start at 4.0g followed by envelope expansion test points performed above 4.0g by increments of 0.5g until one of the pylon interface design loads was achieved or would have been exceeded at the next g increment. The pylon aft attach loads were normally the limiting factor because of the design limit load being a function of a combination of aft side and vertical loads. A similar build-up technique was used for asymmetric manoeuvres, however, not only was there a build-up in g but also in the amount of control input. For example, at a given test condition with an empty centreline EFT, a 4.0g manoeuvre would be performed and followed by a 6.0g, both with half control input. Subsequently, the test points were performed at the exact same test condition with the same g build-up except using a full control input. The progression of test points was contingent on the magnitude of the measured load.

5.0 TEST RESULTS AND DISCUSSION

5.1 Qualification Tests. The qualification test program was successfully completed in September 1987 with the prototype tank meeting or exceeding all the design requirements. From a strength point of view, the tank structure is slightly over-designed, however, the harsh survivability requirements were found to be the overriding design factors.

5.2 Ground Tests. All ground tests were performed successfully although some problems were identified with regard to electromagnetic interference (EMI). Nevertheless, corrective measures were identified and will be implemented in the final design. The fit and function test demonstrated that the 480 EFT functioned as expected on all carriage stations. The minimum clearance requirements as specified in Reference 2 were achieved with the 480 EFTs on the wing stations, however, when installed on the centreline station the clearances required between the tank and both the nose landing gear and ground were not quite met. A waiver was granted as these violations were deemed as not being major safety or capability concerns. The fit and function also proved that loading the 480 EFT on the centreline position was considerably more difficult than with the 330 EFT, therefore, a new loading procedure was developed.

5.3 Ground Vibration Tests. The cantilevered pylon GVT provided vibration data required to develop a good analytical dynamic model of the pylon and tank installation. The natural frequency and mode shape plots of the primary tank vibration modes obtained from the resulting NASTRAN vibration model correlated very well with the measured cantilevered pylon GVT. This provided a good analytical tool for performing the flutter analyses; however, some anomalies with partial tank fuel states were encountered during actual flutter testing (see Section 5.4).

Figure 18 compares the resonant frequencies for the modes found from analysis, the production aircraft GVT and rigging check GVT with the aircraft configured with wing tip missiles off, 2 MK-82 bombs outboard and full 480 EFTs inboard. The frequency and mode shape results show relatively good correlation and are consistent with the level of agreement achieved in other test programs. The test results for the other two aircraft GVT configurations showed even better correlation with analytical predictions. The differences in results were attributed to freeplay in the physical system and coupling between some closely spaced modes. Of significant interest was the identification of an antisymmetric mode at 7.46 Hz which was identified during the rigging check GVT. The mode was not found by analysis nor during the production aircraft GVT. This mode resembled wing first antisymmetric bending mode, however, with reverse relative phasing between tank pitch and fuselage lateral motion. This phenomenon is still under investigation by McAir. A possible cause could be the result of modal interferences of the structure elastic modes with the soft tire suspension system. In summary the production aircraft GVT results, as verified by the rigging check GVT, indicated that the analytical vibration model of the aircraft and 480 EFT was suitable for use in flutter analyses.

No instabilities, sustained oscillations, or unacceptable dynamic response of the flight control system were encountered during any of the SMI GVT testing.

5.4 Flutter/AOC/SMI Flight Tests. Based on preliminary flutter analyses, the critical external tank fuel state for flutter testing for both aircraft configurations was predicted to be with full fuel. Flutter flight testing of configuration 1 validated this prediction and the test results correlated well with the analytical results. The stability of this configuration was further verified by testing out to maximum speed at 4000 feet and demonstration dives out to the allowable flight limits, with adequate damping values being exhibited in all cases. However, during configuration 2 flutter testing with 50% full external fuel tanks the damping ratio was found to be lower than previously measured with full tanks. A near flutter onset condition was observed at 4000 feet pressure altitude at 0.92 Mach during an 8.5 Hz dwell excitation. Real-time monitoring of wing gauge outputs indicated a significant reduction in damping resulting in the test point being aborted. The flutter mechanism appeared to involve wing first bending and fuselage first lateral bending modes. In this case the prediction of a low flutter speed for the full tank was not verified by flight test results. This has resulted in a carriage speed limitation as a function of external tank fuel quantity for this configuration. Figure 19 shows typical time traces of an event in which the damping ratio was estimated to be 1.15%.

Flutter flight testing was plagued with several broken latches on the engine access door underneath the aircraft. Typically each flutter mission resulted in at least two broken latches. This indicated the presence of a high level of turbulence underneath the aircraft in this configuration. Early morning flights were conducted to avoid atmospheric turbulence and ease real-time signal analysis on the strip charts. Only one mission was aborted during the entire flutter program because of unacceptable levels of turbulence. The use of in-flight refuelling early in the flutter program expedited testing, particularly for the test points requiring high drag configurations where a minimum of 5000 lbs internal fuel was required.

For most flights the AOC system was found to be effective in reducing the 5.6 Hz lateral oscillation levels in the cockpit. However, high 5.6 Hz oscillation levels were found to occur with 50% full external fuel tanks at high speed and low altitude. This will result in a carriage speed limitation for this critical configuration.

All SMI taxi/flight tests were completed uneventfully, and the final test results showed that no SMI will occur for 480 EFT carriage on the CF-18, as originally predicted by analysis.

5.5 **Loads Flight Tests**. A total of 18 flights were performed during centreline and wing loads 480 EFT carriage flight testing (9 flights each). The loads testing program was marred by several problems which will be discussed in the following paragraphs.

During the first centreline loads flight on CF-188701 the centreline adapter strain outputs were only producing 10% of their anticipated values. The strains should have ranged from 4000 to 6000 microstrain; therefore, the assumption was that the onboard instrumentation system was not functioning properly. Due to time constraints it was decided to continue on with testing using the trajectory analysis method of determining the centreline loads. After further investigation by AETE it was found that the strain value range supplied by McAir were erroneous by a factor of 10. The instrumentation gains were subsequently changed, but the adapter strain gauges still only provided limited data because they were installed in an area too far away from the primary loads paths. Therefore, the use of the calibrated centreline adapter had to be abandoned as a method of determining centreline interface loads.

The next problem surfaced after CF-188701 performed a 360 degree, full aileron deflection roll at Mach 0.7, 5000 feet MSL, with 2600 pounds (400 gallons) of fuel in the centreline tank. Post-flight analysis indicated that the centreline pylon aft attachment bolt had achieved approximately 108% of design limit load. On the same manoeuvre, the centreline pylon strain gauges, located at a critical fillet radii, exceeded the design allowable pylon strains which were analytically calculated to be 4900 microstrain. One particular strain gauge output was estimated as 10,000 microstrain which corresponds to approximately twice the strain level calculated by analysis. The pylon was removed and inspected by eddy current technique, and fortunately, no defects were found. Flight testing performed by AETE on another program with a centreline 330 EFT indicated that this problem was not unique to the 480 EFT centreline carriage but also to 330 EFT. The unexpected relatively high strains could be because of errors in the analytical stress concentration assumed in the critical area of the centreline pylon and or the centreline loads model itself. There is strong evidence that the stress concentration factor used in the analysis is incorrect, however, this is being investigated by McAir.

Because of the inconsistency between the predicted attachment loads and measured strain levels, centreline 480 EFT testing was completed with empty centreline tank only. The intention was to obtain trajectory data with empty tank, refine the aerodynamic data base, and then analytically include fuel to predict attachment loads to determine a usable flight envelope. Work was also undertaken at McAir to develop calibration equations relating left forward bolt loads to strain levels in the critical area. It should be noted that the strain levels measured with empty 480 EFTs for similar manoeuvres ranged between 4500 to 4800 microstrain.

Loads flight testing on CF-188907 proceeded much smoother until a premature failure occurred in the port inboard wing spar pylon receptacle (see Figure 20). The failure was detected when fuel was found leaking from the receptacle after the aircraft had safely landed from the last mission of the loads flight test program. The completion of the test program was severely delayed because of this major unserviceability. Since the cracked receptacle is an integral part of the #3 wing spar, the wing had to be removed and shipped to the McAir for repair. A new production port wing was installed so that testing could be resumed. The new wing had no provision for installation of test instrumentation, fortunately, the existing instrumentation in the starboard wing was similar to that in the port wing. Therefore, to minimize aircraft down time, the aircraft was rewired to make use of the starboard wing instrumentation.

The cause of the damage is presently under investigation by McAir and to date no conclusive evidence has been found to explain the failure. Several theories have been presented to explain the failure including stress corrosion cracking, static overstress, and low-cycle fatigue; nevertheless, it should be noted that the design loads for the wing spar receptacle had never been exceeded during any portion of the flight test program. During disassembly of the wing receptacle, the pylon hook wing receptacle wear plate installed on the landing which the pylon hook bears and the wall of the wing receptacle was found to be installed incorrectly. The wear plate is properly installed when the thin edge of it is located inboard. Figure 21 shows the localized surface wear resulting from the wear plates being installed incorrectly. The popular belief during the preliminary investigation was that the failure was due to a static overstress from the wear plates being installed improperly, however, this was dispelled after a strain survey on a fatigue test article at McAir showed that a correctly installed wear plate gave about 12% higher strains than one which was improperly installed. These results were suspect, therefore, McAir re-instrumented the pylon receptacle on the test article, repeated the survey and found similar results. After conducting a failure analysis, McAir's Materials and Processes laboratory had concluded that no material discrepancies were found and that there was no evidence of either stress corrosion cracking or fatigue.

Flight testing with the 480 EFT was discontinued after the wing pylon receptacle failure. In order to identify a usable envelope for the 480 EFT a relationship between the inboard pylon hook load and the critical receptacle strains had to be established. This led to the installation of 5 strain gauges in the critical area of the inboard port wing pylon receptacle on CF-188907. Using a three 330 EFT, clean outboard aircraft configuration, a two-flight test program consisting of SSPUs and RPOs was just completed at AETE. The preliminary results confirm that high failure strains do not exist in the pylon receptacle during critical hook load manoeuvres. Sufficient data was gathered showing the relationship between the inboard pylon hook load and the receptacle strains to allow 480 EFT testing to recommence.

5.6 Other Flight Test Results Although not covered in any detail by this paper, S&C and separation jettison flight tests were completed successfully. The aim of S&C testing was to investigate the handling qualities and departure resistance of the CF-18 configured with two 480 EFTs on the inboard pylons, and either a 480 or 330 EFT on the centreline station. A build-up technique of applying half then full control inputs at increasing Mach number and angle-of-attack (AOA) was employed. As mentioned previously, S&C testing was integrated with the loads testing to form one safe carriage test matrix. In general, S&C test results indicated that the 480 EFT configurations were roughly equivalent to the 330 EFT. However, around 0.6 M and 15 degrees AOA, adverse yaw resulted in large sideslip build-up with no warning to the pilot. The worse case was on the dual aircraft, CF-188907, during rolling manoeuvres using full coordinated rudder and lateral stick inputs, as well as aft stick to maintain AOA. Sideslip rapidly built-up to the test limit of 15 degrees during these test points, but no departures occurred. This characteristic is mainly a function of the flight control system implementation in that Mach/AOA range, and the sideslip excursions were only slightly worse than with the already cleared three 330 EFT configuration. McAir is presently reviewing the test data and will recommend a final clearance. Longitudinal stability and take-off characteristics were also evaluated in the heavy-weight interdiction configuration and no adverse effects were noted. Overall, no handling qualities unique to the 480 EFT were observed during the test program.

The aim of separation jettison testing was to determine the safe jettison envelope of the 480 EFT from the inboard wing and centreline pylon stations and to verify the six-degree-of-freedom data base used to predict the jettison trajectory of the 480 EFT. In summary, as predicted by analysis, each flight test resulted in a clean release with virtually no tank roll or yaw. Jettison testing of the 480 EFT from the centreline station was cancelled because of the withdrawal of the RAAF from this clearance program.

6.0 SUMMARY

Phase I of the CF-18 480 EFT stores clearance program is nearly completed in spite of the unforeseen delays which arose during the flight testing portion of the program. All flights required for flutter, AOC, SMI, carriage loads, S&C, and separation jettison have been successfully completed. At the present time 87 missions have been flown, 34 for flutter, AOC and SMI, 18 for carriage loads, 2 for wing receptacle loads, 28 for S&C, and 5 for separation/jettison. Approximately 6 more flights (5 for dynamic response and 1 for ALQ-162 vibration level testing) are required to complete this program. No major problems are anticipated during the course of these flight tests.

Data gathered from all analyses and tests to date indicate that carriage of the 480 EFT on the CF-18 aircraft is viable. Minor deficiencies found with the prototype 480 EFT during early ground testing will be corrected in the final design. As predicted analytically and by GVT, no SMI will occur from the carriage of the 480 EFTs on the CF-18. Although flutter and AOC were completed uneventfully, new flight envelope placards for certain configurations will have to be imposed as a result of flight test results. The higher than predicted loads and strains found during 480 EFT centreline carriage testing as well as the premature failure of the wing pylon receptacle on CF-188907 may result in additional flight restrictions; however, this will require further investigation by McAir. S&C and separation jettison testing were successfully completed and no 480 EFT separation jettison limitations are anticipated; however, some limitations may be required at Mach 0.6 and 15 degrees AOA. McAir is currently analyzing these results, and will provide the appropriate clearance recommendations.

Valuable experience was acquired by AETE during this stores clearance program. The experience and knowledge gained during this program will certainly be put to use in the future as planning for the follow-on testing is presently underway, with flight testing tentatively scheduled to commence later in 1988. Manufacture of the 480 EFT in Canada and associated technology transfer will most likely occur in the near future.

7.0 REFERENCES

[1] MIL-A-8871A, Airplane Strength and Rigidity Flight and Ground Operations Tests, March 1971.

[2] MIL-STD-1289A, Military Standard Ground Fit and Compatibility Test of Airborne Stores

[3] McDonnell Aircraft Company, CF-18 480 Gallon EFT Clearance Program Master Test Plan (CDRL 159), MDC B0058, 7 May 1987

[4] McDonnell Aircraft Company, CF-18 480 Gallon External Fuel Tank Cantilevered Pylon and Aircraft Ground Vibration Test Final Report, MDC B0785 (Preliminary), 5 February 1988.

[5] McDonnell Aircraft Company, CF-18 480 Gallon External Fuel Tank Final Flutter and Divergence Report, MDC B0786 (Preliminary), 5 February 1988.

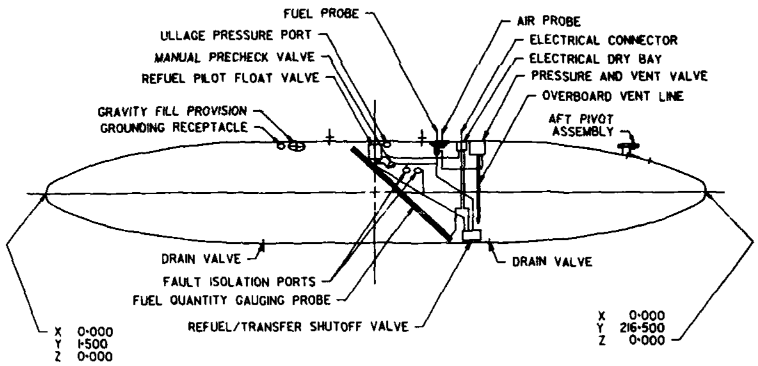


FIGURE 1 - BASIC DESCRIPTION OF 480 EFT

PHYSICAL COMPARISONS EFT'S		
330	VS	480
ALL METAL/SURVIVABLE	WRAPPED	COMPOSITES
188·4/189 IN	LENGTH	215 IN
28·2/28·8 IN	DIAMETER	31·9 IN
220/290 LBS	DRY WEIGHT	250 LBS
330 GALS 2,244 LBS	USABLE FUEL	480 GALS 3,264 LBS

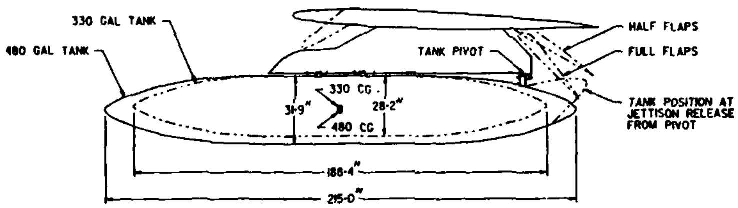


FIGURE 2 - COMPARISON OF EXTERNAL FUEL TANKS

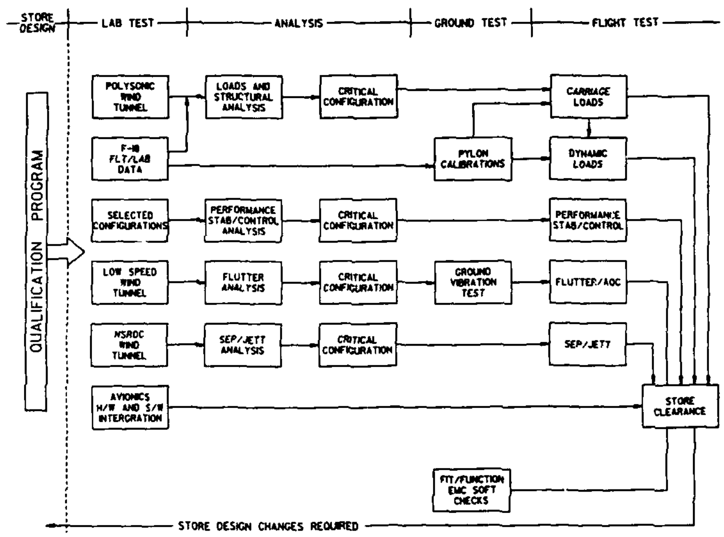
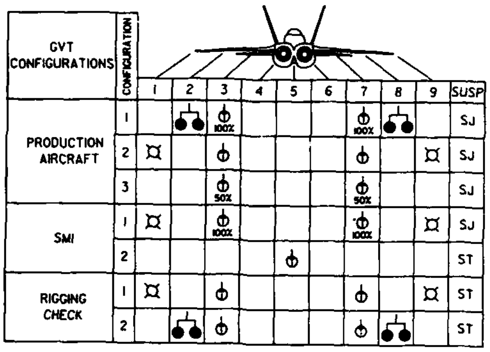


FIGURE 3 - 480 EFT STORES CLEARANCE PROCESS



⊕ - 480 USG FUEL TANK   SJ - SOFT JACKS
⌑ - AIM-9L/M MISSILE   ST - SOFT TIRES
● - MK 82

FIGURE 4 - CRITICAL GROUND VIBRATION TEST CONFIGURATIONS

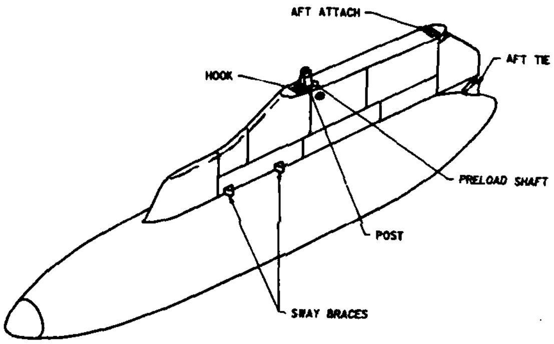


FIGURE 5 - WING/PYLON/TANK INTERFACES

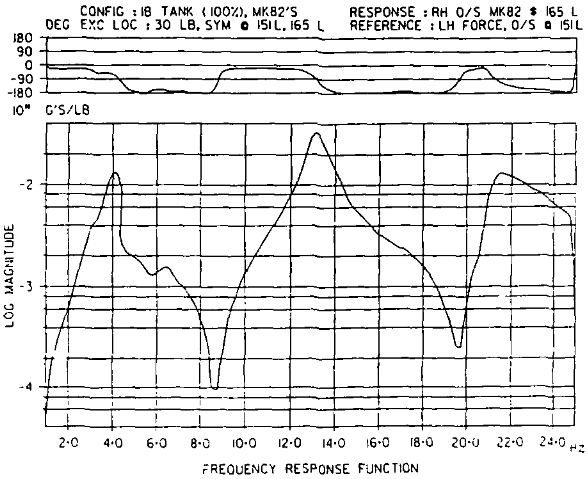


FIGURE 6 - TYPICAL TRANSFER FUNCTION PLOT

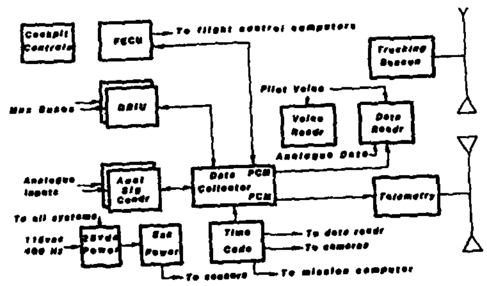


FIGURE 7 - CF-18 BASIC DATA ACQUISITION SYSTEM

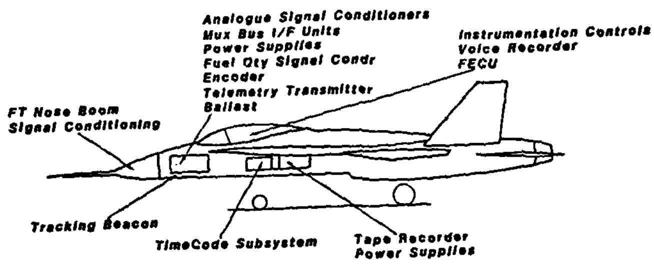


FIGURE 8 - BASIC CF-18 AIRCRAFT INSTRUMENTATION

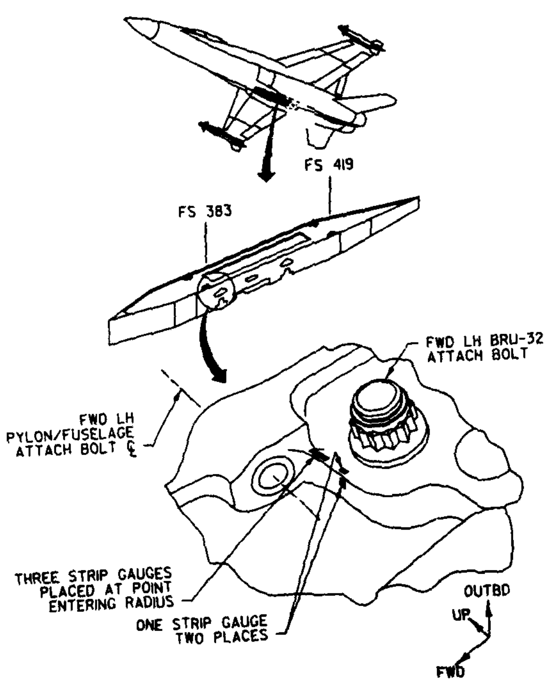


FIGURE 9 - CENTRELINE PYLON INSTRUMENTATION

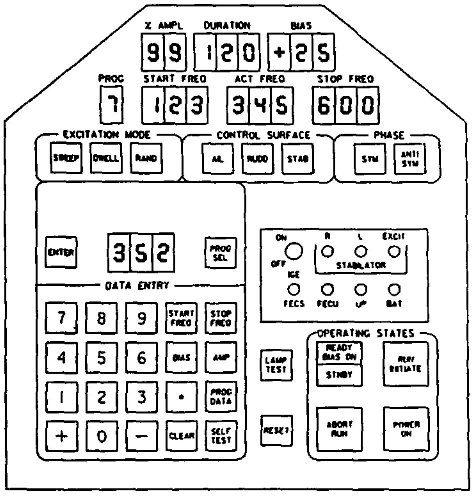
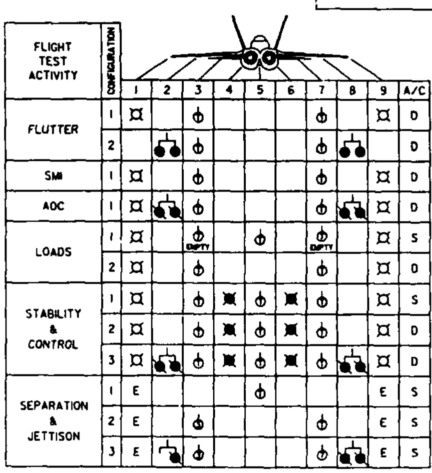


FIGURE 10 - CF-18 FLUTTER EXCITER CONTROL UNIT

FLIGHT TEST ACTIVITY	RISK FACTOR	DAMAGE CAT	COMMENTS
FLUTTER/AOC	LOW	A	REAL TIME MONITORING EXPLOSIVE NATURE OF FLUTTER
LOADS	LOW	C	REAL TIME MONITORING INTERNAL DAMAGE DUE TO PYLON OVERLOAD
STABILITY/CONTROL	MEDIUM	A	REAL TIME MONITORING INACCURATE TRACKING OF DEPARTURE PARAMETERS
DYNAMIC LOADS	LOW	C	REAL TIME MONITORING INTERNAL DAMAGE DUE TO PYLON/WING DYNAMIC LOADING
SEPARATION/JETTISON	LOW	A	NO REAL TIME MONITORING STORE TO STORE OR STORE TO WING CONTACT
PERFORMANCE	NO	—	—

FIGURE 11 - FLIGHT TEST RISK FACTORS



Ⓣ - 480 FUEL TANK
⌑ - AIM-9L/M MISSILE
● - MK 82
E - EXTERNAL CAMERAS
Ⓣ - 330 FUEL TANK
- MK 83
- AIM-7
D - DUAL SEAT AC 907
S - SINGLE SEAT AC 701

FIGURE 12 - FLIGHT TEST AIRCRAFT CONFIGURATIONS

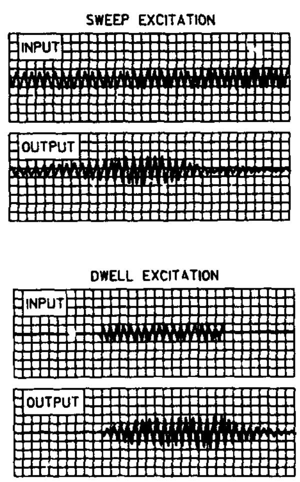


FIGURE 13 - SWEEP/DWELL SINE EXCITATION

GROUP	TEST SEQ	ALT (FT MSL)	MACH NO	FUEL STATE	AIRCRAFT ATTITUDE	AILERON SYM/ANTI	EXCITATION SWEEP/DWELL
A	2	15K	0.95	FULL	LEVEL	ANTI	SW/DW
A	3	10K	0.90	FULL	LEVEL	ANTI	DW
B	4	10K	0.95	FULL	LEVEL	ANTI	SW/DW
C	7	10K	9.95	25-50 75%	LEVEL	ANTI	SW/DW
C	8	10K 2K	0.95	50%	10 CLIMB	ANTI	DW
C	9	10K 2K	0.95	50%	10 DIVE	ANTI	DW
A	1	5K	0.40	FULL	LEVEL	ANTI	SW/DW
B	5	6K	0.90	FULL	LEVEL	ANTI	DW
B	6	6K	0.95	FULL	LEVEL	ANTI	SW/DW
D	10	4K	0.90	FULL	LEVEL	ANTI	DW
D	11	4K	0.95	FULL	LEVEL	ANTI	SW/DW
E	12	6K/4K	0.95	FULL	PUSHOVER	0.5g WITH ANTI-SYMMETRIC DWELLS	
E	13	6K/4K	0.95	FULL	SLOW WIND-UP TURN (WUT), 1 TO 7.5g WITH LATERAL STICK RAPS		
F	14	4K	0.7 TO Vmax	FULL	LEVEL	ANTI	DW
F	15	30K to 10K DIVE	Vmax	FULL		ANTI	DW

FIGURE 14 - FLUTTER TEST MATRIX (CONFIG 2)

MANOEUVRES	MACH	ALTITUDE (MEAN SEA LEVEL) 20K	15K	10K	6K	5K
SYMMETRIC PULL-UP TO 7.5G >2600 LBS CENTRELINE EFT	.9			X		X
	.95					X
	V_L					X
SYMMETRIC PUSHOVER TO -3.0G	.85			X		X
	.9			X		X
HALF STICK/FULL STICK 6.0G ROLLING PULL-OUT (RPO) EMPTY CENTRELINE EFT	.6					X
	.9		X	X		X
	.95			X		X
	V_L			X		X
HALF STICK/FULL STICK 3.0G RPO EMPTY CENTRELINE EFT	.6			X	X	
	.8			X	X	
	.9				X	
	.95				X	
HALF STICK/FULL STICK -1.0G 180° ROLL EMPTY CENTRELINE EFT	.65		X			
	.75		X	X		X
HALF/FULL PEDAL RUDDER KICKS	.6	X	X	X		X
	.7	X	X			
	.8		X			

FIGURE 15 - AOC FLIGHT TEST MATRIX

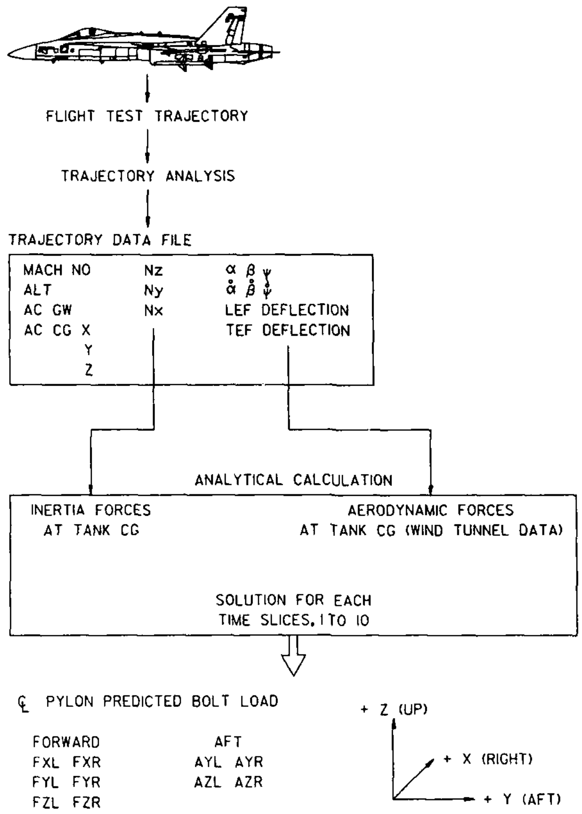
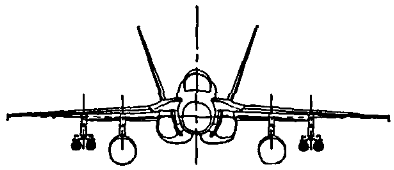


FIGURE 16 - CENTRELINE PYLON LOADS PREDICTION PROCESS

TEST POINT	MACH NUMBER	ALTITUDE (FT MSL)	MANOEUVRE
1	0.70 - 0.93 (or Vmax)	6K, 4K	SLOW 1G ACCELERATION LATERAL STICK RAPS
2	0.75	6K, 4K	SLOW WIND-UP TURN (WUT), 1 TO 7.5G LATERAL STICK RAPS
3	0.81	6K, 4K	SLOW WUT, 1 TO 7.5G LATERAL STICK RAPS
4	0.83	6K, 4K	SLOW WUT, 1 TO 7.5G LATERAL STICK RAPS
5	0.85	6K, 4K	SLOW WUT, 1 TO 7.5G LATERAL STICK RAPS
6	0.88	6K, 4K	SLOW WUT, 1 TO 7.5G LATERAL STICK RAPS
7	0.91	6K, 4K	SLOW WUT, 1 TO 7.5G LATERAL STICK RAPS
8	0.93 (or Vmax)	6K, 4K	SLOW WUT, 1 TO 7.5G LATERAL STICK RAPS
9	0.93 (or Vmax) - 0.70	6K, 4K	SLOW WIND-DOWN TURN, 3.5 TO 4.5G LATERAL STICK RAPS
10	0.60	6K, 4K	0.5G PUSHOVER LATERAL STICK RAPS
11	0.88	6K, 4K	0.5G PUSHOVER LATERAL STICK RAPS

FIGURE 17 - CF-188701 LOADS TEST MATRIX (CONFIG 1)

TIP MISSILES OFF, 2 MK-82 OUTBOARD, 100% 480 EFT INBOARD



ANALYTIC MODE DESCRIPTION	SYMMETRIC			ANTI-SYMMETRIC		
	ANALYSIS HZ	GVT HZ	RIG HZ	ANALYSIS HZ	GVT HZ	RIG HZ
TANK ROLL	2.45	2.51	*NM	3.05	3.14	NM
TANK PITCH	6.00	6.17	6.19	5.75	5.72	5.89
TANK YAW	5.92	6.04	NM	5.93	6.42	NM
OUTBOARD STORE ROLL	4.05	4.41	4.45	4.63	4.73	4.45
OUTBOARD STORE PITCH	7.88	8.14	8.43	7.60	8.60	8.77
OUTBOARD STORE YAW	12.49	13.49	13.19	12.53	13.36	13.37
WING 1ST BENDING	5.53	5.91	5.77	8.31	8.08	8.12
WING 1ST TORSION	17.50	17.36	16.42	17.47	17.19	16.96
WING 2ND BENDING	11.90	12.60	NM	14.78	NM	NM
FUSELAGE 1ST BENDING	9.42	10.33	NM	8.80	9.74	9.31

*NM - NOT MEASURED

FIGURE 18 - SUMMARY OF GVT RESULTS

CF-18 FLUTTER PERFORMANCE

A/C SERIAL #	CF188907	PLOT SET #	
FLT/GND #:	MISSION 23	DATA RUN #/REC #	1
A/C CONFIGURATION	# 2	MANEUVER:	TP#14C:4K/VMAX 480T FUEL BURN 45/ - EOM:8.5Hz DWELLS
MISS DATE:	29-OCT-87		

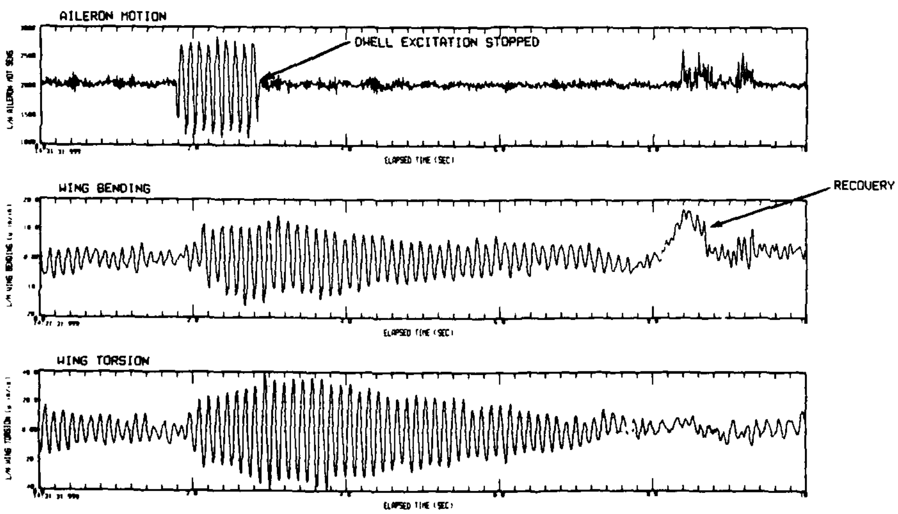


FIGURE 19 - RESPONSE OF WING STRAIN GAUGES AT LOW DAMPING

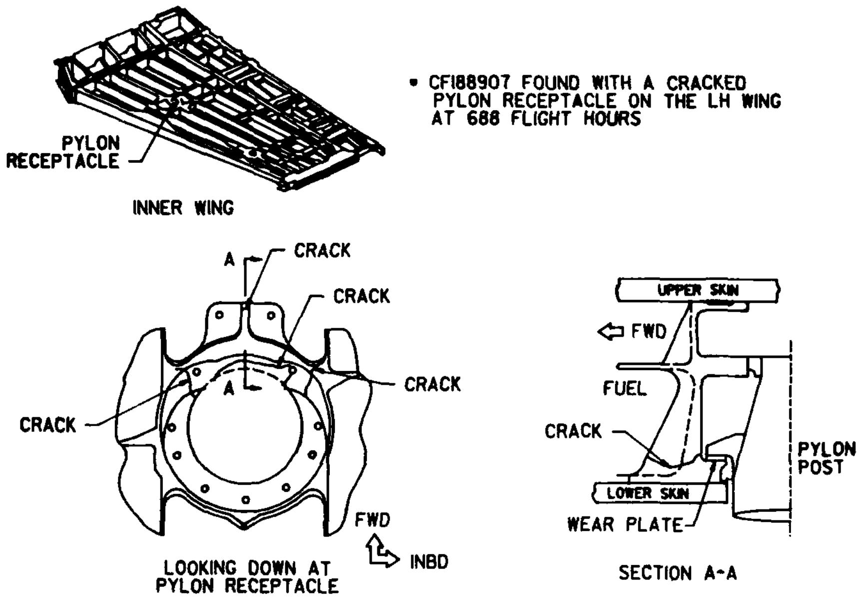


FIGURE 20 - INNER WING PYLON RECEPTACLE CRACK

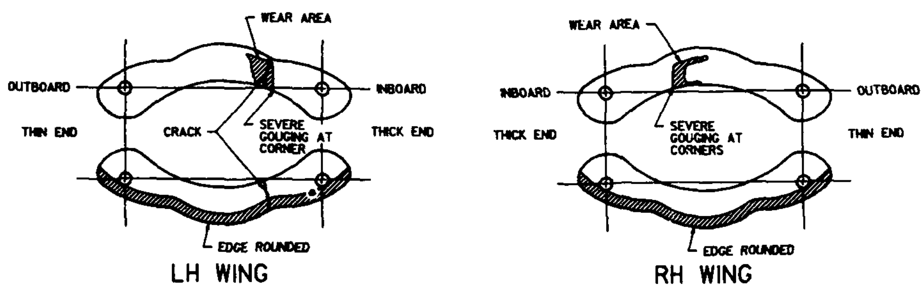


FIGURE 21 - WEAR PLATE INSTALLATION

FLOW VISUALIZATION TECHNIQUES FOR FLIGHT RESEARCH

David F. Fisher and Robert R. Meyer, Jr.
NASA Ames Research Center
Dryden Flight Research Facility
P.O. Box 273
Edwards, California 93523-5000
U.S.A.

SUMMARY

In-flight flow visualization techniques used at the Dryden Flight Research Facility of NASA Ames Research Center (Ames-Dryden) and its predecessor organizations are described. Results from flight tests which visualized surface flows using flow cones, tufts, oil flows, liquid crystals, sublimating chemicals, and emitted fluids have been obtained. Off-surface flow visualization of vortical flow has been obtained from natural condensation and two methods using smoke generator systems. Recent results from flight tests at NASA Langley Research Center using a propylene glycol smoker and an infrared imager are also included. Results from photo-chase aircraft, onboard and postflight photography are presented.

NOMENCLATURE

C_p	pressure coefficient
g	load factor normal to longitudinal axis of aircraft
HARV	high alpha research vehicle
h_p	pressure altitude, km (ft)
IR	infrared
LE	leading edge
LEX	leading edge extension
M	Mach number
NLF	natural laminar flow
R	reattachment location
R_n	Reynolds number based on wing reference chord
S_1	primary vortex separation location
S_2	secondary vortex separation location
TACT	transonic aircraft technology
TE	trailing edge
V	velocity
VSFTE	variable sweep flight transition experiment
x/c	ratio of chordwise distance from leading edge to local chord length
α	alpha, angle of attack, deg
δ	boundary layer thickness, cm (in.)
η	ratio of spanwise distance from aircraft centerline to wing semispan
Λ	wing leading edge sweep angle, deg

Subscripts

i	indicated
max	maximum
∞	freestream

1. INTRODUCTION

The visualization of airflow on and about models in wind tunnels has been used for many years to complement other sources of aerodynamic data to help interpret test results. Methods of flow visualization commonly used in wind tunnels include schlieren photography (ref. 1), oil flows (ref. 2), minitufts (ref. 3), vapor screen (refs. 4 and 5), and smoke (ref. 6). From this array of methods, one can choose specific techniques from which shocks can be visualized, flow direction and regions of separated flows can be determined, boundary layer transition location defined, and vortices identified. Recently, low-speed water tunnels have also been used extensively to visualize simulated airflows (refs. 7 and 8). These techniques and others are well covered in reference 9.

Similarly, a wide variety of in-flight flow visualization techniques for aerodynamic experiments ranging from visualization of boundary layer characteristics to vortical flows have been used at the Dryden Flight Research Facility of NASA Ames Research Center (Ames-Dryden) and its predecessor organizations to help interpret and understand the results of flight tests. These visualization techniques usually give a more global picture of airflow and sometimes are the only way the data can be obtained in flight. Many of the techniques are associated with on-surface flow visualization, including flow cones and tufts, oil flows, a liquid crystal technique, sublimating chemicals, and an emitted fluid technique. Other techniques involve off-surface flow visualization, including two types of smoke generator systems and natural condensation.

The authors discuss the previously mentioned techniques and two other methods used at the NASA Langley Research Center. The reader will be directed to supporting reference material for these in-flight techniques. Verification of the results interpreted from the flow visualization techniques will be made by comparison to other in-flight or ground facility measurements as appropriate.

2. FLOW VISUALIZATION TECHNIQUES

The flow visualization techniques described in the following sections generally require a significant amount of care and fine tuning to get the best results. Ground tests, exploratory flights, or both are usually required to determine the proper viscosity of oil, thickness to apply the liquid crystals, flow rates of emitted fluid, surface color, or proper chemical for sublimation. Flights need to be allocated for development of technique and refinement in the flight planning to ensure favorable results. In addition, careful examination, supporting data, and experience are needed to interpret the results properly.

2.1 Boundary Layer Transition and Shock Visualization Techniques

The techniques that follow are generally used to detect boundary layer transition from laminar to turbulent flow; however, some of the techniques have wider uses. Most of these techniques are dependent upon surface differential temperatures caused by the difference in the laminar and turbulent flow wall recovery factors. Heat sinks such as wing ribs and spars or internal fuel tanks can cause localized cool spots and make these techniques nearly unusable. Smooth insulated fiberglass gloves are generally used for best results.

2.1.1 Oil Flows

Oil flows (refs. 10–13) are obtained by applying an oil mixed with a coloring agent to the surface of an aircraft to study surface flow characteristics. A wide variety of fluid mechanics information can be interpreted from oil flow patterns including location and relative strength of normal shocks, location of boundary layer transition, and identification of areas of separation. These surface flow conditions cause the oil coating to thin, thicken, or puddle, resulting in distinct contrast changes in the oil-coated surface. Various oil viscosities and coloring agents can be blended to provide results at particular flight conditions, such as speed or altitude, or to provide adequate contrast on different colored surfaces.

Results from the oil flow study on the F-111 tactical aircraft technology airplane (TACT)(ref. 12) are presented in figure 1. The study was conducted on a partial span laminar flow glove installed on the left wing panel that was instrumented with a chordwise row of surface static pressure orifices and a boundary layer rake as shown in figure 1(a). For this study, SAE 80-W-90 oil was mixed with powdered black graphite in the proportions of four parts of oil to one part of graphite, by volume. The mixture was applied to the surface with a paint brush approximately 30 min prior to takeoff. Documentation was obtained from chase aircraft photographs.

A dark curved line across the test surface just aft of the midchord can be observed in figure 1(a). This line was attributed to a normal shock wave on the test surface, which causes the oil to build up or puddle, at the location of the shock. This is verified by the corresponding pressure distribution in figure 1(b). The pressure distribution shows a rapid increase in pressure (more positive C_p) at the same location ($x/c = 0.65$), which also indicates the presence of a shock wave. Meyer and Jennett (ref. 12) showed that the darkness of the line in the oil, indicating a shock, could be correlated directly to relative shock strength as determined from simultaneously obtained pressure distribution data.

A more subtle observation that can be made from oil flows is the location of boundary layer transition from laminar to turbulent flow. An example of this is presented in figure 2 from reference 12. In figure 2(a), note the distinction between a light area forward of the 20-percent chord, where there appears to be little or no oil, and a uniform slightly darkened area aft of the 20-percent chord. The light area is considered to be the region of laminar flow, and the darkened area is considered to be turbulent flow. This interpretation is supported by the corresponding pressure distribution of figure 2(b), as well as the boundary layer data of figure 2(c), from reference 12. In figure 2(b), a favorable pressure gradient exists from the leading edge ($x/c = 0$) to approximately the 20-percent chord ($x/c = 0.20$), where the pressure gradient becomes unfavorable and the boundary layer should transition to a turbulent condition. The variation of upper surface boundary layer thickness, obtained from the boundary layer rakes as a function of angle of attack, is shown in figure 2(c) and described in reference 12. The lines on the figure indicate faired calibration results obtained using "trip strips," and the circular symbol indicates the boundary layer thickness when the oil flow pattern of figure 2(a) was photographed. By referring to the location of the circular symbol with respect to the different calibration lines, the point of transition can be determined, independently of the oil flow, at approximately 23-percent chord ($x/c = 0.23$), confirming the interpretation of the photograph of figure 2(a).

A summary of results using oil flows on seven different aircraft from low speed to supersonic speeds is given in reference 13. Reference 13 also provides details on appropriate oil viscosities and coloring agents for particular applications. With proper flight planning, especially for boundary layer transition location determination, multiple test conditions can be obtained on one flight. Test points with the expected transition location near the leading edge need to be flown before tests points with the transition location far aft. It was also found that this technique was limited to approximately 7600 m (25,000 ft) because the oil becomes too viscous to be of use at the colder temperatures of greater altitudes. Documentation of the oil flow patterns is normally obtained from chase aircraft photographs, although photographs have been obtained in real time from on-board cameras.

2.1.2 Liquid Crystals

Liquid crystals similar to those commonly used in desktop thermometers have been used recently in flight (refs. 14–16) primarily for boundary layer transition visualization. Liquid crystals exposed to fluid flow on the aircraft test surface change color with shear forces or temperatures associated with various fluid mechanic phenomena, such as laminar to turbulent transition or normal shocks.

The liquid crystals are applied to the aircraft by first thinning the liquid crystal solution with solvent and then spraying a very thin coat on the test surface prior to takeoff. The test surface must be a dark color to provide suitable contrast. Holmes and others (ref. 14) reported that a flat black surface provided the best results, although other dark surfaces often provide adequate results. In order to obtain optimum color change with the liquid crystal technique, lighting and viewing angle are critical. Optimum results were obtained with the sun positioned behind the viewer (photo aircraft) and the viewer as close to perpendicular to the test surface as possible.

Results (ref. 16) are presented from the F-14 variable sweep flight transition experiment (VSFTE) program. For those tests, a pressure sensitive liquid crystal (so named by the manufacturer) was applied to laminar flow wing gloves on an F-14 aircraft. The liquid crystals were applied to the surface approximately 1 hr prior to takeoff. Documentation was obtained by photographs and video from a photo chase aircraft.

Examples of liquid crystal patterns on the F-14 airplane's left wing glove are shown in figure 3. Transition is indicated by an abrupt change in color of the liquid crystal pattern due to the discontinuous change in temperature and shear at boundary layer transition. The colors themselves are not important, since they change with viewing angle as well as ambient temperature (altitude).

Figure 3(a) represents a case where transition occurred forward on the test section, probably due to cross-flow disturbance generated by leading edge sweep and is indicated by the sawtooth pattern at approximately 5- to 10-percent chord. Figure 3(b) represents a case where transition occurred about midchord, due to the laminar boundary layer encountering an adverse pressure gradient and is indicated by a uniform line at about 30-percent chord. One other observation in figure 3(b) is the presence of a "transition wedge" near the outboard edge of the test section, probably due to an insect impact or dust stuck in the oily liquid crystal mixture.

Liquid crystal and hot-film transition data obtained at the same time intervals are compared in figure 4 from reference 16. The figure shows that transition location indicated by the liquid crystal flow patterns and hot film agreed to within less than 5-percent chord.

A concern using liquid crystals was the effect, if any, the liquid crystal solution on the test surface might have on the laminar to turbulent boundary layer transition, or in other words, the possibility that the liquid crystal solution would affect or bias the fluid mechanic process. Transition data obtained from the hot films with and without the liquid crystal solution on the test surface are compared in figure 5. Transition is plotted as a function of angle of attack at two representative altitudes. At the lower altitude (higher unit Reynolds number), a variation in transition location of up to 30-percent chord is evident. At the higher altitude (lower unit Reynolds number), there is much less difference, only about 5 percent of chord.

2.1.3 Infrared Imaging

Infrared (IR) imaging is a relatively new flow visualization technique, recently used for nonintrusive visualizing of laminar to turbulent boundary layer transition (refs. 17 and 18). Flight tests with an IR imager were recently conducted at NASA Langley Research Center on a T-34C aircraft modified with an NLF(1)-215F airfoil partial span glove (ref. 18). The sensitivity of the IR imager used in these tests is shown in figure 6. As the temperature of the target surface is increased, the imager becomes more sensitive, so that at 30°C (86°F), the imager can detect differential temperatures of 0.1°C (0.18°F).

The IR camera and display unit were mounted in the T-34C aircraft (fig. 7) with the wing glove in the field of view. The flight tests were conducted with the canopy open so that no loss of image sensitivity occurred from the infrared absorption by the Plexiglas window. During the research flights, an experimenter in the aft cockpit adjusted the imager so that temperature differences on the glove could be observed.

An example of typical results obtained from the IR imager on the black surface during daylight is shown in figure 8. The IR photograph shows that the laminar region (light color) is warmer than the turbulent region. This result is due to the increased convection coefficient of the turbulent boundary layer. The free-stream air temperature is lower than the natural laminar flow (NLF) glove surface due to solar heating, and the turbulent flow region is cooled at a faster rate than the laminar region.

On night flights, the surface temperatures of the NLF glove were much lower because there was no solar radiation effect. Due to the lower temperatures, the sensitivity of the imager was reduced as mentioned previously. As a result, the steady-state level flight test techniques used at night were unsuccessful at low speed. Transition data could be obtained by cold soaking the glove at high altitude, then descending into warmer air so that convection tended to heat the wing as the aircraft descended. Using this technique, the color patterns on the IR images are reversed from that seen during day flights. In figure 9, the laminar areas are dark and the turbulent areas are light.

2.1.4 Sublimating Chemicals

The sublimating chemical technique (refs. 19–21) is used primarily for laminar to turbulent boundary layer transition. The greater heat transfer in the turbulent boundary layer results in higher surface temperatures than in the laminar boundary layer. This elevated temperature and shear associated with turbulence of the boundary layer at the location of transition cause a suitable chemical to increase its rate of sublimation in the turbulent region with respect to that of the laminar region. Consequently, the chemical is removed in the turbulent area before the chemical in the laminar area is appreciably affected.

The chemicals are applied to an aircraft surface by dissolving them in a solvent and then spraying to an appropriate thickness. The solvent evaporates, leaving a crystal structure that can be smoothed if necessary prior to takeoff. The chemicals are white and provide the best contrast on dark surfaces, although dyes, such as food coloring, can be added to the chemicals to provide contrast on light surfaces. Reference 20 contains an excellent summary of various chemicals and application techniques at high speeds and reference 21 summarizes the technique for low speeds. Both sources contain tables and figures that can be used as an aid in choosing particular chemicals for specific applications.

The test aircraft is normally established at the desired test conditions as quickly as possible and then maintains those conditions until the chemicals in the turbulent region have sublimated, showing the desired flow patterns. Normally, the test is documented with photographs taken on the ground, postflight, although photographs can be taken in real time from a chase aircraft or on-board cameras. Unlike the previously described methods, only one flow condition or test point can be obtained per flight.

An example of this technique conducted at speeds near Mach 1.8 on the left wing of an F-104 (ref. 20), using phenanthrene, is shown in figure 10. The aluminum wing was painted black for best contrast with the white chemicals. The transition location was observable as shown in the figure. Note that some of the sublimable chemical along the wing-fuselage juncture and in the rear portion of the wing did not sublimate. This is caused by large structural members that act as heat sinks, thus preventing the wing skin from attaining temperatures high enough to cause a high rate of sublimation.

Another example, this time from the right-hand wing of the F-104 at Mach 2 (ref. 20), is given in figure 11. The right wingtip panel of the F-104 was covered with a thin fiberglass glove, and fluorene was used as the sublimating agent. The test conditions required up to 4 min for the indication of boundary layer transition to be complete. Resistive thermometers can be seen in the photograph (fig. 11) of the wing imbedded beneath the fiberglass and gave good correlation with the sublimation technique.

2.2 Flow Direction and Boundary Layer Separation Visualization Techniques

2.2.1 Flow Cones and Tufts

Tufts, usually made of nylon or wool, have been successfully used in many studies (ref. 22) and are considered a standard method for in-flight flow visualization of surface flow direction and indicators of separation. Flow cones (refs. 23 and 24) were developed to overcome some disadvantages of tufts, such as instability or whipping of tufts (not always related to any feature of the flow) and difficulties in photographing tufts at distances between the photo aircraft and test or research aircraft.

Flow cones are rigid, narrow, hollow conical shapes, attached to an aircraft surface with a short string or chain at the apex of the cone. Like tufts, they are lightweight and align themselves with the local surface flow if the local dynamic pressure is sufficient. A typical tuft and flow cone are compared in figure 12. The flow cones are available in a variety of materials and colors. Photographs are usually taken from a chase aircraft to document the flow cone patterns in flight, however, in some cases, photographs are taken from cameras onboard the test or research aircraft. The flow cones are usually more visible than the tufts since they are larger in diameter and can be covered with reflective material. Though lightweight, their weight (0.7–0.9 gram) can be a disadvantage particularly on the lower surface of wings at low dynamic pressure. In addition, they would not be acceptable in front of engine inlets, whereas, wool tufts probably would be acceptable.

Flow cones and nylon tufts were used recently on the upper surface of left wing and canard of the X-29A aircraft. In figure 13, a comparison of the contrast of a tuft and a flow cone on the X-29A airplane can be seen. These particular photographs represent a slow flight condition for the X-29A aircraft. It was particularly difficult for the photo chase aircraft to photograph at close distance to the test aircraft. Representative tuft and flow cones are identified in the figure. As can be seen in figure 13, the flow cones provided significantly improved contrast and resolution as compared to the tufts.

As a verification of the ability of tufts or flow cones to indicate areas of separated flow, figure 14 shows wing and canard surface static pressure distributions corresponding to the conditions shown in figure 13.

Separated flow in the pressure distributions is indicated by failure of the upper and lower surface pressures to return to a common level at the trailing edge. These trends are evident in the wing upper surface pressure distribution (fig. 14(a)), near midspan ($\eta = 0.49$) on the trailing edge over approximately the aft 10 percent of the wing chord. This interpretation of the wing pressure distributions is consistent with flow cone patterns of figure 13, where the flow cones on the trailing edge are pointed inboard approximately parallel to the trailing edge of the wing. In figure 13, neither the flow cones on the canard nor the pressure distributions of figure 14(b) indicate any flow separation.

2.2.2 Emitted Fluid Technique

Fluids for flow visualization of surface streamlines and areas of separated flow have been used successfully at Douglas Aircraft (refs. 25 and 26), and at Ames-Dryden (ref. 27). This method consists of emitting a small quantity of propylene glycol monomethyl ether (PGME) containing toluene-based dye out of small-diameter surface tubes or orifices on the aircraft skin while the aircraft is stabilized. The flight conditions are held constant for 1–2 min while the fluid evaporates and the dye sets or dries. The dye patterns of the surface streamlines and separated flow regions are documented by photographing the surfaces on the ground after each flight.

An example of surface flow visualization technique on the F-18 forebody for an angle of attack of 30° is shown in the postflight photographs of figure 15. An interpretative cross-sectional flow model (ref. 28) showing the structure of the flow about the forebody for symmetrical flow is illustrated in figure 16. Two primary and two secondary vortices are formed on the leeward side with the separation and reattachment lines noted. As can be seen in figure 15, the remaining dye from this technique clearly showed the surface flow streamlines. Where the streamlines merged, the flow lifted from the surface and rolled up into streamwise vortex cores. Along the streamlines where the flows merge, the separation lines for the vortices were defined. Both the primary and secondary separation lines can be seen. A reattachment line on top of the forebody where the streamlines diverge is also noted.

For both the Douglas and Ames-Dryden tests, a red dye was used with a white painted surface for good contrast. One data point was obtained per flight in this manner. This method is particularly useful when a photo-chase airplane is not available or not practical.

A similar method of using colored ethylene glycol and water on a large commercial transport has been reported (ref. 24). The liquid was emitted from small-diameter tubes distributed upstream of the surface under study. The fluid dispensing cycle took approximately 10 min. The principal data acquisition is air-to-air photography; however, ground inspection of the pattern was also useful to observe and document small details.

2.3 Vortical Flow Visualization Techniques

2.3.1 Natural Condensation Flow Visualization

Flow visualization of vortical flow, expansion, and shock waves, as well as attached and separated flows, can sometimes be observed in flight during periods of high humidity. When the humid air expands around an airplane, it can condense, become visible, and illustrate certain flow patterns such as wing pressure fields, wingtip vortices, and shock waves. A thorough review of this subject with clear examples of the phenomena is given in reference 29. No electromechanical or pyrotechnic devices are needed except the photographic or video recorders.

An example of natural condensation flow visualization on NASA's TACT F-111 aircraft showing the spanwise "gull" pattern over the swept wing and the wingtip vortices is shown in figure 17 (ref. 29). The high velocities and low pressure over the F-111 wing and in the vortices cause the static air temperature to drop and raise the local relative humidity. When the local relative humidity becomes high enough, vapor in the air can condense and become visible.

Similarly, flow visualization in the form of visible vortices due to natural condensation from the leading edge extensions (LEX) on the F-18 high alpha research vehicle (HARV) was noted on some of the initial flights. These vortices (fig. 18) were documented with onboard photography with the flight conditions noted. Note also that after the vortex breaks down, the vortex flow field is no longer visible.

Natural condensation can also occur in flight at supersonic speeds (ref. 29). Vortices can be seen trailing from the canard tips and the outer wing dihedral break of the XB-70 aircraft (fig. 19). Condensation patterns are also visible over the canopy and canard, and a shock pattern can be seen at the trailing edge elevon.

2.3.2 Smoke Generator Systems

Because of the unreliability of natural condensation for flow visualization, smoke generator systems have been developed for visualizing vortical flows in flight (refs. 30–37). Primarily two types of systems have been used, a cartridge system and glycol or oil vaporizer system. Smoke generator systems commonly used at airshows are not applicable for flight research since these systems generally inject oil into the exhaust of the engine and not into the vortex system of interest. For the smoke generator systems used for flight research, smoke is emitted from a duct near the origin of a vortex, allowing the smoke to be entrained in the vortex and permitting a view of the vortex path. Three systems, one cartridge-based system and two vaporizer systems, are discussed as follows.

2.3.2.1 Smoke cartridge systems

Cartridge-based pyrotechnic smoke systems have been used on aircraft to visualize the vortex flow on highly swept wings and leading edge extensions (refs. 30 and 31). Military chemical smoke cartridges were used on the HP115 aircraft (ref. 30) to visualize the vortex on the highly swept wing. A single smoke cartridge and tar trap were mounted externally below the wing near the apex with a duct from the tar trap wrapping around the wing leading edge to the upper surface.

On the NASA F-18 HARV, a smoke generator system (ref. 31) was developed that uses smoke cartridges (fig. 20) designed for this application by the U.S. Army Chemical Research, Development, and Engineering Center, in Aberdeen, Maryland. This system is described in detail in a proposed publication.[1] The smoke cartridges are 6.1 cm (2.4 in.) in diameter and 11.9 cm (4.7 in.) long and were designed to produce nontoxic smoke at a high rate while minimizing fire, detonation or handling hazards. The smoke generator system has been designed into two separate housings, one of which is shown in figure 21. Each housing contains six cartridges and fits within the fuselage of the aircraft. To ensure a safe system, each cartridge is contained within a stainless steel cylinder with a pressure relief to external louvers in the event of high unexpected pressures. The cartridges are temperature controlled at 27°C (80°F) during flight to ensure reliable firing. The six cylinders in each housing are manifolded to a common 2.54 cm (1 in.) diameter duct that exits on either the upper LEX surface near the apex or on the aircraft nosecone. Ducting lengths up to 3 m (9 ft) have been used successfully in flight.

The smoke cartridge system selected for the F-18 aircraft provided a rapid and dense smoke production rate for minimal packaging and power requirements compared to other concepts. Multiple packaging allows from one to six cartridges to be ignited simultaneously, offering a wide range in smoke production rates.

Because the smoke from this system is white, the upper surface of the F-18 HARV was painted a flat black trimmed in gold for good photographic contrast. Four video cameras and one 35 mm camera (fig. 22) were installed on the F-18 airplane to document the results during day flights.

Typical examples of flow visualization of the LEX vortices on the F-18 HARV using the cartridge smoke system are shown in figure 23. As the angle of attack increases, the breakdown location moves forward. An advantage that the smoke has over natural condensation is that the smoke persists beyond the vortex breakdown location.

[1] Proposed NASA Technical Memorandum, *A Smoke Generator System for Aerodynamic Flight Research*, by David M. Richwine, Robert E. Curry, and Gene V. Tracy.

The cartridge smoke system provided visible smoke using two cartridges at a time at speeds from 100 to 140 knots. At speeds up to Mach 0.6, three cartridges were necessary. The cartridges ignited reliably at altitudes as high as 9100 m (30,000 ft). There was a time delay of 10 and 30 sec between ignition and useful smoke production. Once started, a steady stream of useful smoke could be expected for 30 sec when using two or more cartridges. The vortex breakdown data using the F-18 smoke generator system showed close agreement with the natural condensation data. The breakdown location for the LEX vortex as a function of angle of attack (fig. 24) shows good agreement using either natural condensation or smoke techniques. This correlation indicates that the injection of smoke and the external duct did not perturb the flow field significantly to alter the vortex breakdown location.

2.3.2.2 Glycol vaporizers

Researchers at the NASA Langley Research Center have developed a smoke system in which propylene glycol is vaporized into a white smoke with electric heaters on an F-106B aircraft (refs. 32 and 33). Propylene glycol is pumped to the vaporizer containing six 1000-W heaters from an 11.4-L (3 gal) tank onboard the airplane. From the vaporizer, the glycol vapor passes through an insulated and heated pipe to an uninsulated probe tip mounted slightly below the wing near the wing leading edge–fuselage junction. This system provided smoke for approximately 1 hr. The persistence of the propylene glycol smoke from the F-106B during a day flight is shown in figure 25.

This system was usually flown at night with a mercury-arc lamp light sheet system illuminating a nearly perpendicular cut through the vortex core. The light sheet system was mounted in the midfuselage (fig. 26). Documentation was provided by a video camera mounted near the cockpit. While the F-106B aircraft wing was bare aluminum, the authors recommend a flat black paint for improved photographic contrast.

An example of flow visualization using the glycol vaporizer and the light sheet system on the F-106B airplane during a night flight is shown in figure 27(a). The view is from the video camera near the cockpit looking aft at the left wing. The light sheet system illuminates just a slice through the vortices.

The black and white videotape data using the light sheet at night on the F-106B aircraft was enhanced by the image processing laboratory at NASA's Langley Research Center (ref. 33). Examples of the enhanced photos from the same data as figure 27(a) are shown in figure 27(b). In the enhanced photos, the core and flow direction can be readily identified.

2.3.2.3 Corvus oil vaporizers

Corvus oil smoke generators were used by Ames-Dryden in the early 1970s to mark the wake vortices of commercial transports (refs. 34–37). The smoker generators were used from near ground level (2300 ft; 700 m) up to altitudes of 12,500 ft (3800 m). A photograph of the unit mounted on the wing of a B-747 is shown in figure 28.

These units are commercially available in two configurations from Frank Sanders Aircraft, Chino, California. One unit is designed to replace a sidewinder missile while the other fits on the 14-in. NATO bomb rack. The system contains 3.8 L (1.0 gal) of fuel (gasoline) and 38–45 L (10–12 gal) of corvus oil and provides 10–12 min of smoke. Fuel is pumped to a nozzle in a combustor where it is mixed with regulated ram air. A capacitor discharge ignition source charges a coil to fire a spark plug, ignite the fuel–air mixture, and provide heat to vaporize the corvus oil. Oil is pumped to a baffled mixing chamber where it is vaporized and discharged as smoke at the tail of the unit. The system can be cycled on and off. White smoke is obtained using corvus oil, while colored smoke can be obtained by using diesel fuel mixed with dye. The white smoke is the most brilliant. Since the unit burns gasoline and ram air to generate heat, the performance of the system is affected by altitude.

The sidewinder missile configuration is 3.33 m (131 in.) long, 0.184 m (7.25 in.) in diameter, and weighs about 991 kg (200 lb) when full. The bomb rack configuration is 2.04 m (80.5 in.) long, has a diameter of 0.25 m (10 in.), and weighs about 54 kg (120 lb) when full. The system requires 12.5 A of 28 V dc electrical power.

An example of the system in operation on the B-747 is shown in figure 29. Kurkowski and others (ref. 36) note that with the smokers operating at peak performance, the vortex marking smoke could be discerned at approximately 5.6–7.4 km (3–4 n. mi.) behind a B-727 in the landing configuration, allowing intentional vortex penetrations by various probe aircraft.

3. Flow Visualization Recording and Time Correlation

Equally as important to flow visualization as making the flow visible or observing changes in flow indicators is the recording of results. Good photographic or video coverage of the flow visualization having time correlation with the known test conditions is necessary to properly document the results. For example, four video cameras and one 35 mm camera were mounted on the F-18 HARV (fig. 22) to document the visualization of the LEX and forebody vortices. Two of the video signals are telemetered to a ground station for recording and display in real time to a control room. The other two video signals are recorded onboard. The 35 mm camera can be operated by either the pilot or an observer in the ground station to record the results during a test point. A telemetered event mark signal is recorded for each 35 mm photo for time correlation.

Onboard documentation is usually the most effective if the proper viewing angle is available. Chase photos usually do not provide such close time correlation, are difficult to obtain during rapid maneuvers such as wind-up turns, and usually provide poor viewing angles from inside the chase plane canopy.

4. DISCUSSION OF TECHNIQUES

The various in-flight flow visualization techniques are summarized in table 1. The systems are categorized by use, hardware, number of test points per flight and description, and ground or in-flight documentation. The applicable references for each method are also listed.

4.1 Boundary Layer Transition and Shock Visualization Techniques

Four methods of detecting boundary layer transition from laminar to turbulent flow are given. Two of the methods (oil flows and liquid crystals) can also be used to visualize shocks. Except for the sublimating chemical technique, all require in-flight documentation. They are all relatively simple except for IR imaging which requires extensive hardware, including a liquid nitrogen-cooled imager and display. IR imaging is the only nonintrusive method, however. Most of the methods are dependent on the differential temperatures of the laminar and turbulent boundary layers and work best when the surface is insulated from heat sinks, such as with a fiberglass glove. Oil flows are limited to altitudes of approximately 7600 m (25,000 ft), liquid crystals to about 10,700 m (35,000 ft), IR imaging to about 5500 m (18,000 ft), while sublimating chemicals have been used as high as 16,800 m (55,000 ft). The IR imaging limit is for an unpressurized instrumentation compartment, but it could be used at higher altitudes with pressurization and special IR-compatible viewing ports.

Liquid crystals are currently the most widely used method of detecting the laminar to turbulent transition location through flow visualization. These, in general, provide favorable results and are capable of continuously and quickly reacting to flow changes, allowing multiple test points to be obtained per flight. However, several precautions need to be taken when using them. Insects and dust in greater amounts tend to adhere to glove surfaces coated with liquid crystals than on uncoated surfaces, potentially causing premature transition and contaminating test results. Uneven thickness in the liquid crystal coat can cause changes of color of the liquid crystal pattern, interfering with interpretation of the patterns. Also, at high unit Reynolds number, the presence of the liquid crystal coat can move the transition location forward.

4.2 Flow Direction and Boundary Layer Separation Visualization Techniques

Two methods of detecting surface flow direction and boundary layer separation are listed. The flow cone–tuft technique is much simpler than the emitted fluid method and provides multiple test points, but does require in-flight documentation. The emitted fluid technique is much more complicated, requires stabilized test points of 1–2 min, but can produce good results when in-flight documentation is not available or practical.

For the surface flow visualization methods using oils, liquid crystals, sublimating chemicals, and fluid, the surface needs to be cleaned of all dirt, grease, or oil prior to application. Gaps and joints in the aircraft skin should be sealed or taped. Many of these techniques use solvents and toxic chemicals, so proper clothing and adequate ventilation should be used.

4.3 Vortical Flow Visualization Techniques

Natural condensation flow visualization can be useful, especially in leading researchers to areas of interest, particularly with respect to vortical flows. However, it is difficult to predict when it might occur, especially for flight planning. It is usually advantageous, especially with the study of vortical flows.

Three different types of smoke generator systems are noted for visualizing vortical flows in the absence of natural condensation. All three systems require fairly extensive hardware installations. The cartridge system and the corvus oil vaporizer supply the largest quantities of smoke for relatively short periods, while the glycol vaporizer can provide smoke for up to 1 hr. Both the cartridge system and the glycol vaporizer can be mounted internally and ducted to the exit port. All three methods require that certain precautions be taken to make them safe for flight research.

5. CONCLUDING REMARKS

In-flight flow visualization has been shown to be useful in the understanding of fluid mechanics. It provides a global view of the airflow and is sometimes the only way the data can be obtained. The results of using ten different methods of flow visualization for flight research can be summarized as follows: oil flows, liquid crystals, and sublimating chemicals are useful flow visualization techniques for detecting boundary layer transition from laminar to turbulent flow. Infrared imaging is a new unintrusive technique for detecting transition but requires extensive hardware. Only sublimating chemicals do not require in-flight documentation, however, each flight yields data for only one test condition.

Flow direction and separation can be determined using flow cones, tufts, and the emitted fluid flow visualization techniques. Flow cones and tufts can provide data during dynamic maneuvering, thereby requiring in-flight documentation. Stabilized test conditions from 1–2 min and extensive hardware are needed for the emitted fluid technique, and only one test condition per flight can be documented.

For visualizing vortex flows, three smoke generator systems are available but all require fairly extensive hardware installations.

Natural condensation is essentially free and can be useful for the flow visualization of vortices and shocks and for leading researchers to areas of interest.

The knowledge of the test flight conditions is nearly as important as the flow visualization itself irrespective of the technique used.

REFERENCES

1. Bracht, K., and W. Merzkirch, "Schlieren Visualization of the Laminar to-Turbulent Transition in a Shock-Tube Boundary Layer," AIAA A79-12898, Oct. 1977.

2. Loving, Donald L., and S. Katzoff, *The Fluorescent-Oil Film Method and Other Techniques for Boundary-Layer Flow Visualization*, NASA Memo 3-17-59L, Langley Research Center, 1959.

3. Crowder, James P., "Add Fluorescent Minitufts to the Aerodynamicist's Bag of Tricks," *Astronaut. & Aeronaut.*, Nov. 1980.

4. Allen, H. Julian, and Edward W. Perkins, *A Study of Effects of Viscosity on Flow Over Slender Inclined Bodies of Revolution*, NACA Report 1048, 1951.

5. Erickson, G.E., L.W. Rogers, J.A. Schreiner, and D.G. Lee, "Subsonic and Transonic Vortex Aerodynamics of a Generic Chine Forebody-Slender Wing Fighter," AIAA 88-2596, June 1988.

6. Nelson, Robert C., "The Role of Flow Visualization in the Study of High-Angle-of-Attack Aerodynamics," in *Tactical Missile Aerodynamics*, vol. 104, M.J. Hemsch and J.N. Nielson, eds., 1986, pp. 43–88.

7. Erickson, Gary E., David J. Peake, John Del Frate, Andrew M. Skow, and Gerald N. Malcolm, "Water Facilities in Retrospect and Prospect—An Illuminating Tool for Vehicle Design," AGARD CP-413, 1987, pp. 1-1–1-28.

8. Werlé, H. "Possibilites D'Essai Offertes Par Les Tunnels Hydrodynamiques a Visualisation de L'Onera Dans Les Domaines Aeronautique et Naval" ("The Onera Water Tunnels Test Possibilities for Flow Visualization in Aeronautical and Naval Domains"), AGARD CP-413, 1987, pp. 14-1–14-16.

9. Merzkirch, Wolfgang, *Flow Visualization*, 2nd ed., Academic Press, Inc., 1987.

10. Johnson, Harold I., and Robert G. Mungall, *A Preliminary Flight Investigation of an Oil-Flow Technique for Air-Flow Visualization*, NACA RM L54G14a, 1954.

11. Bisgood, P. L., The Application of a Surface Flow-Visualization Technique in Flight, A.R.C. R&M 3769, 1974.

12. Meyer, Robert R., Jr., and Lisa A. Jennett, *In-Flight Surface Oil-Flow Photographs With Comparisons to Pressure Distribution and Boundary-Layer Data*, NASA TP-2395, 1985.

13. Curry, Robert E., Robert R. Meyer, Jr., and Maureen O'Connor, *The Use of Oil for In-Flight Flow Visualization*, NASA TM-84915, 1983 (rev. 1984).

14. Holmes, Bruce J., Peter D. Gall, Cynthia C. Croom, Gregory S. Manuel, and Warren C. Kelliher, *A New Method for Laminar Boundary Layer Transition Visualization in Flight—Color Changes in Liquid Crystal Coatings*, NASA TM-87666, 1986.

15. Holmes, Bruce J., and Clifford J. Obara, "Advances in Flow Visualization Using Liquid-Crystal Coatings," SAE Paper 871017, 1987.

16. Anderson, Bianca Trujillo, Robert R. Meyer, Jr., and Harry R. Chiles, "Techniques Used in the F-14 Variable-Sweep Transition Flight Experiment," AIAA 88-2110, May 1988.

17. Körner, H., K.H. Horstmann, H. Köster, A. Quast, and G. Redeker, "Laminarization of Transport Aircraft Wings—A German View," AIAA 87-0085, Jan. 1987.

18. Brandon, J.M., G.S. Manuel, R.E. Wright, and B.J. Holmes, "In-Flight Flow Visualization Using Infrared Imaging," AIAA 88-2111, May 1988.

19. Banner, Richard D., John G. McTigue, and Gilbert Petty, Jr., *Boundary-Layer-Transition Measurements in Full-Scale Flight*, NACA RM-H58E28, 1958.

20. McTigue, John G., John D. Overton, and Gilbert Petty, Jr., *Two Techniques for Detecting Boundary-Layer Transition in Flight at Supersonic Speeds and at Altitudes Above 20,000 Feet*, NASA TN D-18, 1959.

21. Obara, Clifford J., "Sublimating Chemical Technique for Boundary-Layer Flow Visualization in Flight Testing," *J. Aircraft*, vol. 25, no. 6, June 1988. Also available as AIAA 85-2413 and NASA CP-2413, pp. 129–140, Apr. 1985.

22. Montoya, Lawrence C., Paul F. Bikle, and Richard D. Banner, "Section Drag Coefficients From Pressure Probe Traverses of a Wing Wake at Low Speeds," NASA CP-2045, part 2, pp. 601–621, 1978.

23. Crowder, James P., and Paul E. Robertson, "Flow Cones for Airplane Flight Test Flow Visualization," in *Flow Visualization III*, W. J. Yang, ed., Hemisphere Publ. Corp., 1985, pp. 60–64.

24. Crowder, James P., "Flow Visualization Techniques Applied to Full-Scale Vehicles," AIAA 87-2421, 1987.

25. Belevtsov, N., R.E. Brumby, and J.P. Hughes, "In-Flight Flow Visualization, a Fluid Approach," Society of Flight Test Engineers, 14th Annual Symposium Proceedings, Aug. 1983, pp. 4.2-1–4.2-7.

26. Hughes, J.P., R.E. Brumby, and N. Belevtsov, "Flow Visualization From the Ground Up," AIAA 83-2691, Nov. 1983.

27. Fisher, David F., David M. Richwine, and Daniel W. Banks, *Surface Flow Visualization of Separated Flow on the Forebody of an F-18 Aircraft and Wind-Tunnel Model*, NASA TM-100436, 1988.

28. Peake, D.J, D.F. Fisher, and D.S. McRae, "Flight, Wind Tunnel, and Numerical Experiments With a Slender Cone at Incidence," *AIAA Journ.*, vol. 20, no. 10, Oct. 1982, pp. 1338–1345.

29. Campbell, James F., Joseph R. Chambers, and Christopher L. Rumsey, "Observation of Airplane Flow Fields by Natural Condensation Effects," AIAA 88-0191, Jan. 1988.

30. Fennell, L.J., *Vortex Breakdown—Some Observations in Flight on the HP115 Aircraft*, Reports and Memoranda No. 3805, Sept. 1971 (replaces R.A.E. Technical Report 71177, A.R.C. 34 400).

31. Curry, Robert E., and David M. Richwine, "An Airborne System for Vortex Flow Visualization on the F-18 High-Alpha Research Vehicle," AIAA 88-4671, Sept. 1988.

32. Lamar, John E., Philip W. Brown, Robert A. Bruce, Joseph D. Pride, Jr., Ronald H. Smith, and Thomas D. Johnson, Jr., *Operational Performance of Vapor-Screen Systems for In-Flight Visualization of Leading-Edge Vortices on the F-106B Aircraft*, NASA TM-4004, 1987.

33. Lamar, John E., and Thomas D. Johnson, Jr., *Sensitivity of F-106B Leading-Edge-Vortex Images to Flight and Vapor-Screen Parameters*, NASA TP-2818, 1988.

34. Tymczyszyn, Joseph J., and Marvin R. Barber, "Recent Wake Turbulence Flight Test Programs," 1974 Report to the Aerospace Profession—Eighteenth Symposium Proceedings, Soc. Experimental Test Pilots, vol. 12, no. 2, Sept. 1974, pp. 52–68.

35. Jacobsen, Robert A., and M.R. Barber, "Flight-Test Techniques for Wake-Vortex Minimization Studies," NASA SP-409, 1976, pp. 193–219.

36. Kurkowski, R.L., M.R. Barber, and L.J. Garodz, *Characteristics of Wake Vortex Generated by a Boeing 727 Jet Transport During Two-Segment and Normal ILS Approach Flight Paths*, NASA TN D-8222, April 1976.

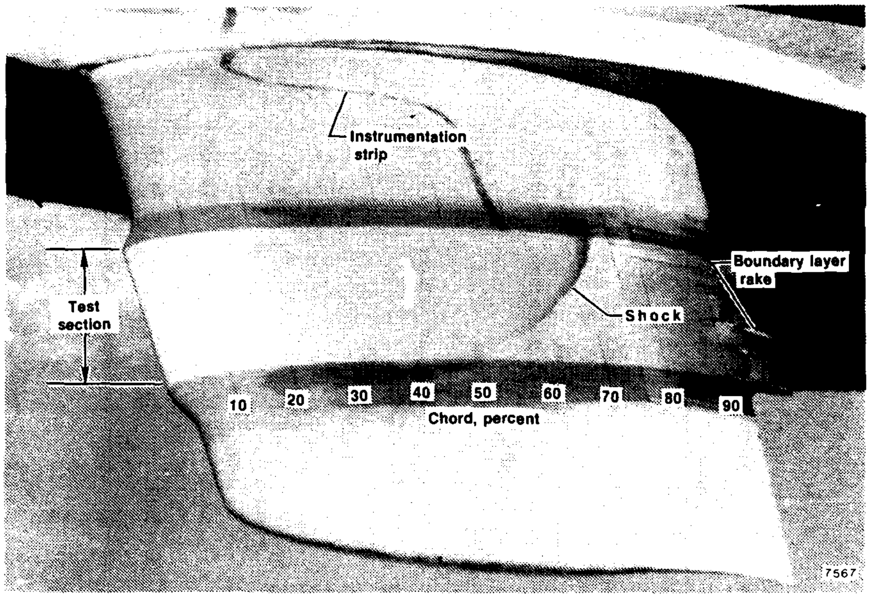
37. Smith, Harriet J., *A Flight Test Investigation of the Rolling Moments Induced on a T-37B Airplane in the Wake of a B-747 Airplane*, NASA TM X-56031, 1975.

6. ACKNOWLEDGMENTS

The authors wish to thank Dr. John Lamar and Jay Brandon of NASA Langley Research Center for material on glycol vaporizers and infrared imaging, respectively, provided for this paper. Thanks are also due to Lisa Jennett Bjarke, Edwin Saltzman, Robert Curry, Bianca Trujillo Anderson, David Richwine, and John Del Frate of NASA Ames-Dryden for their assistance and contributions to this paper.

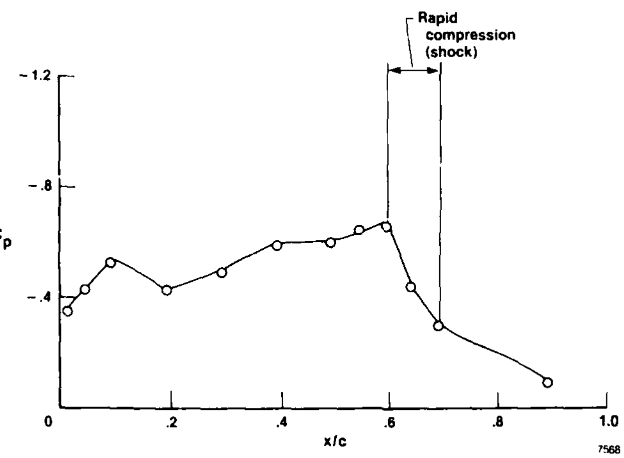
TABLE 1. SUMMARY OF FLOW VISUALIZATION TECHNIQUES

Technique	Primary use	Hardware	Test points–flight	Test point description	Documentation	References
Oil flows	Transition location, shock location	Simple	Multiple	Stabilized	In flight	10–13
Liquid crystals	Transition location, shock location	Simple	Multiple	Stabilized	In flight	14–16
Infrared imaging	Transition location	Extensive	Multiple	Stabilized	In flight	17–18
Sublimating chemicals	Transition location	Simple	Single	Stabilized up to 4 min	Postflight, in flight	19–21
Flow cones–tufts	Flow direction, separation	Simple	Multiple	Dynamic	In flight	22–24
Emitted fluid	Flow direction, separation	Extensive	Single	Stabilized 1–2 min	Postflight, in flight	24–27
Natural condensation	Vortical flow definition	None	Multiple	Dynamic	In flight	29
Smoke Systems						
Cartridge	Vortical flow definition	Extensive	Multiple	Dynamic, ~ 30 sec	In flight	30–31
Glycol vaporizers	Vortical flow definition	Extensive	Multiple	Dynamic, up to 1 hr	In flight	32–33
Corvus oil vaporizers	Vortical flow definition	Moderate	Multiple	Dynamic, up to 12 min	In flight	34–37



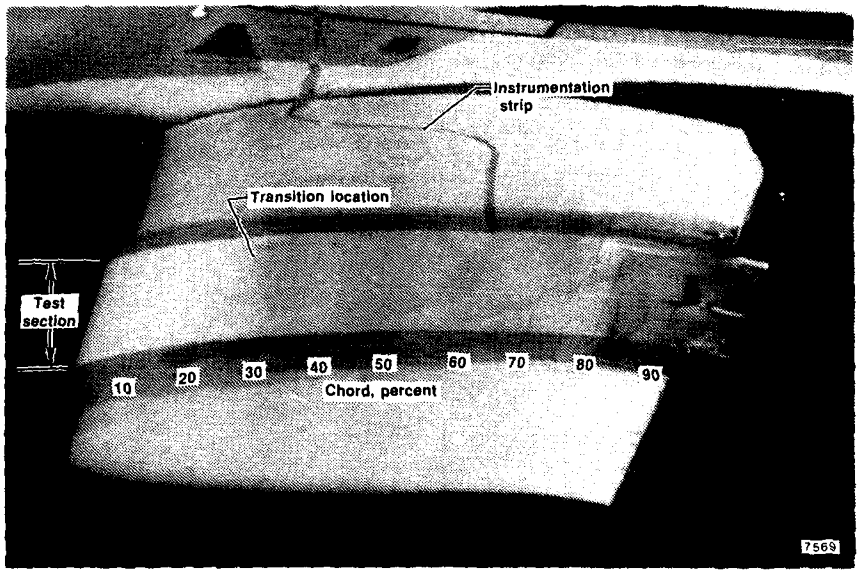
ECN 13418

(a) Glove test section with oil.



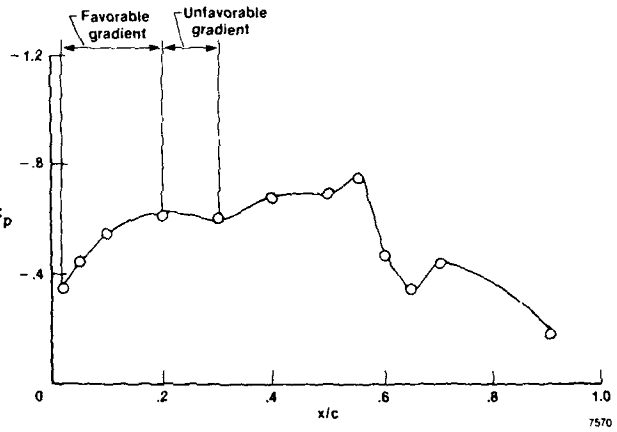
(b) Pressure distribution for glove test section with oil.

Figure 1. F-111 TACT glove test section upper surface oil flow photograph and pressure distribution, $M = 0.85$, $\alpha = 4.7°$, and $\Lambda = 25°$.



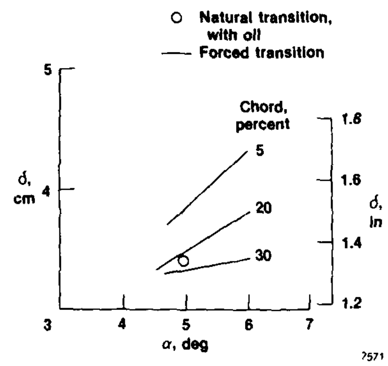
ECN13419

(a) Glove test section with oil.



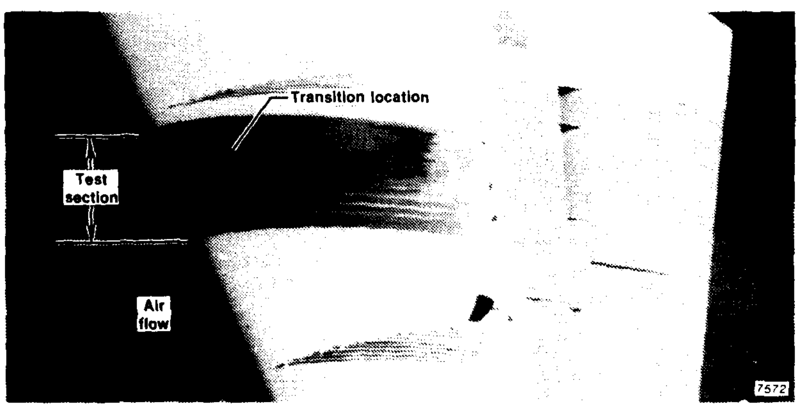
(b) Pressure distribution for glove test section with oil.

Figure 2. F-111 TACT test section upper surface oil flow photograph, pressure distribution, and boundary layer thickness, $M = 0.83$, $\alpha = 4.9°$, and $\Lambda = 16°$.



(c) Determination of transition location using forced transition locations.

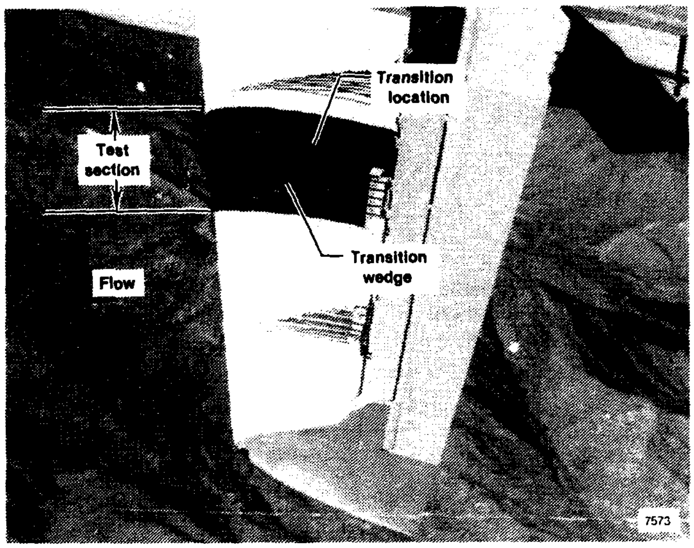
Figure 2. Concluded.



EC 86-33600-018

(a) Sawtooth pattern near wing leading edge.

Figure 3. Examples of liquid crystal patterns.



EC 86-33598-016

(b) Uniform line pattern near midchord.

Figure 3. Concluded.

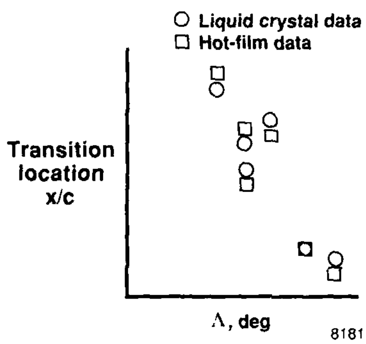
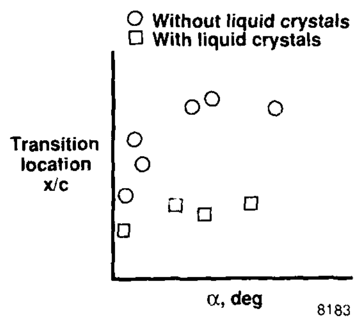
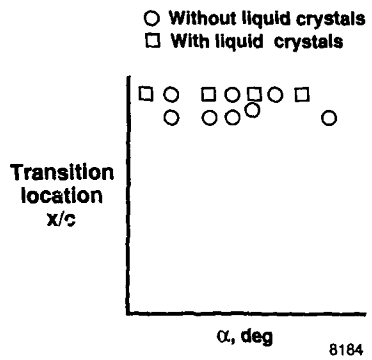


Figure 4. Comparison of transition locations determined by hot-film and liquid crystal techniques.



(a) Low-altitude data

Figure 5. Comparison of transition locations determined by hot-film technique with and without liquid crystals on glove surface.



(b) High-altitude data.

Figure 5. Concluded.

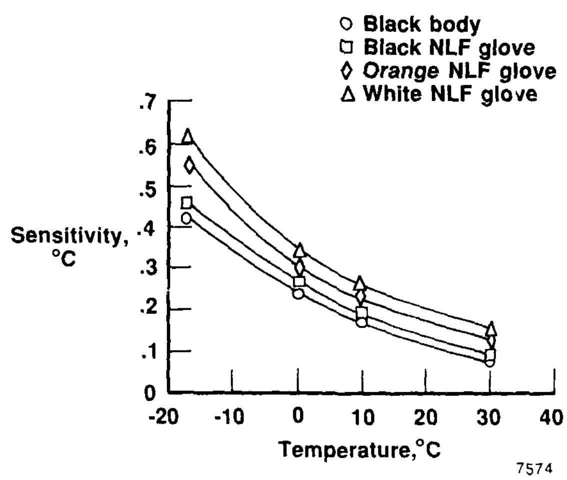


Figure 6. Effect of target temperature on minimum detectable temperature gradient of IR imager.

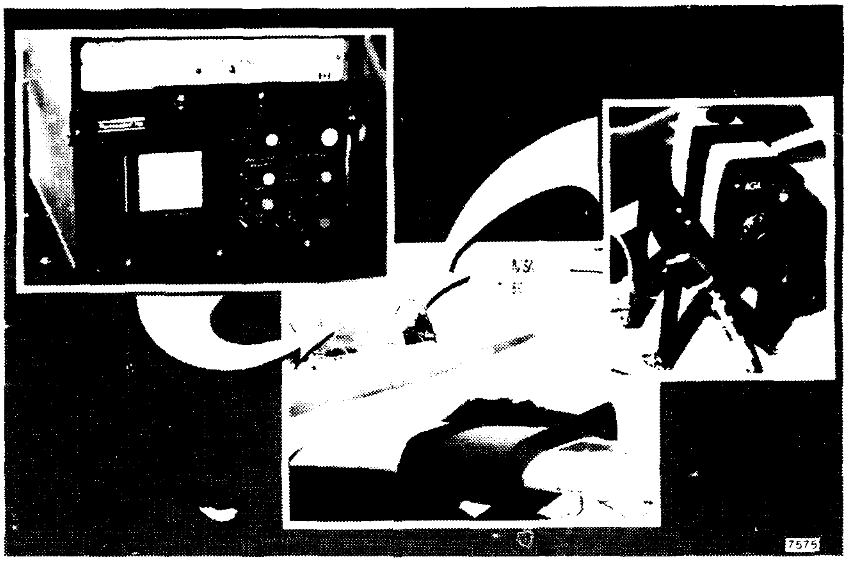


Figure 7. T-34C aircraft and IR imager test setup.

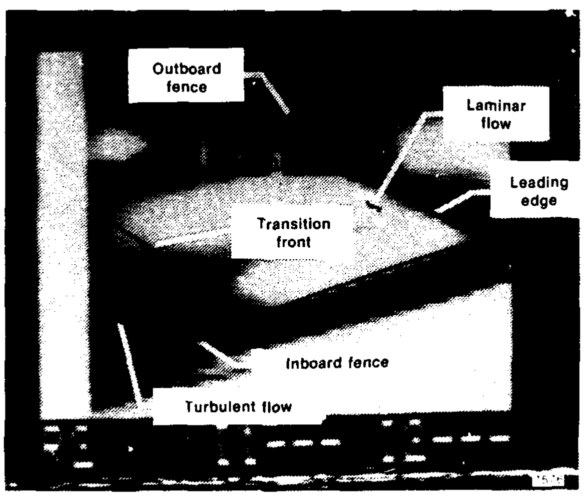


Figure 8. Results of IR imaging from T-34C wing glove during day flight, V_i = 178 knots.

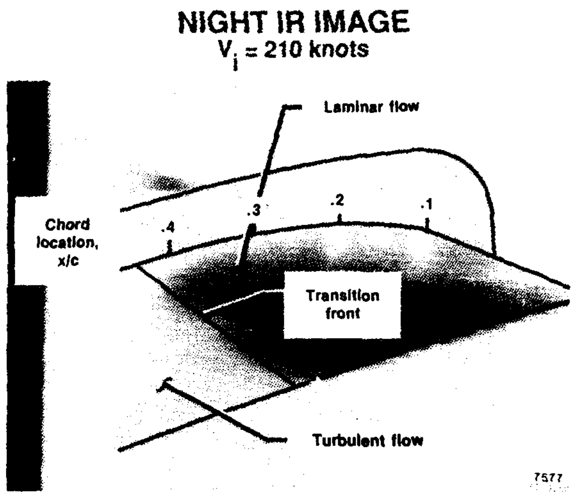
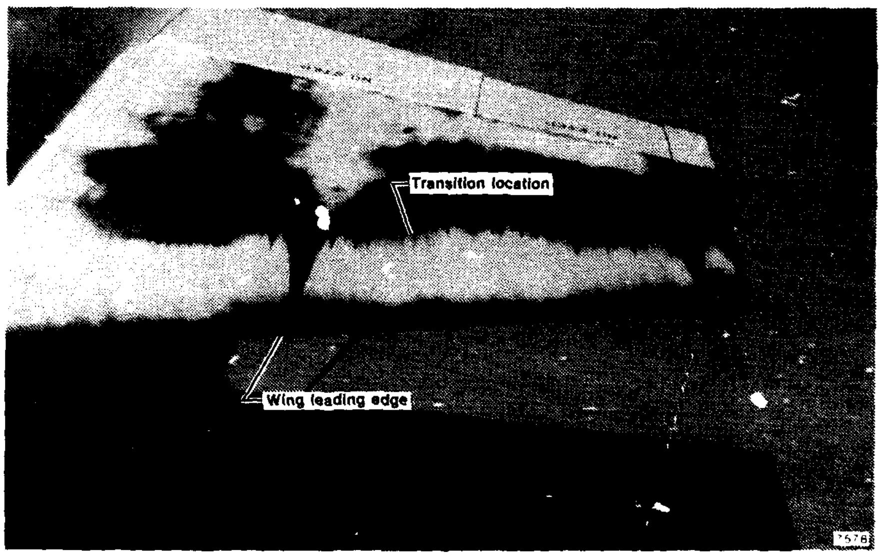
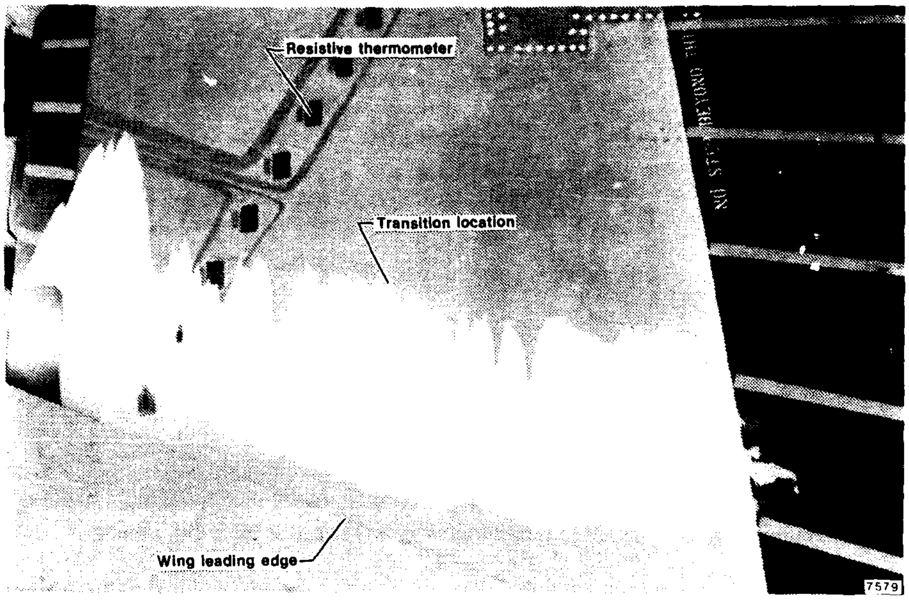


Figure 9. Results of IR imaging from T-34C wing glove during night flight, V_i = 210 knots.



E-3513

Figure 10. Postflight example of sublimating chemical technique on the black painted F-104 left wing, phenanthrene; M_{max} = 1.8, h_p = 16,200 m (53,000 ft).



E-3355

Figure 11. Postflight example of sublimating chemical technique on the fiberglass glove F-104 right wing-tip panel, fluorene; $M_{max} = 2.0$, $h_p = 16{,}800$ m (55,000 ft).

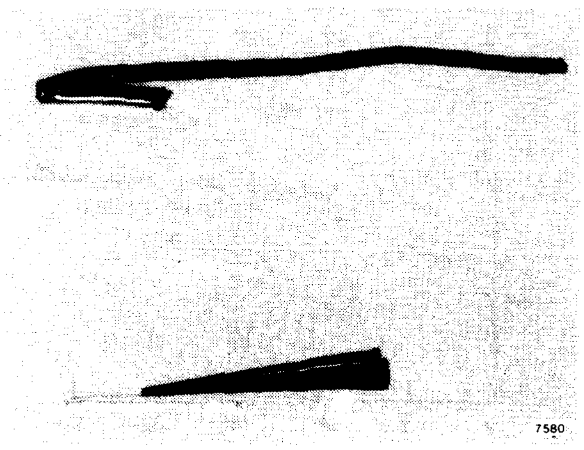


Figure 12. Typical flow cone and tuft.

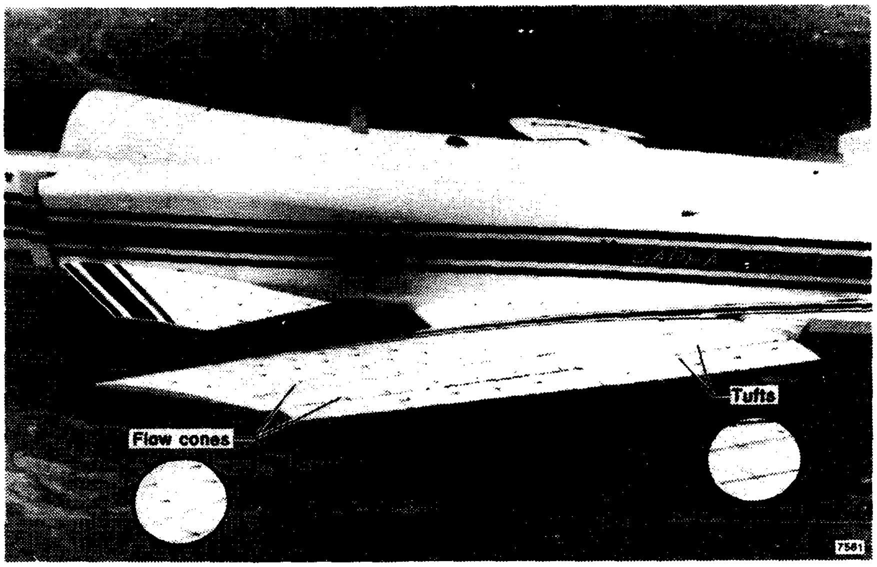
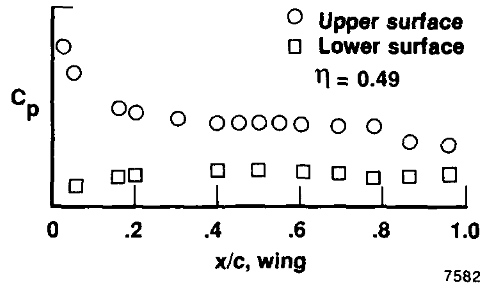
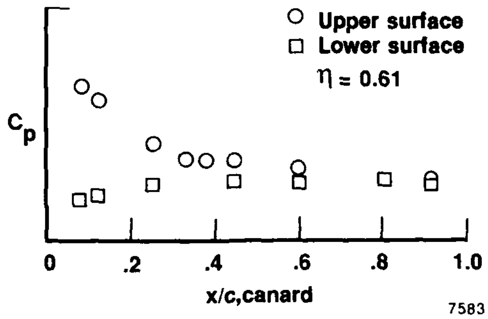


Figure 13. Flow cones and tufts on the left wing and canard of the X-29A aircraft.

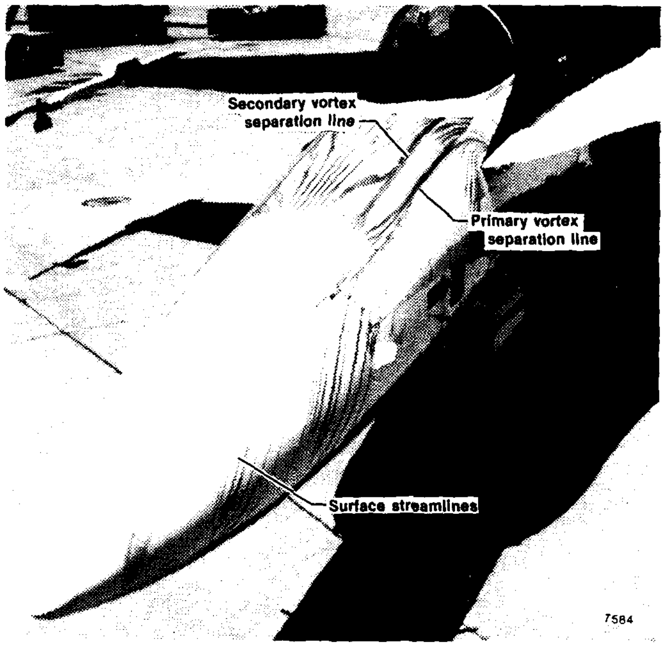


(a) Wing.



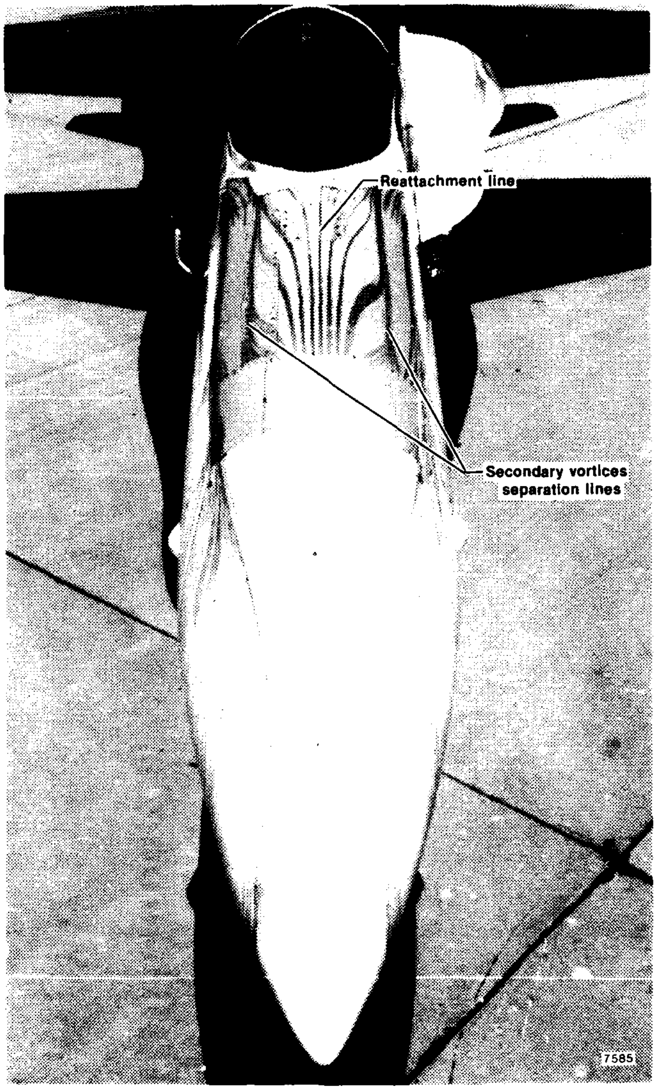
(b) Canard.

Figure 14. X-29A aircraft wing and canard pressure distribution.



(a) 1/4 view of forebody.

Figure 15. Postflight visualization of surface streamlines and lines of separation on F-18 forebody, $\alpha = 30°$.



(b) Head-on view of forebody.

Figure 15. Concluded.

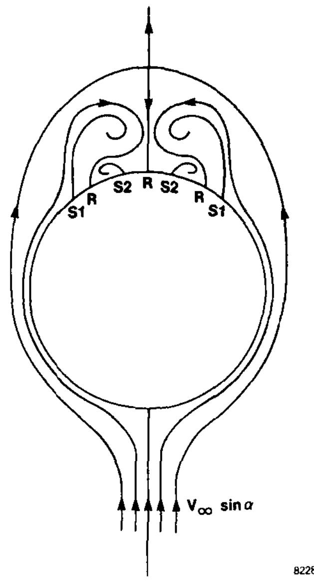
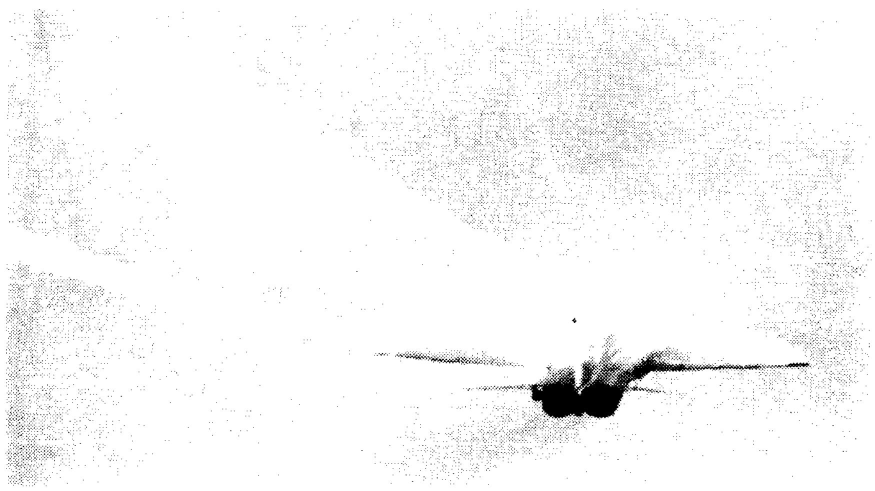
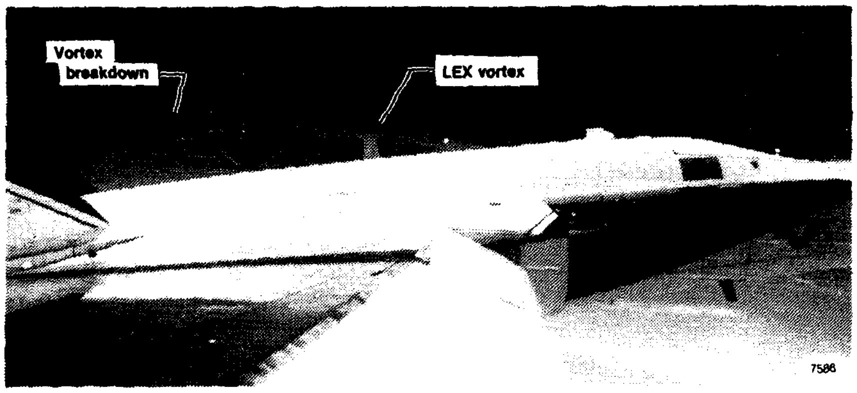


Figure 16. Model of flow about forebody, symmetrical flow (cross-sectional view).



ECN 9514

Figure 17. Wing "gull" pattern and tip vortices on the F-111 TACT aircraft, $M_\infty = 0.82$, $\alpha = 6.9°$, $g = 3.3$.



EC 87 0160-026

Figure 18. Natural condensation of LEX vortices on the F-18 HARV aircraft, α = 19.2°

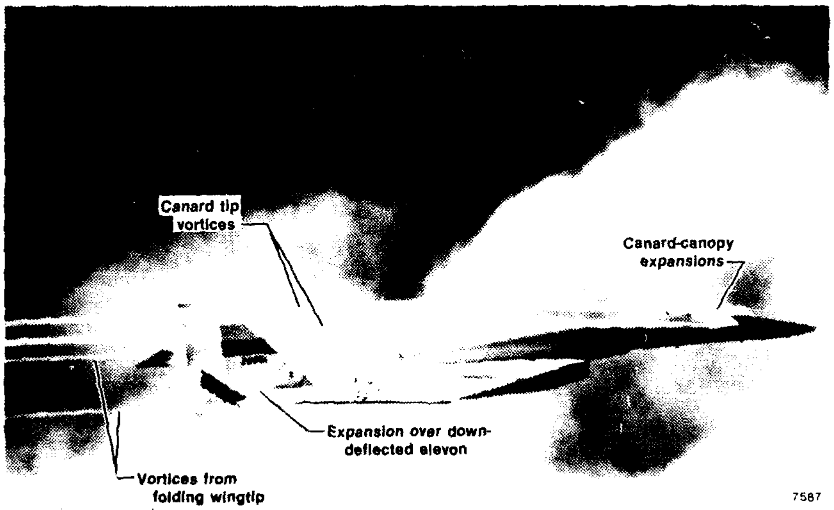


Figure 19. Condensation patterns on the XB-70 aircraft at supersonic speed.

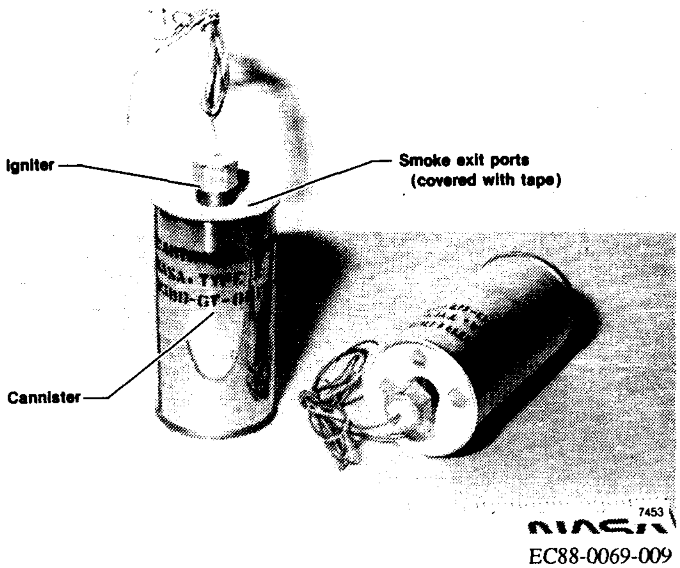


Figure 20. F-18 HARV smoke cartridges.

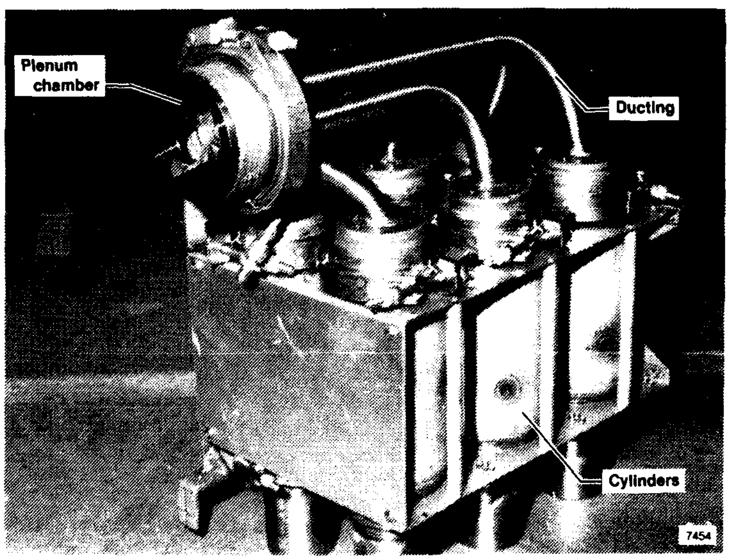
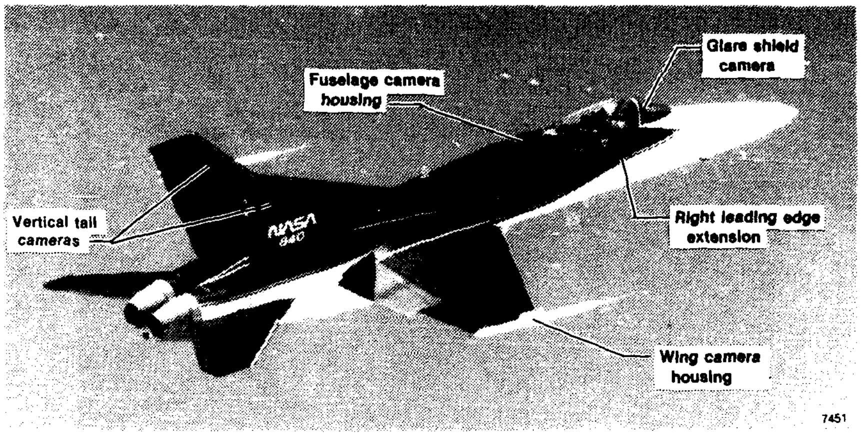


Figure 21. F-18 HARV smoke generator system (heater blanket removed).



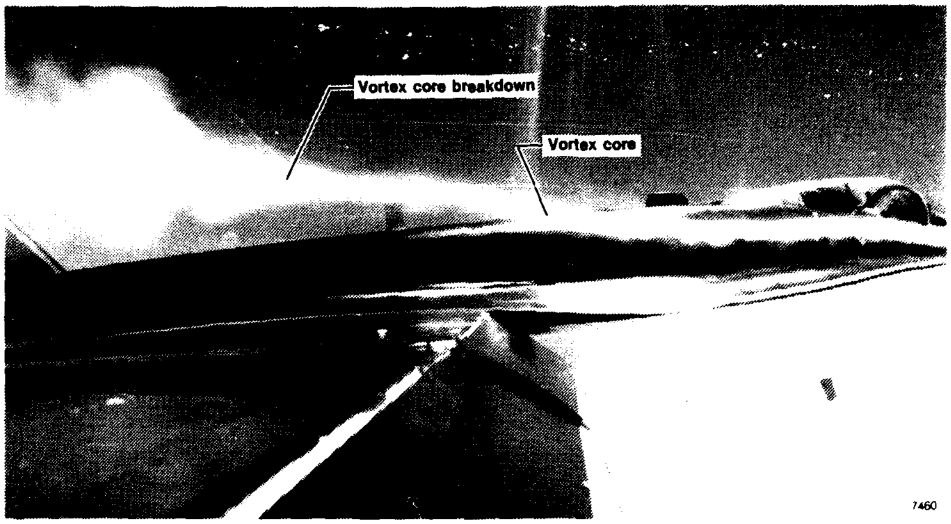
EC88-0095-003

Figure 22. F-18 HARV showing location of cameras.

EC88-0094-010

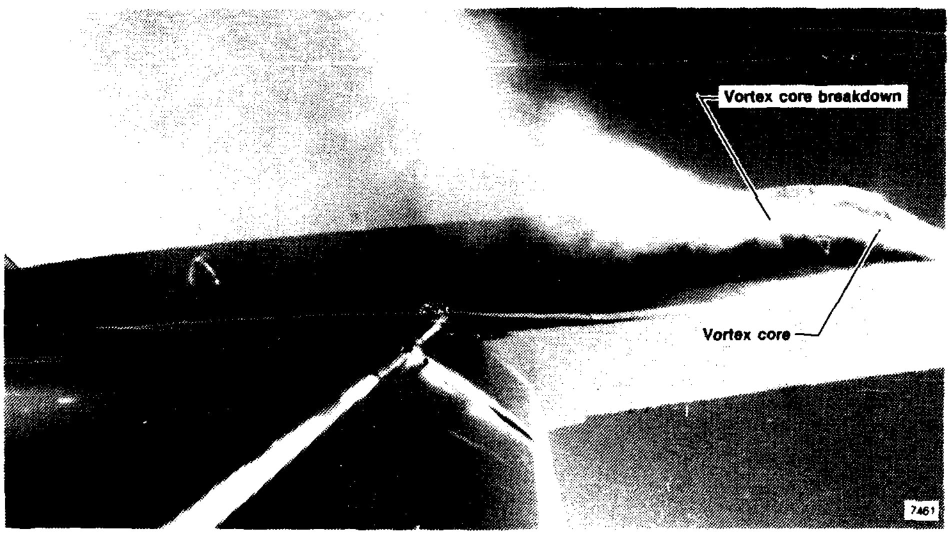
(a) 15.0° angle of attack.

Figure 23. Examples of in-flight flow visualization with smoke generator system on F-18 HARV, view from wingtip camera.



EC88-0094-032

(b) 20.8° angle of attack.



EC88-0094-026

(c) 34.6° angle of attack.

Figure 23. Concluded.

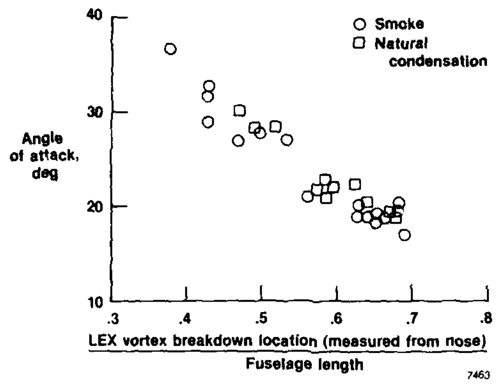


Figure 24. LEX vortex breakdown location variation with angle of attack.

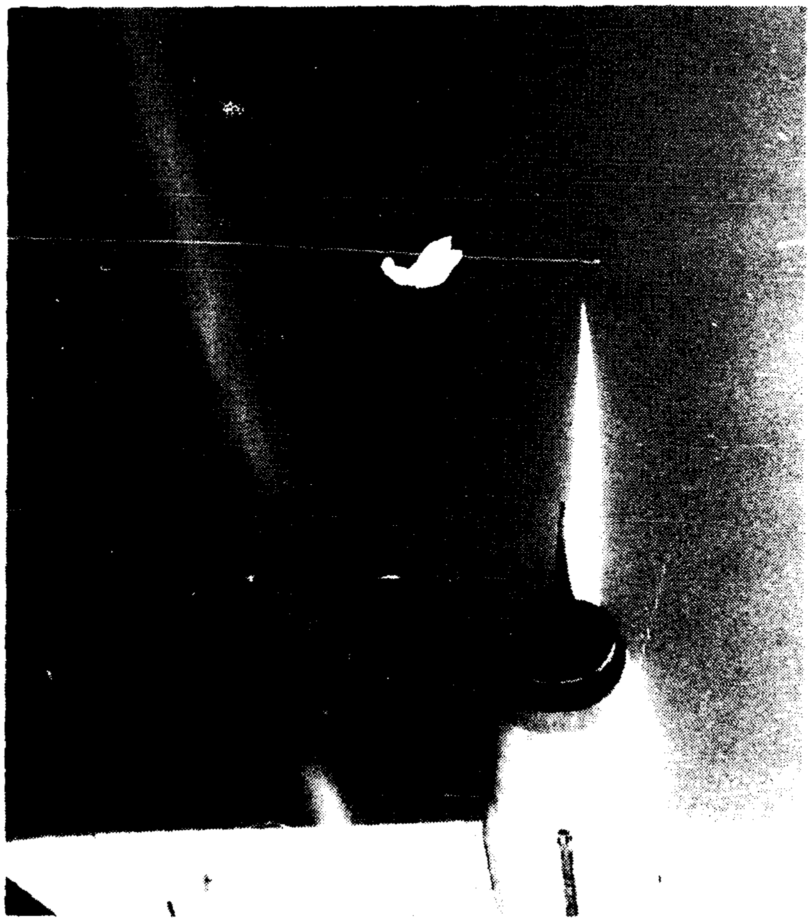


Figure 25. Persistence of p~~ny~~lene glycol smoke from F-106B aircraft.

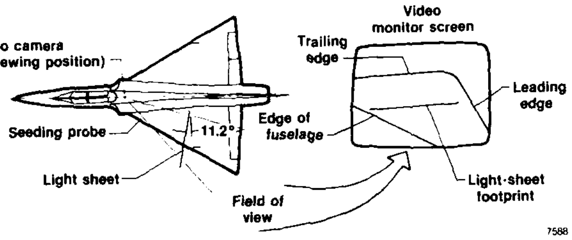
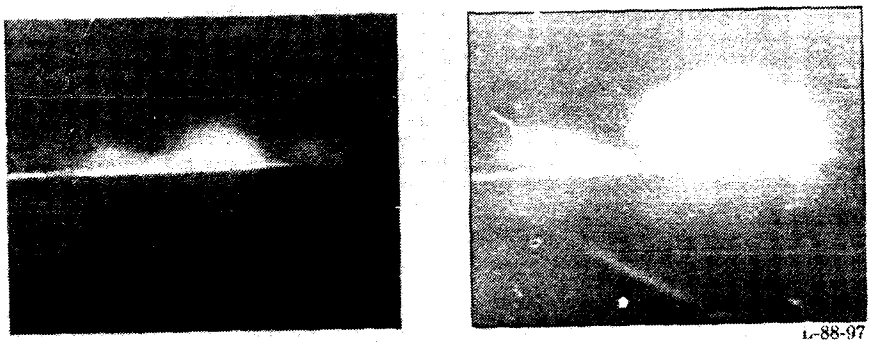
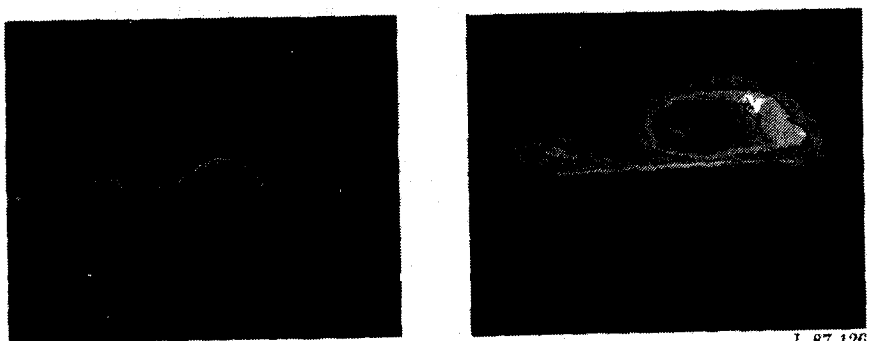


Figure 26. In-flight leading-edge-vortex flow visualization system on F-106B aircraft.



L-88-97

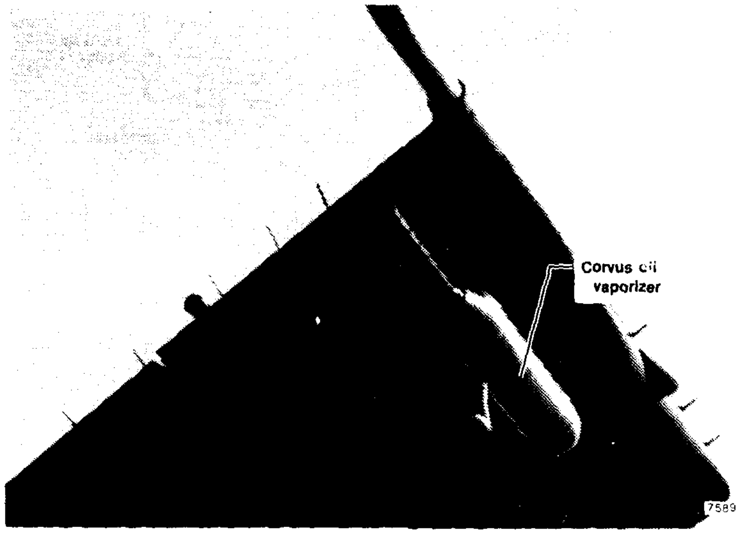
(a) Without digital enhancement.



$\alpha = 17°$ $\alpha = 20°$ L-87-126

(b) With digital enhancement.

Figure 27. Vapor-screen images without and with digital enhancement from F-106B aircraft. $M_\infty = 0.4$, $R_n = 30 \times 10^6$, $h_p = 25{,}000$ ft, 1-g maneuver.



E-27819

Figure 28. Corvus oil vaporizer mounted on B-747 aircraft wing.

ECN 4240

Figure 29. B-747 aircraft wake vortex patterns using corvus oil vaporizers.

AIRCRAFT LIVE FIRE TESTING

by
R.W. Lauzze
and
M.D. Waddell
Survivability Enhancement Branch
Vehicle Subsystems Division
Flight Dynamics Laboratory
Air Force Wright Aeronautical Laboratories
Wright Patterson AFB, Ohio
45433

SUMMARY

Testing military systems in peacetime to determine their ability to successfully operate in a combat environment has been done for years. Some of the tough lessons learned on the battlefield and in the skies during wartime can be examined in the laboratory to help assure that new systems do not incorporate the shortcomings of those of the past. Live Fire Testing (LFT) of aircraft, that is, actual ballistic testing of full scale aircraft/aircraft subsystems, can be performed to evaluate their ability to resist or tolerate battle damage. This testing, combined with analytical tools, can assist the designer in assuring that his aircraft incorporates optimum protection for the system and crew at minimum weight. Events such as a fuel or hydraulic fire in flight, hydraulic ram damage due to a tumbling projectile in a fuel tank, or synergistic effects such as bleed air enhancing a dry bay fire are extremely difficult to analyze. Live fire testing allows evaluation of these and other effects and allows a check of vulnerability analyses that have been performed. It also provides an effective tool for evaluation of Aircraft Battle Damage Repair (ABDR) techniques.

I. INTRODUCTION

Threat forces facing the U.S. military are continually improving the quality of their equipment. This upgraded equipment, plus a numerical advantage, is forcing the U.S. to improve the readiness and sustainability of its combat forces. The deployment of combat survivable aircraft and preparation for efficient battle damage repair are two effective ways of decreasing the cost of maintaining a superior combat ready force. The high cost of today's systems, plus the continuing need to protect aircrews, requires an accurate assessment of aircraft vulnerability be made.

Traditionally, aircraft vulnerability assessments rely on analytical methods. The high cost and destructive nature of ballistic testing to determine aircraft vulnerability have generally constrained such testing to components and small subsystems. These tests can help validate many elements of the analytical methodologies. Analytical techniques, though used extensively, do not always model the complex processes which occur during ballistic impact and rarely capture synergistic or seondary effects. For a truly accurate vulnerability assessment to be made on aircraft, analyses must be supplemented with full scale ballistic testing.

The design of the live fire test depends on the mission of the aircraft, its design features, and the confidence in existing vulnerability estimates. The confidence in existing estimates depends, to a large extent, on prior ballistic test data on similarly configured systems at similar test conditions. A subsystem, deemed to have a high vulnerability, may not require testing if sufficient evidence exists to allow confidence in its analytically predicted vulnerability. Of course, the consequences of that high vulnerability must be recognized and acceptable. If however, there is little confidence in the prediction and/or a method is sought to reduce that vulnerability, full scale ballistic tests may be warranted. The mission of the aircraft will define the threat(s) to be used and the test conditions. To ensure applicability of the data and to maximize the use of expensive test assets, test conditions are chosen which avoid severe threat over or under match.

Live fire tests are designed to examine areas of maximum uncertainty, areas of highest sensitivity to crew and system survivability, and areas of possible synergism and secondary effects, while conserving test and target resources which are generally severely constrained.

The U.S. currently has two major initiatives underway in the live fire test area. The Joint Live Fire Program (JLF), started in 1984, has a primary objective to gather empirical data on the vulnerability of currently fielded U.S. systems to foreign weapons and to provide insight into design changes that may be necessary to reduce their vulnerabilities. The Live Fire Test (LFT) Legislation, effective in 1987, states that all major new weapons systems must be tested for vulnerability of the system in combat by firing munitions likely to be encountered in combat. These tests are required sufficiently early in the development phase of the system to allow any design deficiency demonstrated by the testing to be corrected.

II. APPROACH TO LIVE FIRE TESTING

Aircraft live fire testing can be considered a detailed laboratory examination of a sequence of snapshots of an aircraft in combat at the time of a ballistic impact. The design of the live fire experiment depends on the desired output of this sequence of snapshots. It should be remembered that live fire test is only a tool, and an expensive one, used to arrive at a confident estimate of an aircraft's vulnerability. The scope of the test can and, since it is so expensive, should be custom tailored to complement any other existing analyses and test data to arrive at an acceptable confidence level (Figure 1).

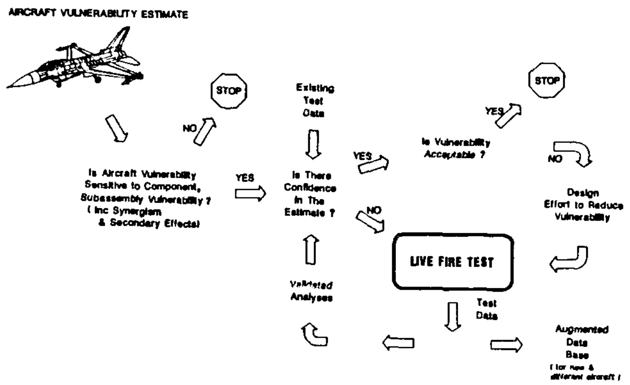


Figure 1. Relationship of Analysis, Prior Test Data, and LFT

11.1 DETERMINATION OF OBJECTIVE

The objective of the particular live fire test must be thoroughly considered prior to any test initiation. Live fire can be used to explore the upper limits of an aircraft's battle damage tolerance, evaluate ballistic response to a typical set of threat projectiles or missiles, or proof test vehicle protection concepts. The objective of the test narrows the choice of a wide variety of possible test conditions. An evaluation of maximum battle damage tolerance requires that highly damaging threats be used and test conditions be tailored to match those critical for the aircraft. To examine critical vulnerability conditions for a fighter wing, for example, the test might require a limit load be applied to the wing, a stoichiometric mixture of fuel and air exist in the wing fuel tank, and testing be conducted with a medium caliber high explosive (HE) projectile. A similar test at "typical" conditions would probably incorporate a lower load with a fuel/air mixture determined by typical aircraft fuel temperature and quantity, using a small HE or armor piercing (AP) projectile. The probability of catastrophic damage is, of course, less likely in the second case. A comparison of the results of typical versus critical conditions can be seen in Figure 2.

Figure 2(a). Catastrophically Damaged Wing

Figure 2(b). Wing with Typical Ballistic Damage

11.2 SELECTION OF TEST THREATS AND SHOTLINES

The mission(s) of the aircraft will define the threats to which it will be exposed and, therefore, evaluated. Aircraft designed for close air support (CAS) will likely be exposed to a different set of threats than a cargo aircraft - although some may be the same. Typical projectile threats are shown in Figure 3. The mission of the aircraft, together with the objective of the test, further narrows the choice of test conditions.

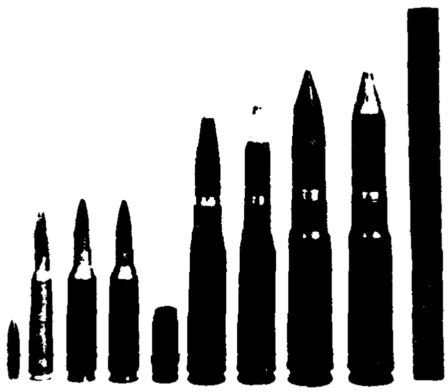


Figure 3. Various Test Projectiles

The selection of shotlines for a live fire test depends, to a large extent, on engineering judgement applied to computer engagement simulation results, combat data, and test hardware availability. At first glance, a selection of random shotlines appears to be an ideal approach; however, having enough hardware to obtain sufficient data on questionable areas may be economically prohibitive. Combat hit data on other similar aircraft is useful, but likely will not be extensive enough to fully define shotlines for a newer aircraft. Care must also be exercised in interpreting combat data. For example, a lack of impact data on a particular area of an aircraft may indicate no hits were made in that area (low suscepibility) or that it is an area of particularly high vulnerability (aircraft was lost). (See Figure 4.)

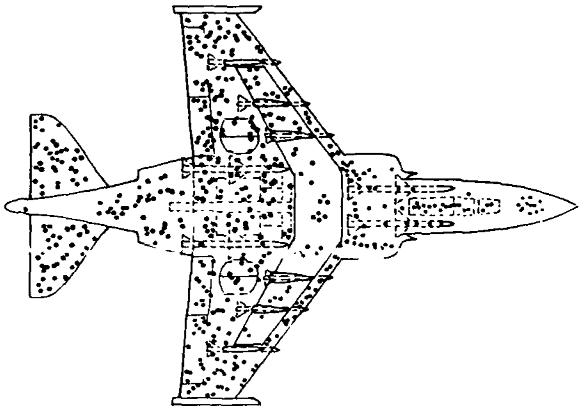


Figure 4. Hit Plot on an Aircraft Showing Typical Impact Locations

Many times there are issues of particular interest (e.g., low confidence in the vulnerability assessment of a particular subsystem) which require test data to resolve. It may be necessary to conduct tests to study cause/effect relationships. For example, determination of hydraulic fluid flammability requires repeated impacts into realistically pressurized lines at operational temperatures, flow rates, etc. The U.S. and U.K. recently cooperated on a hydraulic fluid flammability evaluation. The U.K. tested various fluids at static conditions while the U.S. tested similar fluids incorporating airflow conditions with velocities up to 400 kts. Figure 5 shows the test setup used to evaluate the fluid flammability under airflow conditions.

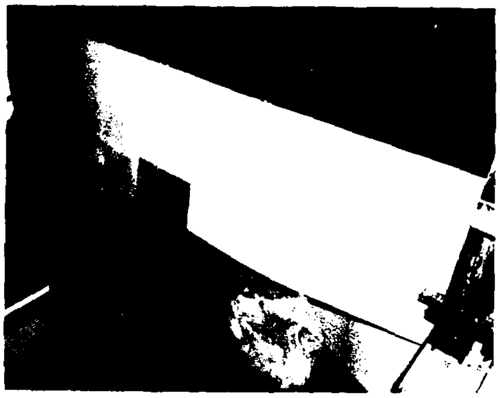


Figure 5. Setup for Evaluation of Hydraulic Fluid Flammability

11.3 COMPONENTS VS COMPLETE AIRCRAFT TESTING

The subject of component vs. complete full size aircraft testing can generate considerable controversy. The U.S. LFT legislation previously mentioned states that the test for vulnerability is to be performed on the system configured for combat, that is, loaded with all dangerous materials (including flammables and explosives) that would normally be on board in combat. With unlimited resources, in order to comply with the legislation, one could design a series of tests to perform on several complete aircraft assuming, of course, that the test hardware would be available. A realistic approach to LFT requires, however, that a method be adopted which conserves funds and test articles. Often, with aircraft costing many millions of dollars, obtaining sufficient hardware to test is the most difficult part of the process. During the development process, an approach which includes evaluation of critical components and subassemblies (Figure 6) as they become available will help prevent surprises later.

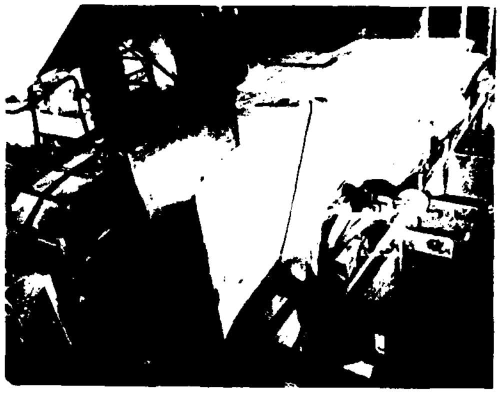


Figure 6. F-15 Wing Assembly Ready for Test

Occassionally, testing a replica of the target before evaluating the actual aircraft component/subassembly is cost effective. A replica test article is generally a full scale duplicate of the aircraft component or subassembly in question. It does not usually include all of the expensive (but usually non-essential from a vulnerability standpoint) features of the aircraft, like chemically milled skin materials. It could be an expendable, simple sheetmetal model or a steel frame with easily replaceable skins and internal parts. Use of a replica may cause some loss in test fidelity, but generally allows collection of a large amount of good data at a relatively low cost. Over ten years ago, a replica of an A-10 wing was built by the Air Force to evaluate fire probability. See Figure 7. Its replaceable wing skin material allows its use today to evaluate ballistic impact effects on composite materials.

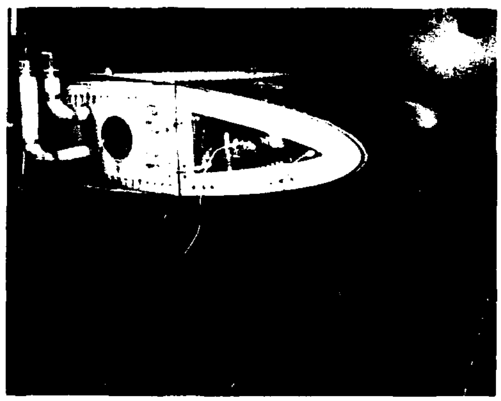


Figure 7. A-10 Wing Replica Used in Vulnerability Tests

The size of the test article depends on the requirement for realism, threat size, and the expected results. Obviously, to understand the effects of a given caliber HE projectile on an aircraft requires a larger target than a similar size AP projectile. When evaluating secondary or synergistic effects, it is necessary to make the target large enough to capture all of the primary damage effects plus any other effects which could also occur.

When possible, test targets may be previously used developmental test hardware such as static or fatigue test articles. Developmental ground or flight test aricraft can usually be modified, for vulnerability evaluation purposes, to match the operational aircraft. Salvaged crash aircraft or otherwise damaged assemblies, production rejects, or surrogates can sometimes be used, depending on similarity to the production article and the severity of any damage (see Figure 8). An ideal live fire program may contain a replica target to collect vulnerability data at several test conditions, a surrogate target to examine effects on installed components such as hydraulic lines, and an actual aircraft assembly to be used for proof testing.

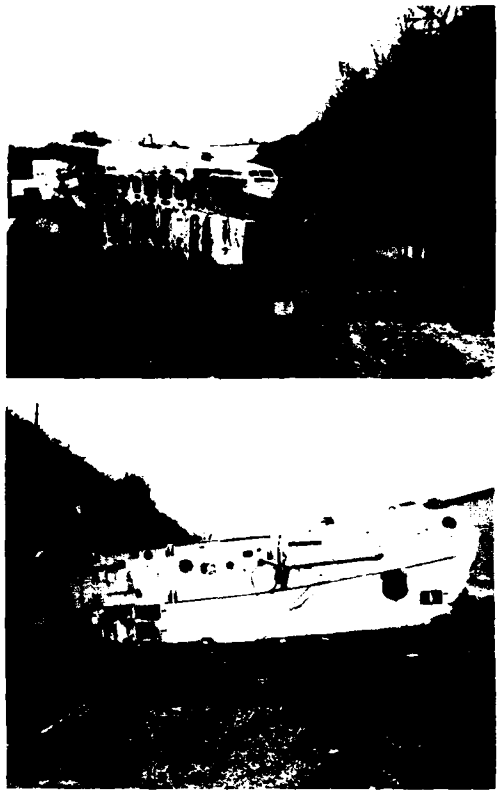


Figure 8. Photos of Salvaged F-15 Test Hardware

11.4 TEST SET-UP

Setting up for aircraft live fire testing requires consideration of the test parameters which are necessary to duplicate for realism. See *Figure 9*. *The environment in which the test is performed can* significantly impact the results and must be modeled carefully. For example, a ballistic impact into a static filled fuel cell will often result in a fire. A repeat of the test with a representative 300 kts. airflow across the impact surface will likely result in no fire.

Some of the more important parameters to consider for a successful simulation are:
(a) Threat - projectile: caliber, type (high explosive, armor piercing, etc.), impact velocity, fuzing, obliquity, etc.; fragmenting warhead: standoff distance, fragment mass, material, density, velocity, shape, etc.
(b) Fuel/Fuel System - amount, type, temperature, pressure, volume, depth, impact direction, airflow velocity, etc.
(c) Flight Controls/Hydraulics - control surface load, tubing material and size, hydraulic fluid type, temperature and pressure, flow rate, accumulator size, impact bay size and clutter, etc.
(d) Propulsion/Fuel Ingestion - engine operating parameters, shielding, fuel ingestion modeling parameters (inlet pressure, temperature, mass flow rate, and fuel tank pressure), etc.
(e) Structure - aerodynamic load, obliquity, fuel amount and depth (hydraulic ram), etc.

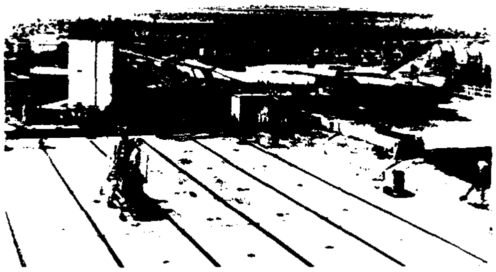


Figure 9. Aircraft Ready for Test at the Naval Weapons Center

Specialized facilities for live fire testing exist in each of the U.S. Armed Services. The Air Force's Aircraft Survivability Research Facility (ASRF) has been used for live fire type testing since the Southeast Asia conflict era and has seen testing on the Air Force A-7, F-4, A-10, and F-111 as well as developmental testing on vulnerability assessment and reduction technology. The ASRF is located at Wright Patterson AFB, Ohio. Similarly the Navy's Weapon Survivability Laboratory (WSL) at the Naval Weapons Center, California has been involved in vulnerability research for over twenty years and has been used for evaluation of such systems as the F-4, A-4, A-7, and F-14. The Army operates aircraft live fire test facilities at Aberdeen Proving Ground, Maryland and Ft. Eustis, Virginia. While these are not all the aircraft vulnerability test facilities available in the U.S., the four mentioned here can be seen in Figure 10.

Figure 10. Four U.S. Vulnerability Test Facilities

III. AIRCRAFT BATTLE DAMAGE REPAIR (ABDR)

Live fire testing provides a unique opportunity to inspect, assess, repair, and evaluate the repair of actual combat damage on aircraft. In the recent past, much of the "battle damage" evaluated and repaired by aircraft techicians was limited to older, sometimes obsolete, aircraft. The "battle damage" was generally inflicted with fire axes, homemade pipe bombs, or other methods which resulted in unrealistic damage. Statically initiated fragment simulator devices have been developed and are in use, but they do not duplicate all the effects, such as hydraulic ram, of a high velocity projectile impact.

In most live fire testing, multiple shots with different shot lines are accomplished on a single operational or developmental aircraft/aircraft component. Repair is generally required after each test so that the test article can be used again. The use of ABDR personnel to make the repairs gives them experience repairing the effects of actual ballistic impacts on up-to-date systems plus returns the test article to a test-ready configuration. The USAF used ABDR teams in its recent Joint Live Fire Program tests of F-15 and F-16 wings (Figure 11). The lessons learned in this exercise are being used to update and refine the technical repair manuals and upgrade the contents of the ABDR tool and material kits.

Figure 11. An Air Force ABDR Team Repairing Ballistic Damage

IV. RECENT TESTING ACCOMPLISHMENTS

As previously mentioned, the Department of Defense sponsored Joint Live Fire (JLF) Program is currently underway in the U.S. The JLF Program on aircraft systems is a joint services program managed by the Joint Technical Coordinating Group on Aircraft Survivability (JTCG/AS) in which the Army, Navy, and Air Force are working together, sharing resources, facilities, and data to evaluate our current combat systems.

An example of recently completed work is a Navy/Air Force cooperative effort to evaluate turbofan engine fuel ingestion tolerance. Fuel ingestion generally results when a projectile which impacts a fuselage fuel tank also penetrates an engine inlet duct wall allowing fuel to leak into the front of the running engine. A schematic of a fuel ingestion event is shown in Figure 12.

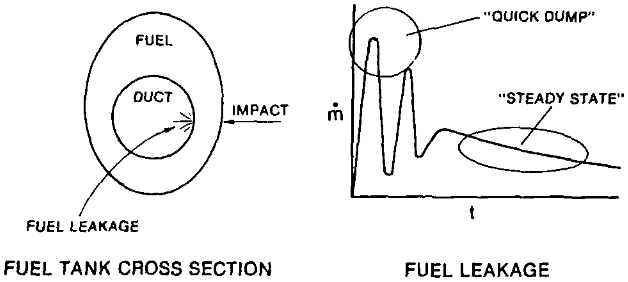


Figure 12. Sketch of a Typical Fuel Ingestion Event

A series of ballistic and simulated duct wall penetrations were conducted to determine the fuel ingestion flow rates which a turbofan engine can tolerate. The test inlet duct was sized to match current aircraft and tests were run with fuel ingested at three locations to simulate different hit locations on an actual aircraft. The test engine was running at typical operating conditions and the inlet duct was pressurized with facility ram air to simulate flight. The test set up is shown in Figure 13. These tests provided valuable data needed on newer engines as well as insight into possibilities of designing more survivable engines in the future.

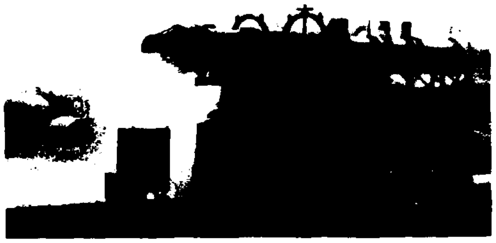


Figure 13. Turbofan Engine Fuel Ingestion Tolerance Test

Other tests conducted under JLF evaluated various aircraft wings against existing threats for structural damage and fire or explosions. Shots were fired at wings which were fueled and structurally loaded, with external airflow applied, simulating conditions expected if the aircraft were in combat. See Figure 14. Some shots were designed to test existing fire and explosion suppression systems by ensuring optimum conditions existed for a fire or explosion (e.g., a large volume of stoichiometric ullage in the fuel tank). Structural tests were accomplished by loading the wings to typical flight loads during impact. After impact, the wings were loaded to higher loads to determine any changes in load carrying capacity. Strain gages and displacement potentiometers were used to evaluate stress distributions and stiffness changes in the wings due to damage.

Figure 14. Live Fire Test of Aircraft Wing

The Army is conducting similar live fire tests to evaluate the vulnerability of helicopter engines, main rotor blades, fuel tanks, external stores, flight control components, and main and tail rotor drive components. On the engine tests, a large percentage of the vulnerability data was collected prior to actually subjecting the engine to ballistic damage. This was done by drilling holes in specific components of the engine and remotely opening the holes while the engine was running under a simulated flight load. The test set up is shown in Figure 15. The degradation in performance was then monitored and recorded under controlled conditions. Actual ballistic tests on the running and loaded engine followed these experiments to verify the prior results and examine secondary and synergistic effects.

Figure 15. Helicopter Engine on Test Stand

V. PAYOFF

The primary purpose for live fire testing is to gain knowledge about the effects of potential threats to weapon systems in order to predict how they will perform in combat. As aircraft become increasingly more complex and their subsystems become more interactive, analytical predictions become progressively less reliable. Much can be accomplished by modelling the ballistic event, however there is no substitute for actual testing. The synergistic effects involving numerous subsystems in close proximity to ballistic impacts on an aircraft frequently present surprises even to veteran test engineers.

Live fire testing can and does result in improvement in the protection of aircraft and aircrews from the effects of combat threats. It also results in the ability to plan with significantly improved confidence. This is reason enough to conduct the tests, but other payoffs like improvements to prediction techniques and the validation and upgrade of existing computer models will eventually serve to decrease the frequency and quantity of tests required to gain confidence in the survivability of new weapon systems.

FLIGHT TESTING A NEWLY-DESIGNED STAND-OFF WEAPON DISPENSER

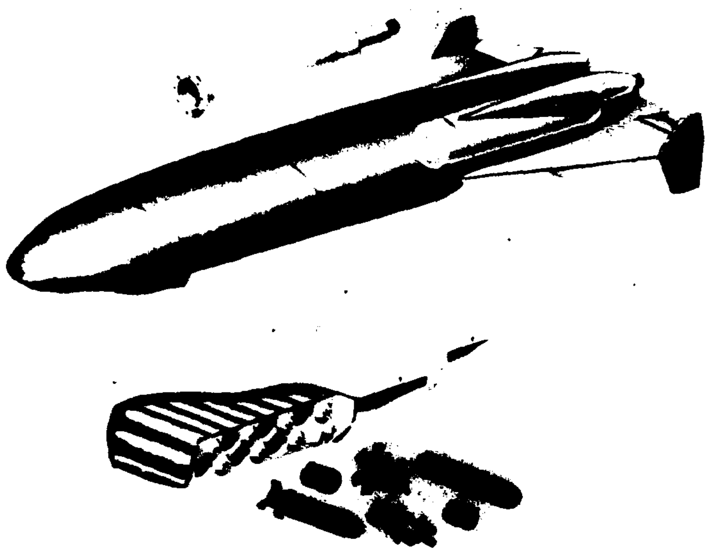
G. Ferretti
Chief Flight Test Department

P. Duranti
Chief Flight Test Engineering

AERITALIA
Combat Aircraft Group
Flight Test Department
I-10072 Caselle Torinese (Italy)

SUMMARY

"Skyshark" is a newly-designed Stand-off Weapon Dispenser recently developed by AERITALIA and SNIA-BPD. These Italian companies formed in 1986 a joint venture named CASMU, which in Italian stands for Multi Usage Stand-off Armament Consortium. Designed specifically for being launched by AM-X and Tornado aircraft, the Skyshark will be easily integrated with most of the modern attack aircraft.
Released at a distance from the target variable with speed/altitude conditions this new weapon, capable of following a pre defined path, will operate independently thanks to a proper inertial navigation system coupled to a pre-programmed computer for stabilization and manoeuvering. Once reached the target, a dedicated armament system will come into action dispensing submunitions that can, as the case requires, be selected amongst various combinations and timed for the attack.
Five of the planned six demonstration launches have been performed so far with satisfactory results, allowing to acquire significant data about aerodynamics, flight-control and armament systems characteristics.
This paper deals with the flight test activities carried out during the "Demonstration" phase, aimed at proving the correctness of the general concept.
The most interesting peculiarities related to test flying such unusual flying machine and the experiences acquired are herewith described.



AERITALIA-SNIA BPD Stand-off Weapon Dispenser "Skyshark"

INTRODUCTION

It is a persistently growing belief, when considering a closely defended target, that a very effective weapon to be used today is a stand-off armament released from the carrier aircraft at a great distance from the target and capable of accurate and autonomous flight against the designated objective. For this reason intensive and wideranging research is presently underway relative to these systems.

The concept of stand-off armament, however, is not a new idea. In fact, the first stand-off weapon in the world, named "Telebomb", was developed in Italy during the first World War by two engineers of the Italian Military Administration, Major Guidoni and Colonel Crocco (fig.1).
That innovative weapon was designed to be launched by airships, thus greatly reducing the vulnerability which had proven to dramatically affect the operational effectiveness of these slow aircraft.
It consisted of a 90 Kilos unpowered winged bomb (1.5 meter wing span,biplane) provided with tail control surfaces. The mission profile was preplanned by means of a cam the action of which mechanically positioned the control surfaces at predefined times by means of air-compressed actuators. After being released from the carrier airship at about 8000 ft the Telebomb performed a steep dive in order to achieve a target speed of approx. 200 Kts and covered a distance of 7 - 8 miles before dropping over the target. Six test launches were performed in 1918 with partial success. It can be easily understood what serious problems were met as far as the navigation accuracy is concerned. Furthermore, at that point in time, it was too late for trying to start the production for actual use in war and the project was then abandoned.However it was considered promising by the U.S. Navy which bought the patent.

After 70 years Aeritalia is now actively involved with SNIA-BPD in a national joint venture, forming a consortium named "CASMU", the prime objective of which is to extend the range of specialized stand-off weapon products.
Our first realization is a stand-off weapon dispenser, named "Skyshark", to be air-launched against fixed targets, conceived for outstanding operation, flexibility and accuracy and capable of autonomous navigation, search, identification and neutralization of the target. The reduction of the aircraft's attrition rate and the high effectiveness of its capability to carry several optimized combinations of "smart" submunitions are its key characteristics.
Special attention has also been paid during the design phase to optimize its stealthness.
Two basic versions,unpowered and powered (by means of rockets) were initially expected to be developed. Both of them, differing in range, foresee to be equipped with an autonomous guidance and control system for low altitude launch and flight,assuring high speed during close-in and dispensing phases.

Because of its dimensions, Skyshark can be carried under either the wings or the fuselage of a wide range of aircraft such as Tornado, AMX, A-4, A-7, F-4, F-15, F-16, F-111, Mirage, Jaguar, etc.

At present, the development phase is successfully coming to an end. Five of the six foreseen development launches have been performed so far, yelding to satisfactory results, and have led to the gradual improvement of the system's performance and reliability.

The subject of this lecture is the flight test activity related to the development phase, carried out by means of a series of "Demonstrators".
Their standard is developed based on the knowledge acquired from each launch, both from the systems' and the aerodynamics' standpoint.

1. THE "DEMONSTRATORS"

The goals of the development demonstration phase are to prove the correctness of the general concept, to define a final aerodynamic configuration, to become familiar with the operational peculiarities of this kind of weapon, to assess the on-board systems effectiveness (flight control and armament) and to investigate on mission potentialities and possible limitations.
Four prototypes have been manufactured, with the intention to use one for ground tests and the other three an average of two mission each, thus bringing at a number of six the total amount of launches available for flight test purposes.
For the demonstration phase a bay has been created in the front side of the body, capable of housing a parachute to be used, as it will be explained later, for the retrieval after each launch mission.
It is expected that the final production standard will not differ too much from that of the Demonstrator in terms of general philosophy, geometry and main aerodynamic configuration.
The Demonstrator is not powered and has no navigation device yet. It will allow, however, for a certain extent of interventions aimed at possibly refining the system's and the aerodynamic configurations during the course of the experimental activity.

Figure 2 shows the main characteristics of Skyshark at the standard of Demonstrator n° 1.
As one can notice it is very similar to a lifting body. It exhibits a well-blended/low-drag shape and is provided with a couple of limited-surface/low-aspect-ratio wings in its back side.
At the wings tips two small winglets improve the wing effectiveness and the overall lateral stability.
Two elevons are fitted to the wings trailing edge, being the only control surfaces, which provide both longitudinal and lateral controls.
A ventral fin, normally kept retracted during the carriage, is lowered before the launch.
Two armament modules are easily connected with the main structure by means of four plugs.
Each module contains a series of firing tubes arranged perpendicular to the direction of flight, crossing the body from side to side.
Four frangible barriers, two on each side, cover the tubes faces, creating a properly shaped surface which molds to the external body contour. They get destroyed during the firing when the submunitions are ejected.

Another goal of the demonstration phase is to make a preliminary assessment of the submunitions pattern.
The lateral ejection velocity (Ve) has to be optimized in such a manner that, when combined with the Dispenser airspeed, results in a proper and manageable dissemination.
At the same time Ve has to be kept at a minimum value in order to minimize the reactions on the body produced by the firing of the charges.

The final SWD configuration implies its launch being performed with the control surfaces fully operational. This will allow a prompt stabilizing reaction capable, if necessary, of compensating any disturbances during the separation phase, like unpredictable turbulence or possible side-wash effects present during launches from under-wing pylons.
As soon as the problem of launching with active controls was taken into account, it appeared clear that this would have been possible only after a successful, deep and burdensome analysis in term of Flight Control Computer (FCC) and systems reliability.
It was clear as well that the initial FCC standard was suitable for experimental purposes only, as it had not yet been fully assessed from a reliability standpoint.
A possible FCC hardover during the release phase would have led to catastrophic consequences.
For this reason it was decided from the very initial testing stage that the Demonstrator's elevons were to be kept stuck during the store separation and that they had to take over after a suitably short time, to be defined as a best compromise.
Two contradicting needs had to be dealt with : the duration time of free-flight had to be sufficient to guarantee a safe separation but not long enough to allow the store to acquire excessive misattitudes before the computer's intervention.
This might have led to plan one or more launches with dummies, thus concentrating the testing attention solely on the separation phase.
However, it was thought that the cost of one or more dummies bound to be lost and their relative launches was unjustified if compared with the limited amount of information acquirable on the whole system's behaviour.

Once the decision was taken to launch a store provided with full FCC and control systems, a problem arose on how to recover each body after a test.
In fact, the "operational" mission is concluded when the submunitions are fired and when the onboard "self destroy" device produces its final effect.
In order to recover the Dispenser this device was not fitted to the units to be tested and a further mission phase was added for experimental purposes only.
It consists of a climb which follows the firing and leads to a speed suitable for deploying a parachute.
A slow descent by chute allows the body to reach the sea surface undamaged.
A reasonably slow sinking speed via chute was achievable only if the Dispenser weight was low enough. The reduction in weight which follows the firing of the submunitions was thought of favourably coping with this requirement. (To be noted that, during these launches, the submunitions are obviously dummies).

2. THE EXPERIMENTAL MISSION

Based on the previous considerations it was decided that the most cost-effective [illegible] of mission during the initial testing phases consisted of the following steps ([illegible] 3) :

(1) release from the parent aircraf (initially a Panavia "Tornado")

(2) [illegible]ration (Flight Control Computer switched-off one sec. long)

(3) [illegible]ontrolled flight and deceleration (according to a pre-planned scheme of gradually increasing complexity, launch after launch)

(4) firing of submunitions (dummies, actual weight)

(5) lightening and climb

(6) deployment of parachute

(7) sink and alighting on water

(8) retrieval via helicopter

It was planned that, during the first three launches, the mission had to evolve in the longitudinal plane only. Subsequently, the lateral control effectiveness would have been assessed by means of turns and track-acquiring manoeuvres.

3. ON-BOARD SYSTEMS CONFIGURATION

Four main systems are fitted to the SWD Demonstrators (fig. 4) :

- a mission control system, consisting of a Flight Control Computer (FCC), actuators, sensors and interfaces
- an armament system consisting of two armament modules and an armament control system (named "safe and arm")
- a parachute deployment system
- a Flight Test Instrumentation system.

Suitable control panels are fitted to the carrier aircraft aimed at allowing the crew to managing and monitoring the launch.

a) mission control system

The main component of the mission control system is the Flight Control Computer (FCC) which is in charge of the following activities :

- to hold the dialogue with the launch control panels located in the cockpit of the parent aircraft (pre-launch monitoring and launch management)
- to perform a cyclic BITE control before and during the mission development
- to monitor the position of the elevons before the launch, in order to ascertain that no inadvertent deflections take place during the carriage after their accurate rigging on the ground
- to activate the electrical actuator which controls the fin position
- to control the elevons positions in flight, by means of their electrical actuators, as required by the pre-planned mission (pitch attitude acquire & hold, bank angle acquire & hold)
- to perform an automatic stability control both on the lateral and the longitudinal plains
- to provide the armament control system (safe & arm) with punctual inputs aimed at commanding the submunitions firing
- to provide suitable inputs to the FTI interface, to be multiplexed and transmitted
- to command the parachute release

A vertical gyro provides pitch and roll reference inputs to the FCC for attitude control and stabilization purposes (Demonstrator configuration only).
A suitable set of air-data probes feeds the barometric airspeed/height sensors for generating proper trim commands through the control laws implemented within the FCC.
After the first launches a significant position error was found, difficult to be corrected at all incidence/sideslip angles. Therefore a new improved air-data system has been designed, which relies on a pitot boom. During the third launch, the problem was solved.
A group of control devices on the Tornado's pilot and navigator control panels allows the crew to perform a proper pre-launch monitoring and to safely manage the launch commands through a very limited amount of simple controls.
Once the carrier aircraft has reached the range area, the pilot switches the PRE-ARM control. The fin is automatically deployed and, after checking the correct elevons position,a "GO" lamp is lit if the BITE finds all the computer parameters ok; otherwise a "NO GO" lamp is lit. At this point the pilot can push the release button.

b) **armament system**

From the armament point of view the Demonstrator has the purpose of locating possible specific problems of this kind of systems, thus laying down the fundations for a final solution from both the structural and the system's point of view.
The armament modules represent the payload and will play a significant role in determining the weapon's effectiveness.
For this reason considerable efforts were devoted to allocate, within the body's volume, the maximum number of submunitions, minimizing the weight of the other components. As a consequence the modules had to be characterized by a high level of technology. Nevertheless one of the prime requirements for the project was low cost.
The actual modules architecture is the result of these contradictory requirements.
Significant efforts have been devoted to safety aspects in order to avoid the risk that, for any reason, during the carriage phase, the armament control system might be subjected to anticipated activation.
The armament control safety system guarantees an absolutely safe operation on the ground and in flight by means of a proper combination of mechanical, pneumatic and electrical protections.
These devices are automatically switched-off by a FCC command after a definite time interval from release, which allows the Dispenser to reach a distance of approximately 300 metres from the aircraft.

c) parachute system

An additional section of the control system is devoted to manage the parachute deployment.
Once the FCC's clock has counted a pre-defined time, a check is done against the maximum allowed speed for the parachute's opening.
If the check is successful, a command is sent by the FCC to the explosive bolts which lock the parachute's bay panel. This is fired away and the chute can deploy allowing a slow descent on the sea.

d) flight test instrumentation

In fig.5 the block diagram of the FTI system is shown.
The parameters acquired and transmitted are provided both by the FCC in serial PCM format and by dedicated FTI transducers.
The telemetry system takes care of transmitting the PCM stream and the analogue signals previously mixed by a multiplexer.
The transmission power is sufficient to ensure a reliable telemetry link both with the ground station and with the carrier aircraft, which is also provided with a receiver and a recorder.

The real time transmission allows the ground station engineers to perform a pre-launch monitoring additional to that carried out by the FCC.
As it can be seen in fig. 2 the Tx antenna is located on the top forward section of the body.
The FTI system fitted to the first Demostrator unit gives signals provided by the following sensors and sources:

- air data sensors
- alpha and beta vanes
- vertical gyro (pitch and roll)
- FCC internal clock
- FCC digital parameters (including elevons angular deflection) and events

After each of the first three launches the FTI system was gradually improved.
The configuration of the third launch was enhanced with, among other things, three rate gyros and three linear accelerometers, providing a precise set of inertial data for finely tuning the control laws.
A group of three linear accelerometers (high frequency, y axis) have also been installed in order to measure the dynamic response of the body during the firing phase.
The flight test instrumentation includes also a group of eight high frame-speed kinecameras suitably located in the fuselage and within under-wing pods of the parent aircraft.

Based on their films a precise post-flight trajectory analysis of the separation phase can be performed.

4. PRELIMINARY GROUND TESTS

Before undertaking the experimental flight phase several ground tests have been carried out, consisting of the following:

4.1 Wind tunnel tests

In addition to the wind tunnel tests campaign aimed at optimizing the aerodynamic shape of the weapon and at identifying a reliable data set, a dedicated series of separation tests was conducted taking into account the significant wall effect of the carrier aircraft during the release.
As previously pointed out the separation takes place, for safety reasons, with the autopilot switched-off. It is therefore mandatory that the elevons guarantee the correct trimming effect, thus maintaining a suitable attitude during the time interval which elapses before the FCC takes over.
A proper matching between wind tunnel data, mathematical model and flight test results led to the progressive improvement of the separation dynamics.

4.2 Functional tests

Before the first carriage test-flight a series of ground resonance, EMC and functional tests were carried out, involving all the on-board systems and the weapon's structure. Some peculiar floating tests have also been required in view of the foreseen retrieval phase.
In fact one of the main operational problems to be solved was the definition of the procedure to be followed by the retrieval team when approaching the weapon floating on the water. It might have been dangerous to approach the weapon in order to link it to the helicopter's rope not knowing yet whether all the firing charges had worked properly.

4.3 Safety and integration tests

A number of cartridges have been fired both under water and in plein air.
This permitted to define proper "safety bells" and to provide a sufficient background upon which to base the retrieval procedure. Before the first launch a final overall functional and integration test was conducted, simulating the complete mission by properly stimulating the air data sensors and by changing the body attitude accordingly. This final test also involved the armament system , which was realistically tested up to igniters level.

4.4 Carriage tests

Once the first SWD Demonstrator was ready to fly a short carriage flight test campaign took place, consisting of two flights.These gave the opportunity to :

- assess the interface control panels and their effectiveness to manage all pre-launch procedures

- verify the absence of significant vibrations and to assess the aeroelastic behaviour of the SWD's structure when fitted to the parent aircraft

- assess the overall vibrational environment, with particular regard to the control surfaces (elevons and retractable fin) and equipment

- optimize the complex ground procedures and ·o check the FTI telemetry systems.

During the first carriage flight some aeroelastic coupling occurred at high speed between the elevons and the wings.
Therefore the elevons were lightened and their leading edge was fitted with suitable balancing masses. The second flight proved that the problem was completely solved and the full carriage envelope was achieved.

5. RESULTS ACHIEVED

In the period May 1987-September 1988 five launches were carried out, achieving satisfactory and encouraging results. Most of the goals foreseen for the demonstration phase were accomplished, namely :

- pre-flight and emergency procedures

A gradual refinement of the on-ground operations led to optimize the pre-flight procedures, especially those concerned with the management of the active armament modules during their integration with the Dispenser and those related to the final fitting of the Dispenser to the carrier aircraft.

- pre-launch procedures

The carriage flights and the launch-flights gave several opportunities to check the reliability of the interface control panels and the suitability of the FCC pre-launch monitoring.
This system performs a sound check of most critical parameters, including the elevons position.

One of our main concerns, in fact, was due to the possible risk that, for any reason, the elevons happened to shift in flight from the position at which they had been accurately rigged on ground before take-off. This would have impinged on the flight safety of the carrier aircraft during the launch. The mechanical link which connects the elevons with their electrical actuators consists also of a friction gearing. This element, although severely rig-tested, might have exhibited an unexpected behaviour when operating into the actual flight vibrational environment.

- separation phase

At the time of starting the development phase the on-board control systems had not been yet fully qualified. For this reason, in order to avoid any possible damage to the FCC caused by shocks, it was decided that the release had to take place without intervention of gun strokes.
Therefore the safe separation was guaranteed solely by means of an appropriate setting of the elevons. This, as previously pointed out, was the result of a large wind tunnel campaign.

Based on flight trials results and additional wind-tunnel tests the aerodynamic data set was gradually refined after the first and the second launch, thus leading to optimizing the separation's dynamic.

- mission flight, aerodynamics, control laws

A few aerodynamic modifications were applied to the body's shape during the course of the experimental activity performed so far.
Some tuning of the pitch control laws was also required, however the manoeuvering and stabilizing capability was proven thus demonstrating that the typical low-height mission will be possible as far as the longitudinal plain is concerned.

- firing phase

One of the main doubts about this phase was that the correct attitude of the SWD might be compromised during the firing because of the significant lateral reactions applied to the body by the dummy submunitions during their ejections.
The sequence of firing had been carefully optimized with the aim to keep this effect at a minimum and nothing significant was noticed as far as the lateral stability is concerned.

- post mission flight

The flight phase which follows the firing, although introducing some additional load to the engineering and operational work, has proven to be economically worth to be designed.
The structure of the SWD prototype and its on-board systems are quite expensive and the possibility of recovering and of re-using them led to a significant cost saving.
The good experience acquired to this respect will be put to good use during the next phases of the program, when the SWD will carry on-board more sofisticated and e ensive navigation systems.

5.1 FIRST LAUNCH

The mission to be performed after the first launch consisted of a steady dive followed by a levelled decelerating flight at a constant height up to the firing phase. The launch was not completely successful. In fact it lasted only a few seconds before the flight was abruptely stopped by an anticipated splash-down. The Demonstrator was lost but the data acquired proved to be sufficient for carrying out a deep analysis.
The reasons of such misbehaviour were then clarified and proper corrective actions undertaken. The separation phase pointed out some discrepancies between predictions and actual results (fig. 6). The body, in fact, left the carrier aircraft very safely but too fast because of an eccessive pitch-down. When, one second later, the Flight Control Computer took over, the foreseen dive path resulted too steep and the levelled flight was never recovered.
A first reason of mismatching between theoretical predictions and flight data was found in minor differences between the shape of the actual body and that considered as a reference for wind-tunnel tests and data-bank construction.

In particular the actual SWD was devoid of blisters at the junction wing/body. As illustrated in fig.7 the lack of blisters implies a significant contribution to pitching moment at incidence angles lower than 5°.
After readjusting the mathematical model by means of this and other minor corrections, a satisfactory matching was achieved. The FCC trim laws and controls were amended and a new deflection of the elevons at release was defined.

5.2 SECOND LAUNCH

The mission profile of the second launch was precautionally simplified with the aim to minimize the disturbances induced by the manoeuvre. The attention was mainly concentrated on the system's stabilizing capability in steady flight, which had not yet been assessed. The launch did confirm the effectiveness of the above modifications with respect to the separation phase (fig.8) and to the gliding pitch behaviour at high incidence. *An unexpected misbehaviour was instead exhibited at relatively low* incidence, in terms of lateral unsteadiness and pitch-roll dynamical coupling. The time histories shown in fig.9 point out very clearly that the unsteady dynamics of the SWD develops quite suddenly into a steady flight when the angle of attack crosses the threshold of approximately 12° (t = 10 sec.).

After a deep analysis it was found out that some apparently slight alterations in the shape of the body could significally affect the directional and longitudinal aerodynamic characteristics. The most effective shape alterations effectively tested in wind-tunnel were (fig.10):

- sealing of the retractable-fin slot
- installation of a peripheral spoiler on the backward side of the body
- *addition of two fins* on the top wing surface
- addition of proper blisters between the dummy engines fairing and the fuselage.

Based on those results it appears that the configuration of the second launch presents a range of α where the yaw coefficient is negative. By sealing the fin's slot a certain improvement is achieved although still a range of instability remains. A further improvement is produced by the adoption of a small spoiler on the tail. Very powerful is the effect of the two fins the beneficial action of which develops just at low incidence while at high incidence the shade of the wing makes their influence less large. *The adoption of these modifications implies improvements of the* longitudinal characteristics as well. In fact the trimmed condition (for δ elevons = 0 deg.) results shifted towards higher values of C_L, thus reducing the trim drag and allowing a larger range of δ available for actuating the control surfaces.
As a consequence of such aerodynamical re-design a new prediction was necessary as far as the separation phase was concerned.

This, again, was performed by properly mixing wind tunnel results with those obtained by means of a mathematical model which properly takes into account the aircraft wall effect (fig.11).
The autopilot control laws were also slightly readjusted. The airspeed signal which back-feeds the pitch trim was filtered because of eccessive noise. The pitot tube, initially placed on the top forward side, was replaced by a pitot boom. Immediately after the firing phase, in fact, the body is subject to a pitch-up which follows the sudden lightening. Due to the shading effect of the body's nose the pitot had provided unreliable information (lower speed) which had caused an anticipated deployment of the parachute.

5.3 THIRD LAUNCH

Also this launch foresaw pitch manoeuvres only. It developed in good agreement with the predictions.
Only minor discrepancies were found. After the launch and the FCC activation the SWD, instead of proceeding in level flight, undertook a straight trajectory slightly climbing (γ ~2°, fig.12). This was due to a small constant error present in the airspeed measurement loop, which caused a slight mistrim.

Fig.13 compares the theoretical trim relationship (θ - V) stored within the FCC and the actual one measured in flight. The two curves are just shifted as much as large was the airspeed error (~ 25 Kts).

A few problems were again exhibited during the firing of the submunitions. In fact some of the envelopes which bring the dummy submunitions out of the launching tubes did not exit completely from the body's sides (fig.14).
These protrusions severely affected the aerodynamic shape of the SWD which presented unbearable asimmetries. Therefore, during the final part of the firing phase, the SWD performed abrupt and uncontrolled manoeuvres which led to reach anomalous attitudes. As a consequence the submunitions' pattern resulted affected, which is operationally unacceptable. This problem, however, was easily solved by weakening the contours of *the pre-cut holes printed on the lateral frangible barriers.*

5.4 FOURTH LAUNCH

No modifications were introduced before the fourth launch whilst minor modifications were introduced into the trim control laws and the pitot's position error was compensated.
The mission profile incorporated also manoeuvres in the lateral plane (30° bank turn left followed by level flight recovery). The flight developed very stable and successfully thus allowing for a more complex mission.

5.5 FIFTH LAUNCH

More complex manoeuvres were planned in order to investigate on the system's capability in a larger range of attitudes. It foresaw, after the launch, a steep descent, a 30° bank-to-bank, a pull-up and a levelled flight up to the firing point. The mission performed fully successful. Fig.15 shows the pitch and roll attitudes time histories, recorded in flight compared with those simulated. One can notice a satisfactory matching. (To be pointed out that the launch was performed in presence of severe turbulence). One of the aims of this fifth launch was a preliminary assessment of the submunitions pattern. This was performed by means of a series of cartriges sufficiently representatives of the dummy submunitions in terms of inertia and aerodynamics, containing balloons self-inflating at the moment of their impact with the sea surface. This technique did prove to be successful. After the launch and the completion of the SWD mission, the carrier aircraft overflew the firing area taking pictures form a preliminary assessment of the ground pattern.

5.6 SIXTH LAUNCH

A sixth and final launch is foreseen by the end of October 1988. The mission will develop as close as possible to one of the most complex operational missions. It will imply the highest launching speed and 60° bank angle turn.
An accurate measurement of the submunitions pattern will be also performed by means of a larger number of floating markers.

6. CONCLUDING REMARKS

Although the Demonstration Phase is not completely concluded yet, one can reasonably maintain that the main objectives have been successfully met.
In fact, thanks the encouraging results acquired so far, the go-ahead for a further phase has been given.Based on the Demonstrator's aerodynamic concept the development of a stand-off "Cluster bomb" is already in progress.
This will consist of a body aerodynamically very close to that of "Skyshark", still unpowered, but provided with navigation capability. The new experimental phase will start within one year and will be mainly aimed at assessing the navigation system accuracy, effectiveness and reliability as well as at defining the operational procedures.

Some general conclusions can be drawn about flight-test experience acquired during the Demonstration phase. Aeritalia's Flight Test Center had, so far, developed a remarkable knowledge about military aircraft and weapons like missiles, but some experimental aspects related to testing the Skyshark resulted beyond the usual experience.
In particular, the requirement to coexist with the economical restrictions typical of a private venture like this led to the need of recovering and re-using the body after each mission. Being the launch performed some miles off-shore for obvious safety reasons significant efforts were required from the operational stand-point.

As far as technical aspects are concerned, this programme forced Flight-test and Design departments to tighten even more their relationship in order to accelerate the process of analyzing, interpreting and matching the data acquired and repredicting the dynamics of the subsequent launch. In some cases, in fact, this loop had to be completely performed in a few days, to opportunely allow for the longer procedure of introducing the necessary hardware and/or software modifications into the Flight Control Computer.
Very interesting and instructive resulted some aerodynamical peculiarities pointed out by such unusually shaped lifting body. In fact it showed quite a high sensitivity to minor shape asymmetries and excrescencies with respect to winged bodies, where the lift is more largely distributed. This requested a continuous refinement of the aerodynamical data set by means of flight-test results, wind-tunnel data and mathematical model. Fig.16 shows, as a typical example, the matching obtained so far between theoretical and experimental polars.
On concluding one can say that, after one year testing activity consisting of five launches the general concept has proven to be effective, the aerodynamic configuration has been developed up to a satisfactory stage and the on-board systems largely tested. Based on all that, the bases for the next phase have been soundly layed down.

ACKNOWLEDGEMENTS

The information presented are the results of Aeritalia/SNIA BPD team work.
To the successful performance of the flight-test campaign which is the subject of this paper a sound operational contribution was provided by several bodies of the Italian Air Force and by the Officials of the Salto di Quirra Joint Forces Missiles Testing and Training Range.
The authors intend to acknowledge the contribution of all those who have collaborated to realize the "Skyshark" program.
The authors are also indebted to the students of the "A.Volta" Technical Institute (Alessandria) that, not being any drawing of the Telebomb available, obtained one by directly measuring the actual bomb. This, at present, is subject of a restauration work carried out by an Italian Aircraft Amateurs Association, GAVS, which in Italian Stands for Group of Friends of Historical Aircraft.

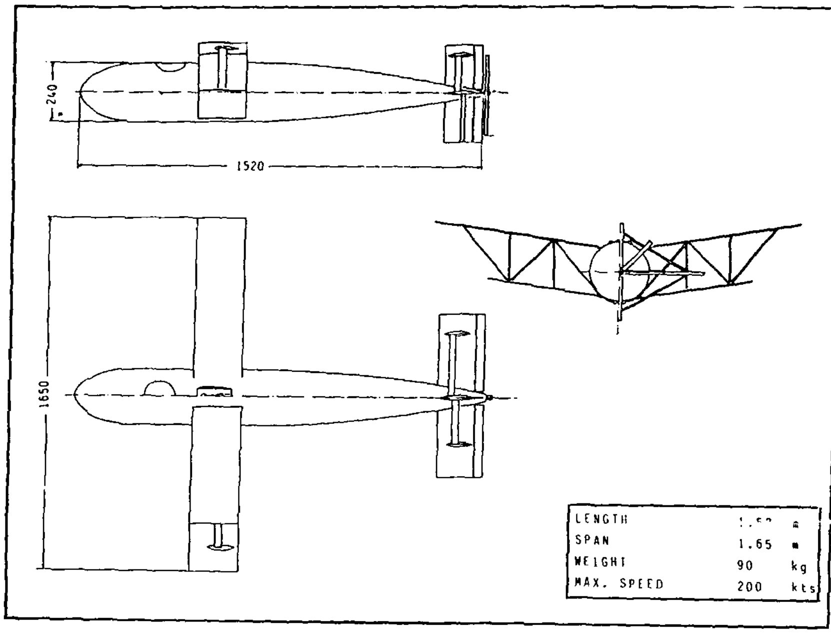


Fig.1 – 1918 : "Telebomb" Guidoni – Crocco

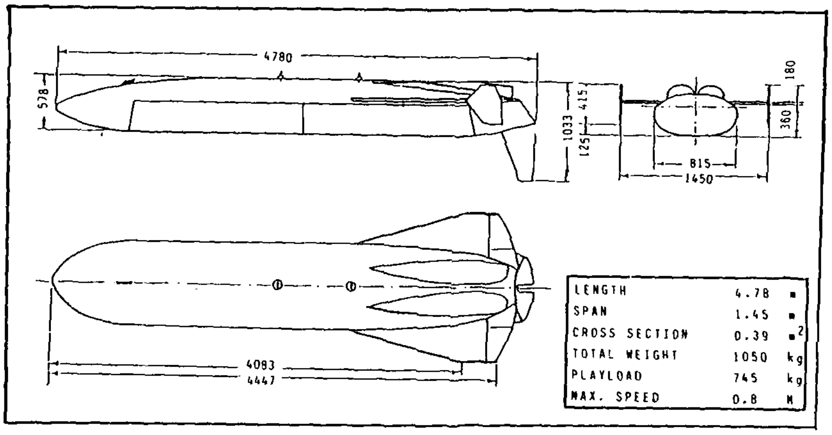


Fig.2 – 1988 : "Skyshark" Stand-Off Weapon Dispenser

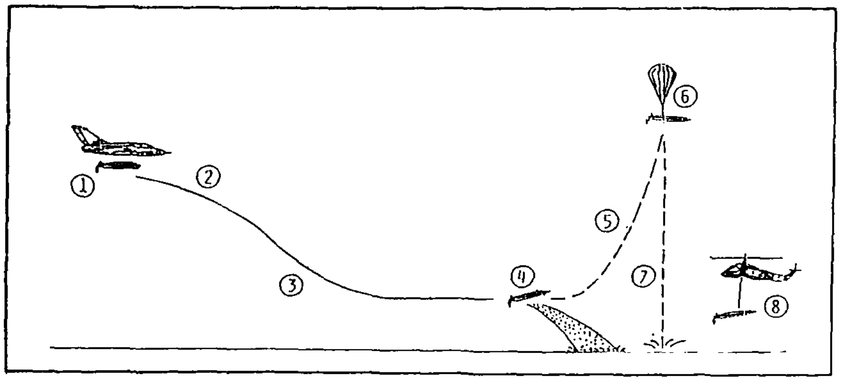


Fig. 3 - Generic Experimental Flight Mission

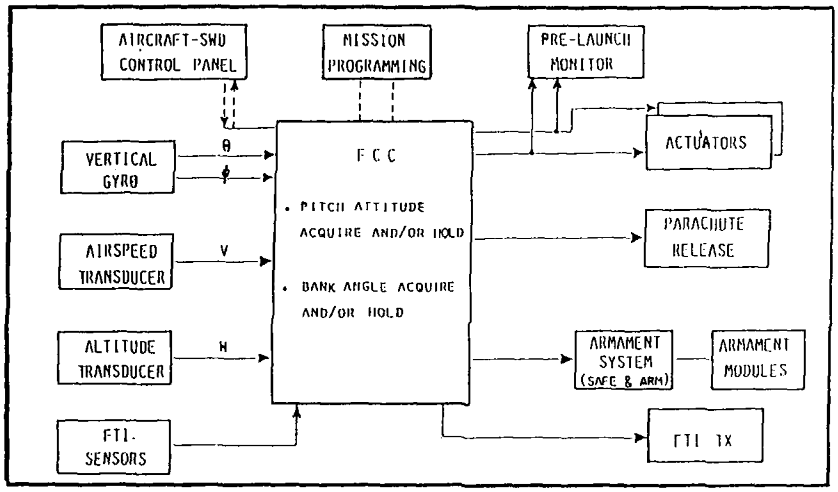


Fig. 4 - On-board Systems Configuration

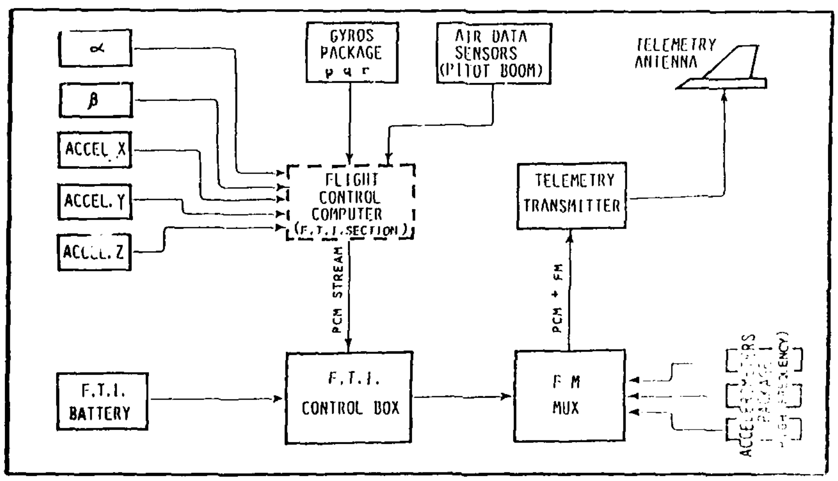


Fig. 5 - Block Diagram of F.T.I. System

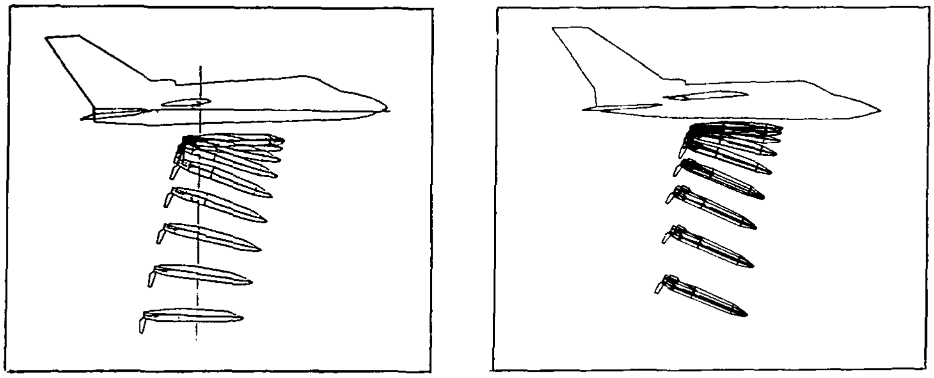


Fig.6 - First launch : predicted and actual separation

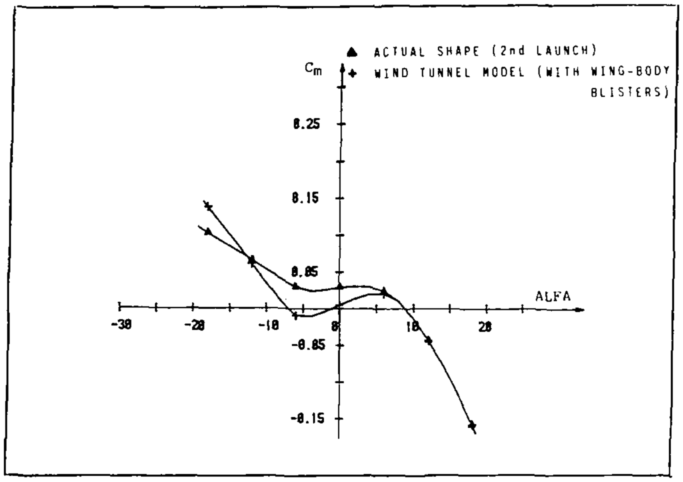


Fig.7

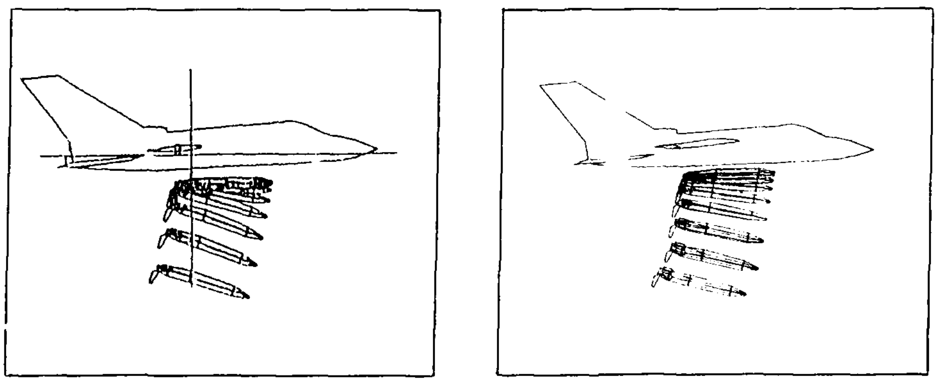


Fig.8 – Second launch : predicted and actual separation

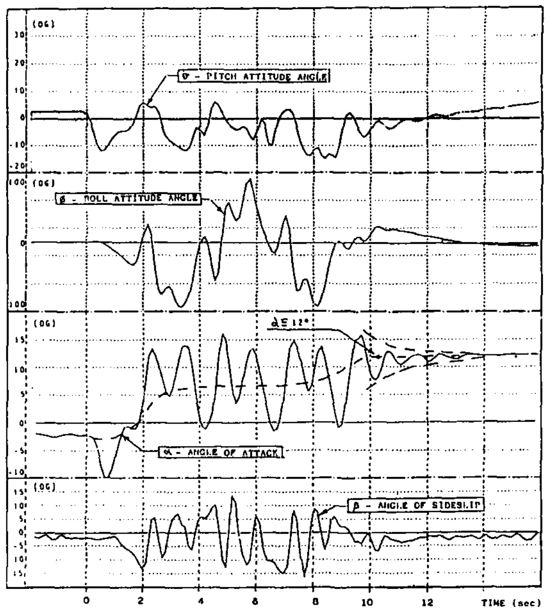


Fig.9 - Second launch : time histories

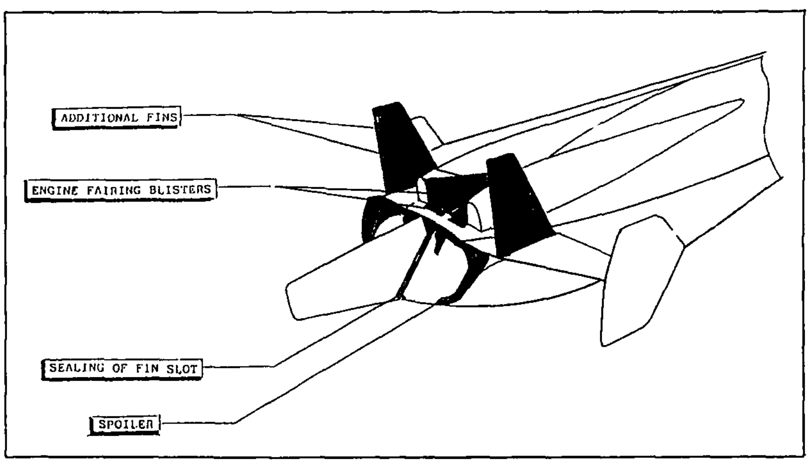


Fig.10 - Modifications after second launch

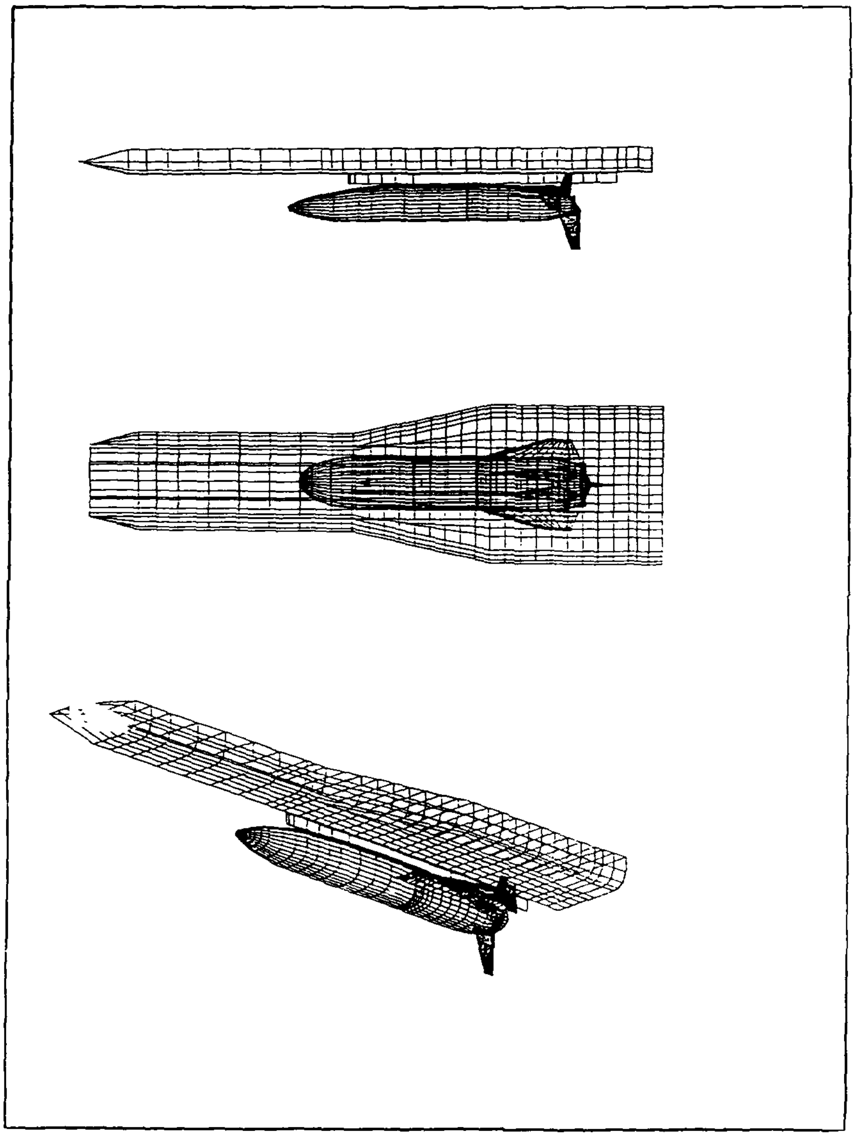


Fig.11 - Separation, mathematical model

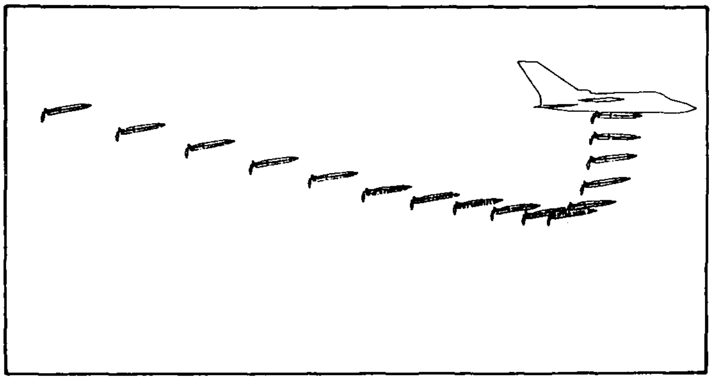


Fig. 12- Third launch : slight climb due to speed indication error
(Time interval between two positions of the store .40 sec.)

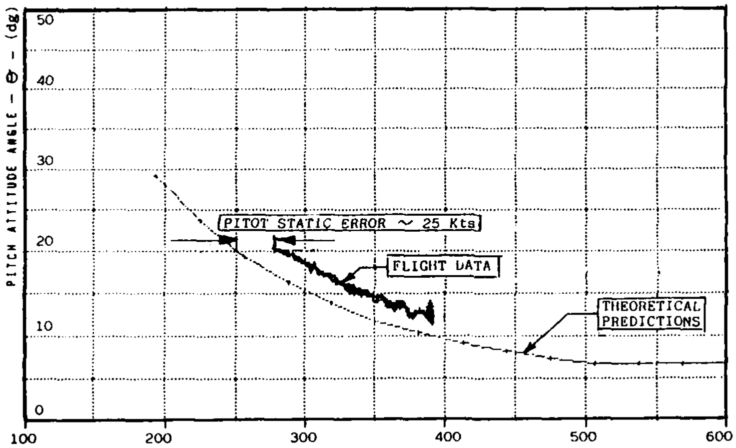


Fig. 13

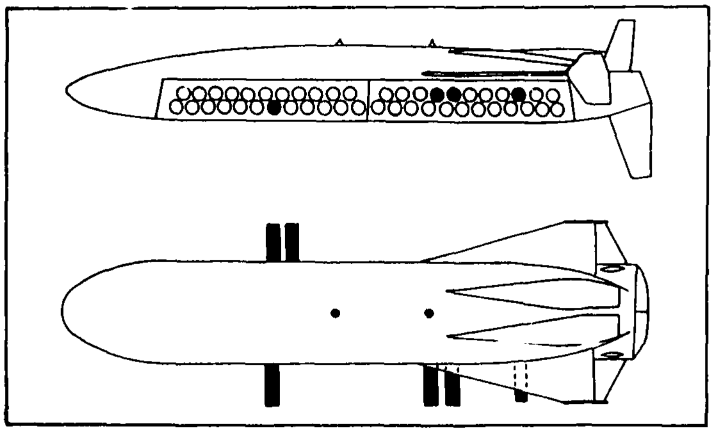
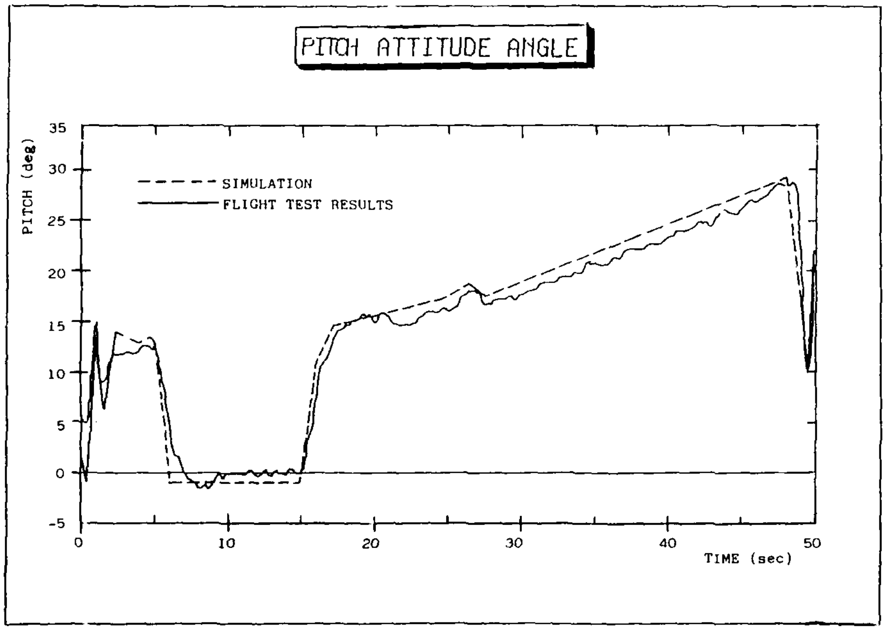


Fig. 14 – Defective expulsion of dummy submunitions



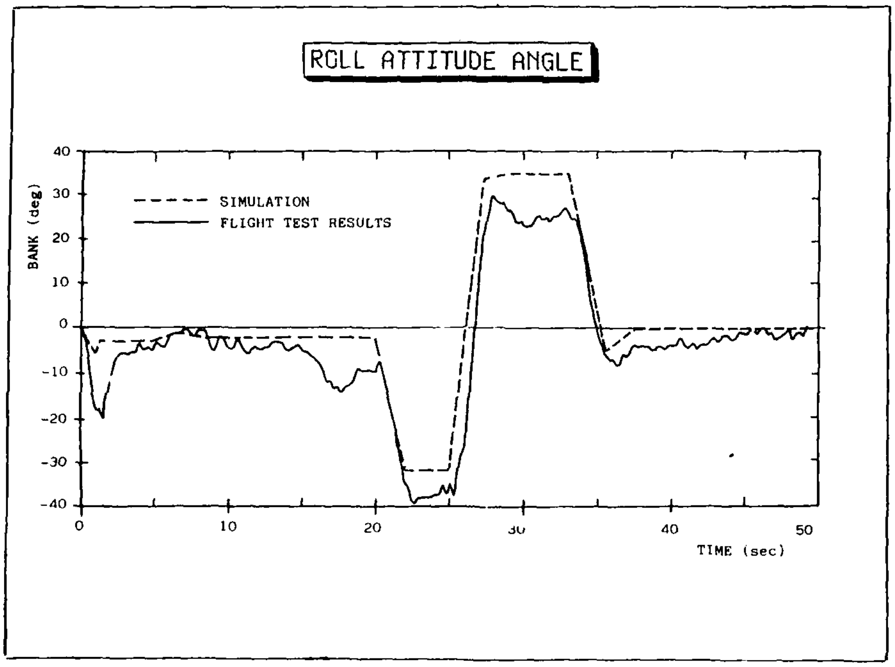
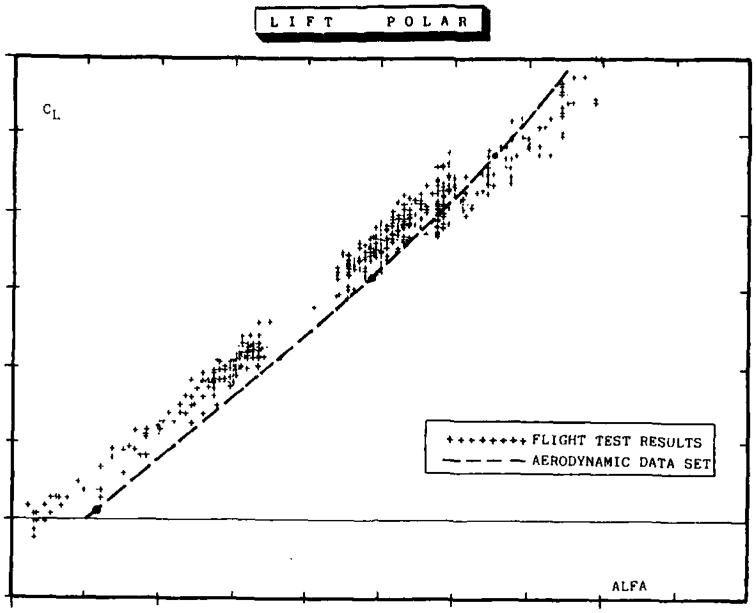


Fig.15 - Fifth launch



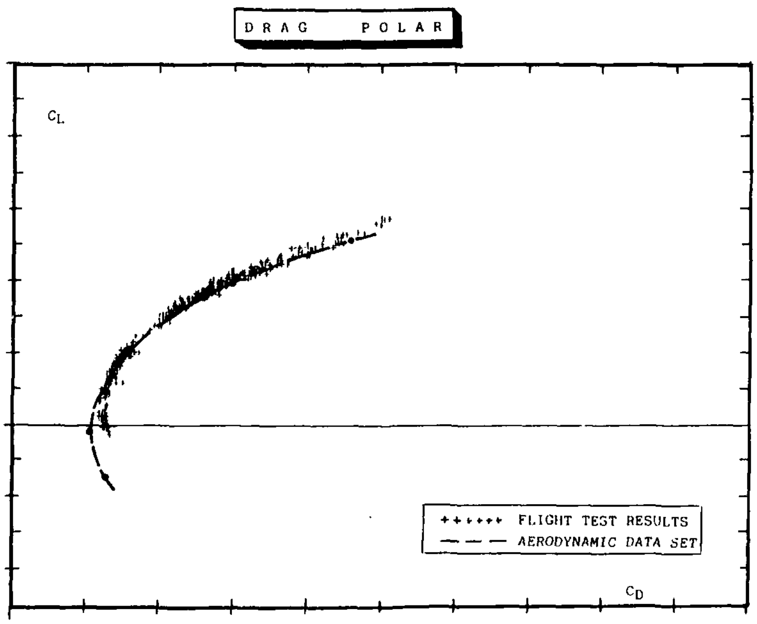
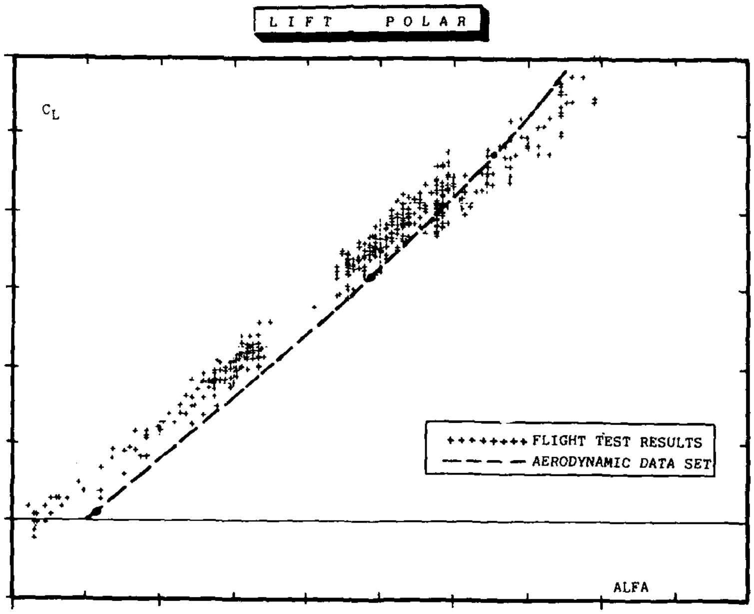


Fig.16 – Lift and drag polars (theoretical and measured)

(scales intentionally missing)



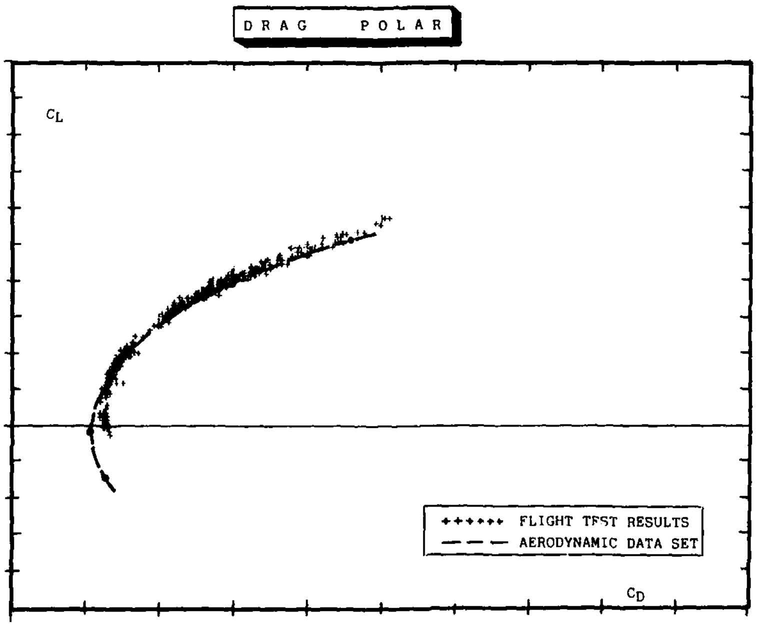


Fig.18 – Lift and drag polars (theoretical and measured)

(scales intentionally missing)

IDENTIFICATION FRIEND, FOE, OR NEUTRAL JOINT TEST

Mr James E. "Jack" Haile
Technical Advisor

IFFN JTF
KIRTLAND AIR FORCE BASF, NEW MEXICO 87117

SUMMARY

The Identification Friend, Foe, or Neutral (IFFN) Joint Test Force (JTF) located at Kirtland AFB, NM, has developed a testbed that is composed of high fidelity, real time man-in-the-loop simulators designed to replicate the NATO Central Region Integrated Air Defense System. The purpose of the test is to assess the ability of this air defense system to correctly identify and engage enemy aircraft. The testbed represents the largest real time command and control (C^2) simulation which consists of 57 medium and high fidelity tactical consoles and over a million lines of code. The OSD-sponsored testbed development and test is scheduled to run through July 1989. After this the testbed will become the Theater Air Command and Control Simulation Facility (TACCSF) operated by the USAF Tactical Air Warfare Center (TAWC). The facility will be used by both Army and Air Force commands to resolve joint operational issues and support test and evaluation of the NATO Air Command and Control System (ACCS).

INTRODUCTION

One of the most serious problems facing the NATO air defense forces is the correct and timely identification of aircraft in a tactical air environment which includes large numbers of friendly and hostile aircraft, electronic warfare threats, and surface-to-air, and air-to-air missiles that operate beyond visual range (BVR). Numerous studies have concluded that the current electronic identification capabilities are too slow, individually unsuitable for positively identifying hostiles and friends, have insufficient range, and are subject to interference from electronic countermeasures and other environmental factors.

Using real time, man-in-the-loop simulations of Army, Air Force, and NATO C^2 and weapons systems, IFFN was designed to: assess the baseline capabilities within the NATO Integration Air Defense C^2 system to perform the identification function; identify deficiencies in the performance of the identification function; and, define potential near-term procedural and equipment changes to overcome the deficiencies.

TESTBED DESCRIPTION

The overall operational scope of the test is summarized in Figure 1. It focuses on the NATO Integrated Air Defense System within the Lauda Battle Management Area. Consistent with defense planning guidance, a standard 14/10 air scenario is used with the NATO air defenses defending against a Warsaw Pact three wave attack. Although NATO offensive air and support operations are not represented with manned simulators, they are explicitly flown in the air scenario. In addition, the operations focus on a 1989 time frame using current AAFCE air defense plans and air space control procedures.

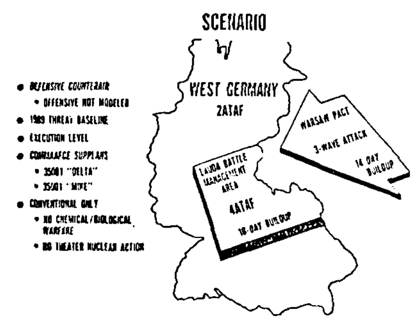


Figure 1

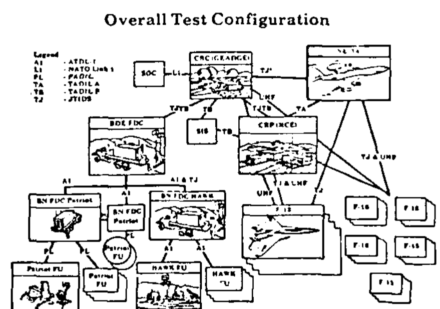


Figure 2

The overall testbed configuration representing the Lauda Integrated Air Defense Systems is depicted in Figure 2. There are three types of weapons systems: PATRIOT, HAWK Army surface to-air missile systems, and the Air Force F-15. Associated with these weapons are their respective command, control, and communications (C^3). In the case of the PATRIOT and HAWK there are battalion (BN) fire direction centers (FDCs) with a brigade FDC (AN/TSQ-73) controlling the two BNs. For the F-15's, the primary C^2 is a CRP (Modular Control Equipment) and a NATO E-3A. Controlling the overall Army and Air Force is the NATO CRC (German Air Defense Ground Environment equipped). Every node has its appropriate voice and digital communications replicated. A total of 57 high and medium fidelity simulators comprise the testbed. The high fidelity simulators are used for data collection and are run by actual field operators. These are depicted in Figure 2 by the large boxes. The lower fidelity simulators are operated by trained technicians and are used to load the high fidelity simulators. These are represented by the smaller boxes.

The overall testbed design is depicted in Figure 3. A central simulation system is used to generate the comprehensive aii scenario with appropriate tactics, airspace control procedures, and terrain. This scenario is then provided to four different types of simulation through a communications subsystem. These simulators are: PATRIOT, HAWK, F-15, and air C^2. All simulators are located at Kirtland AFB, NM, with the exception of the PATRIOT which is at Ft Bliss, TX.

TESTBED DESIGN

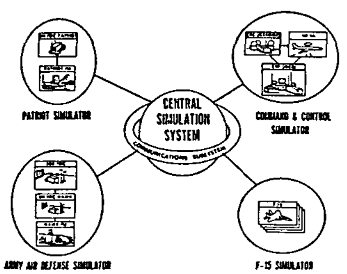


Figure 3

PATRIOT SIMULATOR

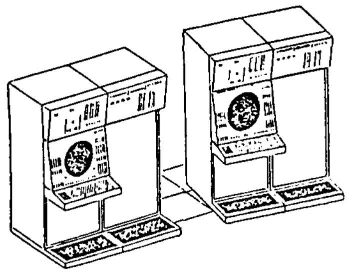


Figure 4

Figure 4 depicts the PATRIOT Tactical Operations Simulators at Ft Bliss, TX. The consoles depicted replicate a PATRIOT fire unit; another is used to replicate the PATRIOT BN FDC. These high fidelity systems are used by the Army for prototype software development.

Next in Figure 5 is a picture of the HAWK fire unit simulator. All key functions of the HAWK system are replicated and eight are used in a BN.

HAWK SIMULATOR

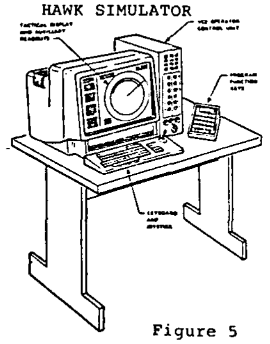


Figure 5

F-15 SIMULATOR

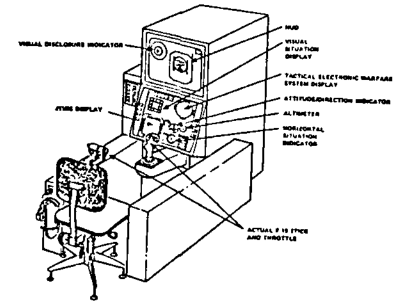


Figure 6

Figure 6 is the F-15 simulator. It uses an actual F-15 stick and throttle (MSIP) with appropriate switches. The upper screen is the F-15 heads up display which projects airspeed, altitude, attitude, radar target box, and missile parameters. The lower screen includes all the flight instruments, radar, Electronic Warfare Warning System (EWWS), Tactical Electronic Warfare System (TEWS), and Joint Tactical Information Distribution System (JTIDS). This flight simulator is unique in that hundreds of friendly and enemy tracks can be presented to the pilots in a realistic jamming environment.

Figure 7 is the IFFN tactical command and control display (TCCD). The right screen is a touch screen that commands system functions. The left display is in color and portrays appropriate air situations. The special function keys on the left are software reprogrammable and are set up with a template. Thirty of these TCCD's are used in the IFFN testbed.

TACTICAL COMMAND AND CONTROL DISPLAY

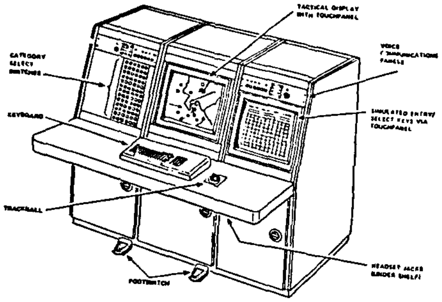


Figure 7

IFFN TESTBED

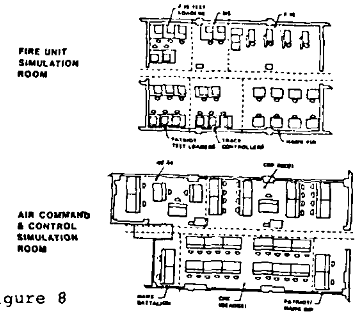


Figure 8

Figure 8 portrays the two simulation rooms in the IFFN testbed. Situated in the Air Command and Control Simulation room are the 30 TCCD consoles which are configured as follows: 5 for the NE-3A, 8 for the CRP, 12 for the CRC, 2 for the HAWK BN FDC, 2 for the Brigade FDC, and 1 for the Sector Operations Center. All of these consoles are software reprogrammable so other configurations could be addressed. The fire unit simulation room contains: 4 F-15's, 8 HAWKs, 6 PATRIOT test loaders, 5 F-15 test loaders, and 2 test loaders for the special information systems. In addition, there are the four consoles at Ft Bliss for the PATRIOT fire unit and BN FDC.

Figure 9 summarizes some of the overall testbed features that makes it the largest air C^3 simulation in the world.

TESTBED FEATURES

- o 59 HIGH TO MEDIUM FIDELITY CONSOLES
  - SIMULATING 12 TYPES OF TACTICAL SYSTEMS
  - USING 7 TYPES OF DATA LINKS
- o 1.2 MILLION SOURCE LINES OF FORTRAN CODE
- o COMPUTERS
  - 18 CONCURRENT 3250/60/80 COMPUTERS
  - 21 ARRAY PROCESSORS
  - 462 MILLION INSTRUCTIONS PER SECOND CAPACITY
- o EXERCISE CAPACITY
  - 2000 X 2000NM WITH DMA TERRAIN DATA
  - 2000 ACTIVE AIRCRAFT
  - 128 MAXIMUM EXERCISE PARTICIPANTS

Figure 9

IFFN TESTING

The IFFN JTF to date has conducted four operational tests involving the PATRIOT, PATRIOT BN FDC, HAWK, and the F-15. Four more operational tests are scheduled through 1988 with the HAWK BN, Brigade FDC, NE-3A, and CRP. The grand finale is the ninth test in which the CRC controls all the fire units and subordinate C^2 nodes. This testing will show how each element with its associated communications both individually and collectively contributes to the air defense process. Testing to date is showing that complete situation awareness is difficult to obtain, there are limitations in correctly identifying aircraft, and there is a clear need for integrated Army/AF/NATO operational testing and training under simulated wartime conditions.

FOLLOW-ON TESTING

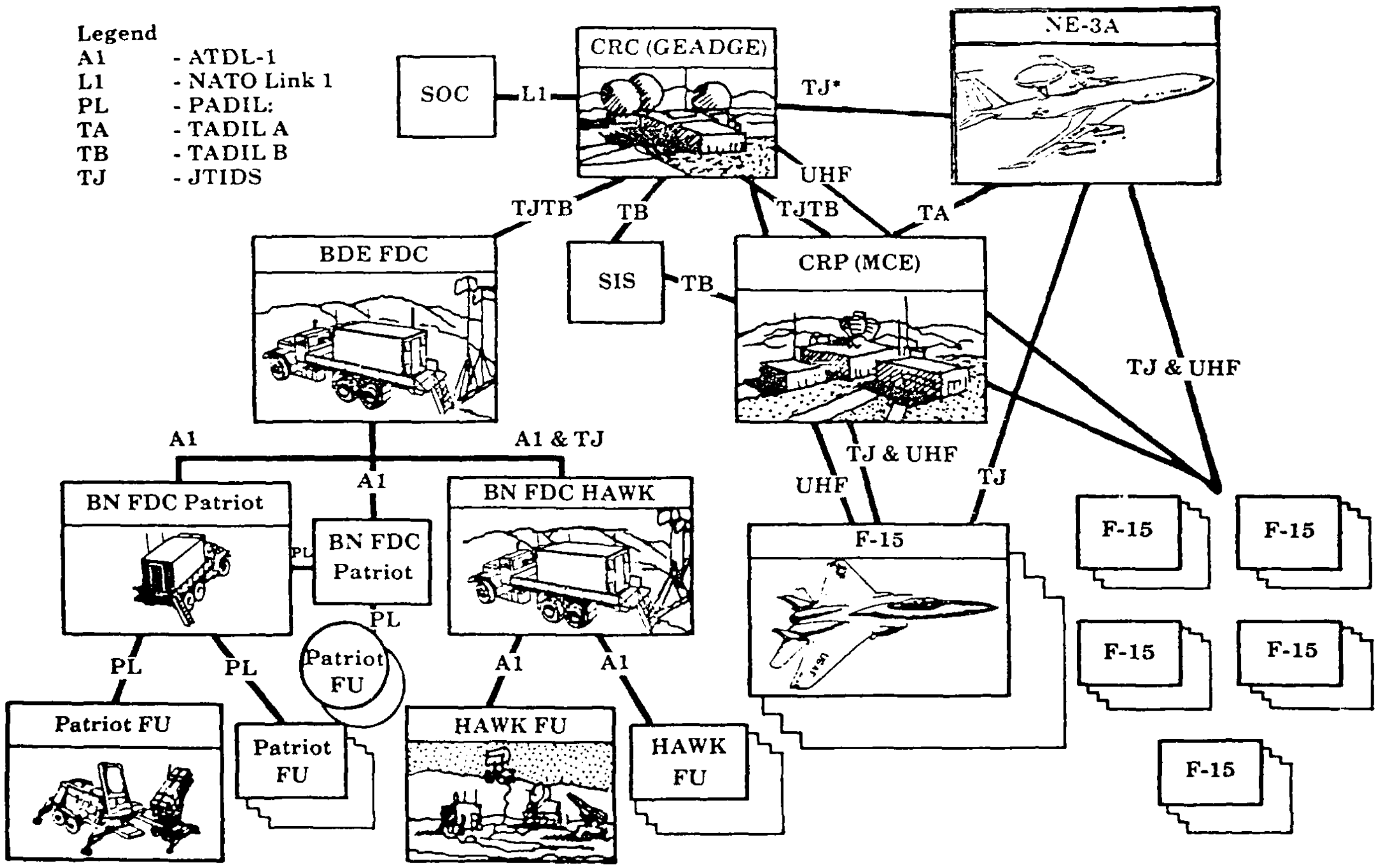
In 1989, the IFFN JTF will complete its OSD directed testing. At that point, the USAF Tactical Air Command (TAC) and Army Training and Doctrine Command (TRADOC) have agreed to assume ownership of the testbed. TAC will be the executive and the name of the facility will change to Theater Air Command and Control Simulation Facility. Users will be the Army, Air Force, OSD, and NATO. The new mission will encompass:

- Concepts, tactics, and procedures development
- Developmental and operational testing
- C^2 systems integration and training.

SUMMARY

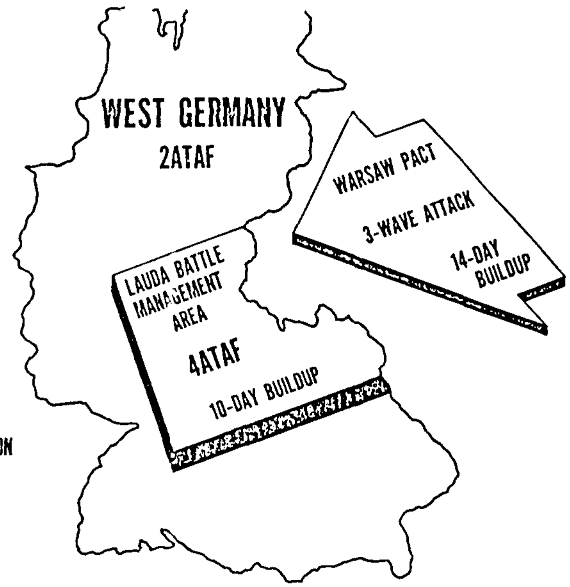
In summary, the IFFN testbed development is on schedule and demonstrating viable capability. IFFN test results to date have produced significant operational insights that are being addressed by the Army, Air Force, and NATO operational communities. The Army, Air Force, and NATO see great benefits in using the TACCSF to investigate a host of very complex C^3 issues starting in 1990.

Overall Test Configuration



SCENARIO

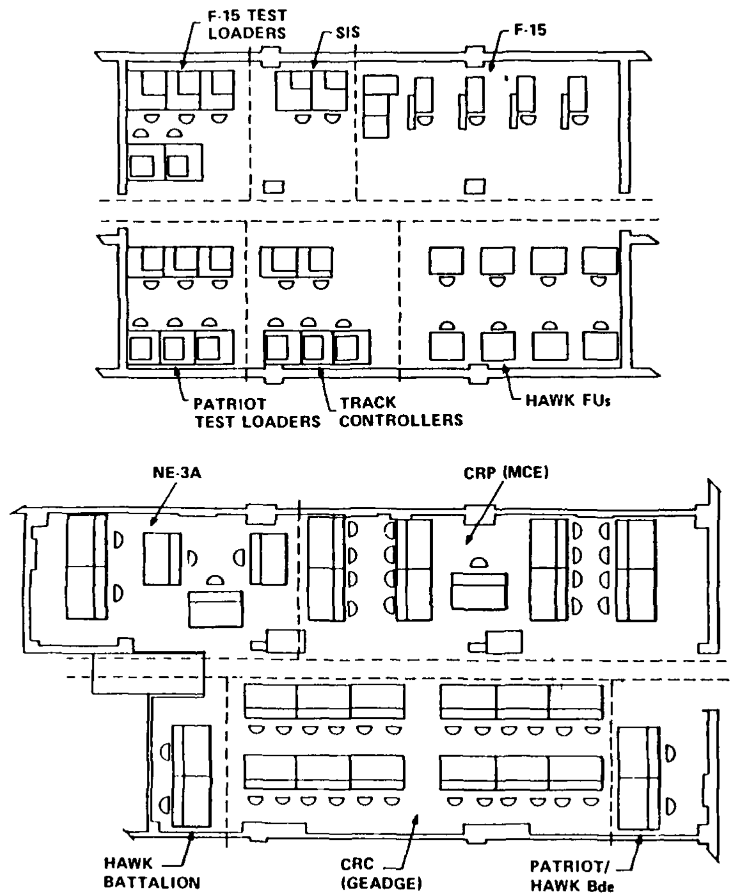
- DEFENSIVE COUNTERAIR
  - OFFENSIVE NOT MODELED
- 1989 THREAT BASELINE
- EXECUTION LEVEL
- COMMAAFCE SUPPLANS
  - 35001 "DELTA"
  - 35101 "MIKE"
- CONVENTIONAL ONLY
  - NO CHEMICAL/BIOLOGICAL WARFARE
  - NO THEATER NUCLEAR ACTION



IFFN TESTBED

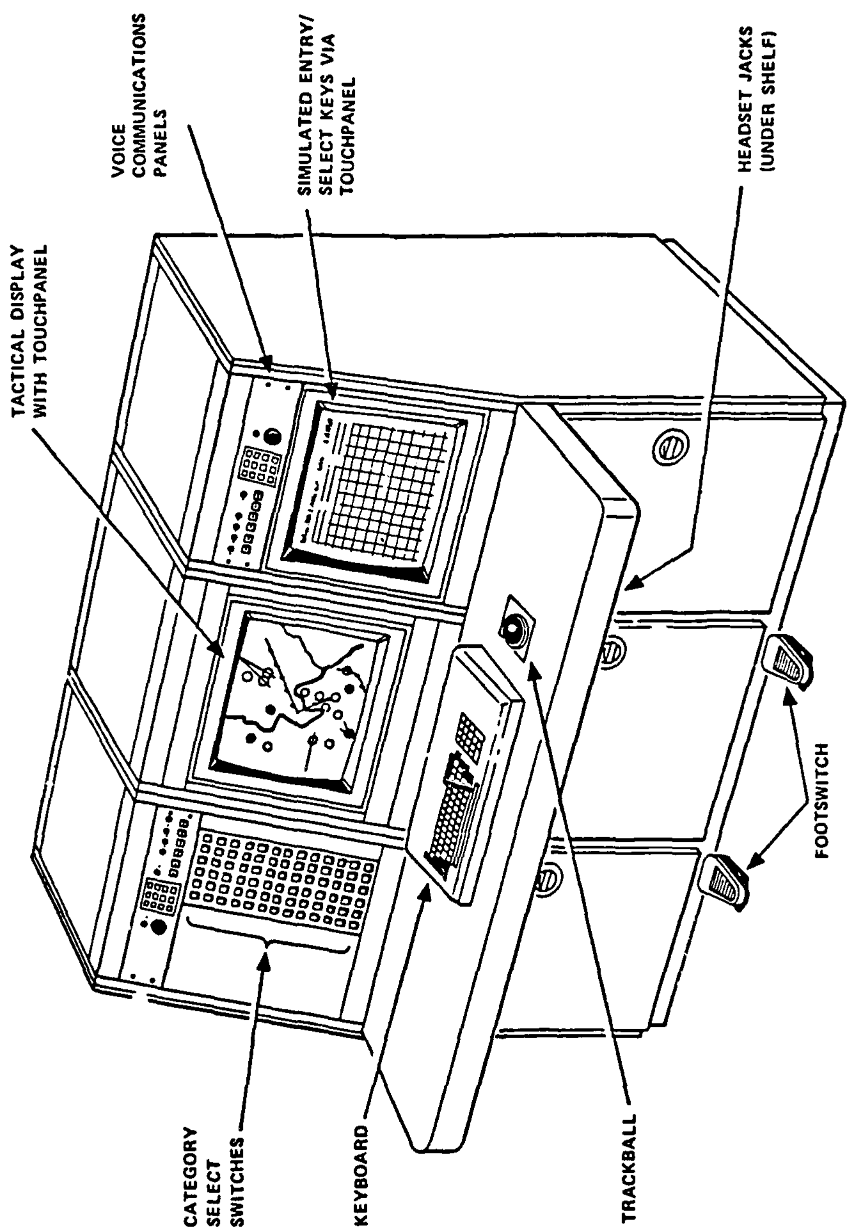
FIRE UNIT SIMULATION ROOM

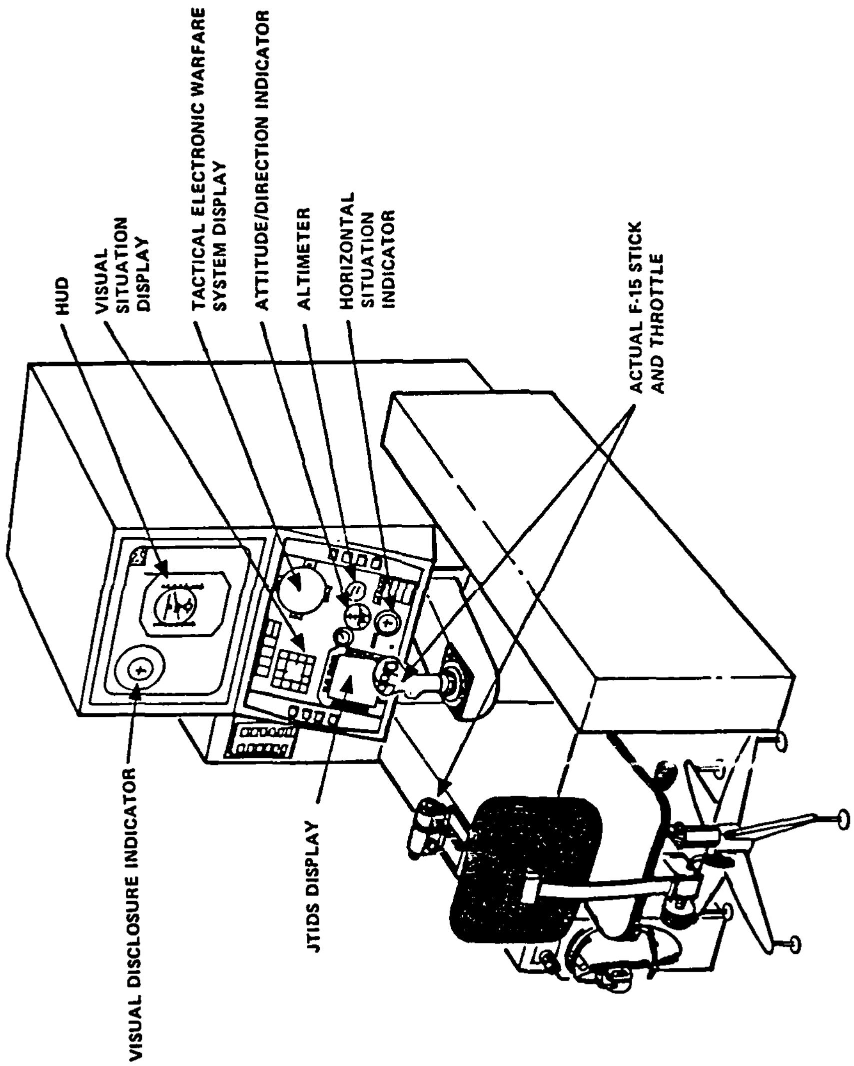
AIR COMMAND & CONTROL SIMULATION ROOM

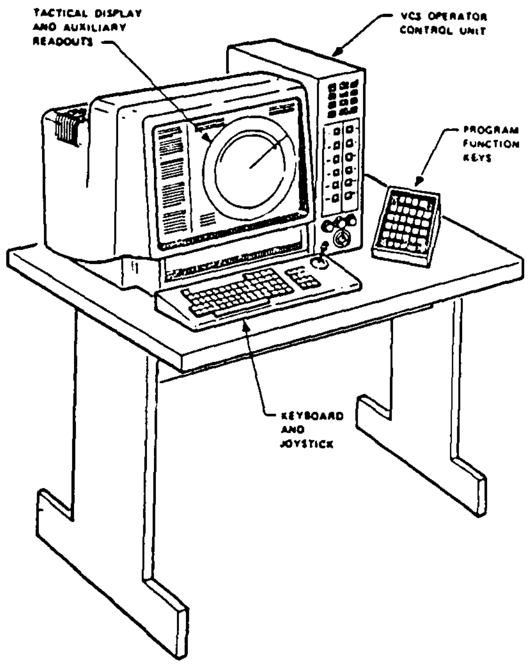


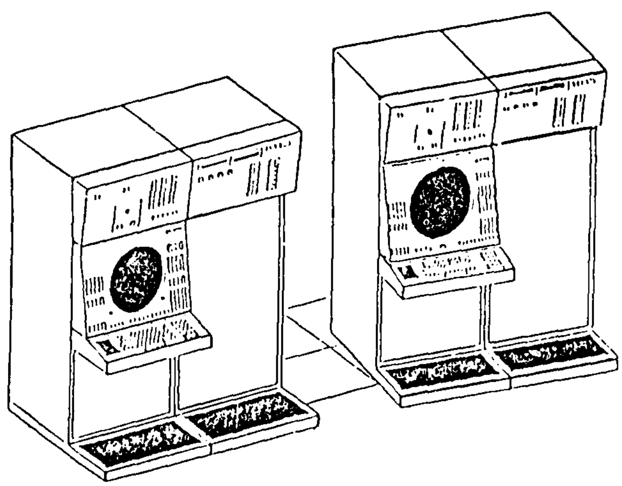
TESTBED FEATURES

- o 59 HIGH TO MEDIUM FIDELITY CONSOLES
  - SIMULATING 12 TYPES OF TACTICAL SYSTEMS
  - USING 7 TYPES OF DATA LINKS
- o 1.2 MILLION SOURCE LINES OF FORTRAN CODE
- o COMPUTERS
  - 18 CONCURRENT 3250/60/80 COMPUTERS
  - 21 ARRAY PROCESSORS
  - 462 MILLION INSTRUCTIONS PER SECOND CAPACITY
- o EXERCISE CAPACITY
  - 2000 X 2000NM WITH DMA TERRAIN DATA
  - 2000 ACTIVE AIRCRAFT
  - 128 MAXIMUM EXERCISE PARTICIPANTS

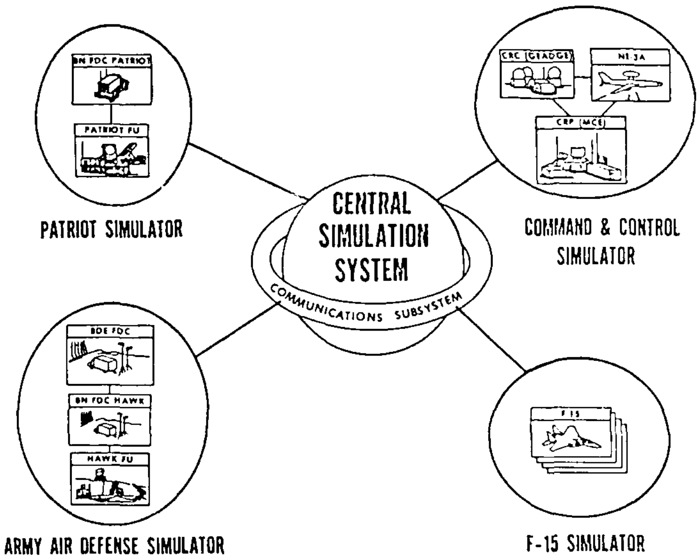

TACTICAL DISPLAY
WITH TOUCHPANEL
VOICE
COMMUNICATIONS
PANELS
SIMULATED ENTRY/
SELECT KEYS VIA
TOUCHPANEL
HEADSET JACKS
(UNDER SHELF)
FOOTSWITCH
TRACKBALL
KEYBOARD
CATEGORY
SELECT
SWITCHES


HUD
VISUAL SITUATION DISPLAY
TACTICAL ELECTRONIC WARFARE SYSTEM DISPLAY
ATTITUDE/DIRECTION INDICATOR
ALTIMETER
HORIZONTAL SITUATION INDICATOR
ACTUAL F-15 STICK AND THROTTLE
VISUAL DISCLOSURE INDICATOR
JTIDS DISPLAY


TACTICAL DISPLAY
AND AUXILIARY
READOUTS
VCS OPERATOR
CONTROL UNIT
PROGRAM
FUNCTION
KEYS
KEYBOARD
AND
JOYSTICK



TESTBED DESIGN



FLIGHT TESTING AND FLIGHT RESEARCH: FROM THE AGE OF THE TOWER JUMPER TO THE AGE OF THE ASTRONAUT

RICHARD P. HALLION
Visiting Professor of Military History
Military History Institute
Upton Hall, Room 115
U.S. Army War College
Carlisle Barracks, PA 17013-5008
U.S.A.

Since the beginning of flight, aerospace vehicle design has depended upon data gathered from the performance of actual flight vehicles. This flight testing and flight research process has led to today's air-and-space-craft, and points the way for future flight. Within this process, the flight test planner, test pilot, and flight test engineer occupy positions of critical importance.

A review of the history of flight testing and flight research indicates that both have been traditionally characterized by a professional approach emphasizing the derivation and utilization of methodologies to best enable investigators to acquire a maximum amount of information as quickly as is consistent with safety. As the technological sophistication of aircraft systems has increased, so, too, has the necessity for improving, modifying, and adding to the capabilities of the flight testing and flight research process. This pattern may be expected to hold true for the subsequent development of future advanced aerospace vehicle systems.

Flight testing and flight research are as old as flight itself. There is the myth of Icarus, who experienced structural failure from heating effects, leading to subsequent loss of control. More factually, Eilmer of Malmesbury, a Benedictine monk (and the first test pilot worthy of the name), made a short gliding flight marred by a landing accident from a loss of longitudinal control about 1000 A.D., from Malmesbury Abbey in Wiltshire, England.[1] In the Nineteenth Century, a coachman and a small boy flew for a few yards in experimental gliders designed by Sir George Cayley, a pioneer generally recognized as the "Father of Aerodynamics" as well as the individual who first postulated the modern airplane configuration (wings, fuselage, and a tail group).[2] Then, of course, comes that towering figure of early flight testing, Otto Lilienthal. In 1896, the year of his death in a glider accident, he wrote, "One can get a proper insight into the practice of flying only by actual flying experiments." A trained mechanical engineer, he combined shrewd theoretical studies of birdflight with his own bold experiments with a series of monoplane and biplane gliders. He recognized the price flight researchers are sometimes required to pay, remarking that "Sacrifices must be made," an especially appropriate and poignant epitaph for his own career.[3] Lilienthal profoundly influenced subsequent researchers, notably Octave Chanute, and Wilbur and Orville Wright.

The Wright brothers deserve credit for developing the first powered and manned aircraft capable of making a sustained and controlled flight. They were brilliant, intuitive flight researchers who recognized the vital partnership between theory, ground testing, and research aloft, and the need for acquiring reliable data. Wilbur Wright compared the testing of an airplane to riding a fractious horse. He stated, "If you are looking for perfect safety you will do well to sit on a fence and watch the birds, but if you really wish to learn you must mount a machine and become acquainted with its tricks by actual trial."[4] Beginning with theoretical studies, the brothers moved to kite-gliders. When early designs proved disappointing, they refined their thinking and improved their understanding by ground testing including use of a small wind tunnel. Then, with their confidence bolstered by flight trials with the 1902-1903 glider, they built the epochal 1903 powered machine. On December 17, 1903, the era of powered flight dawned, a trimph of flight research that fulfilled the dream of centuries. Orville, the test pilot, summarized the flight as follows:[5]

> "Wilbur ran at the side, holding the wings to balance it on the track. The machine, facing a 27-mile wind, started very slowly. Wilbur was able to stay with it until it lifted from the track after a forty-foot run. The course of the flight up and down was exceedingly erratic. The control of the front rudder [canard elevator-ed.] was difficult. As a result, the machine would rise suddenly to about ten feet and then as suddenly dart for the ground. A sudden dart when a little over 120 feet from the point at which it rose into the air, ended the flight. The flight lasted only twelve seconds, but it was nevertheless the first in the history of the world in which a machine carrying a man had raised itself by its own power into the air in full flight, had sailed forward without reduction of speed, and had finally landed at a point as high as that from which it started."

For its time, this is a model test flight report. It presents the test conditions, a critical analysis of the airplane's stability and control, and, finally, a summation of the flight's significance. Nowadays, of course, such information is presented accompanied by extensive quantitative analysis, but the ideas behind the report are the same. One can compare the very successful and practical approach to flight research taken by the Wrights (and, to a lesser extent, by Lilienthal and Chanute) to the overblown methods of a Samuel Langley or Hiram Maxim, who went to elaborate lengths on paper and with costly testbeds to develop what were ultimately grotesque and unsuccessful vehicles.

By the time of the First World War, flight testing had already taken on the trappings of professionalization. Designers, pilots, and engineers worked closely together, and emphasis shifted from choosing just good "stick-and-rudder" men as test pilots to choosing good stick-and-rudder men with solid technical credentials. In Great Britain, Edward Busk had introduced scientific methods to flight testing at Farnborough. [6] After Busk's tragic death in an aircraft accident, Frederick Lindemann, William Farren, and Henry Tizard, trained scientists all, continued this trend. Lindemann--who eventually became Prime Minister Winston Churchill's scientific advisor during the Second World War--conducted a major experimental study of spinning, complementing a theoretical analysis of the problem that he had already undertaken.[7] This "scientific" influence extended across the Atlantic as well, to early American test pilots such as Edmund T. "Eddie" Allen (who flew for the U.S. Army at Britain's Martlesham Heath testing center), and Thomas Carroll of the National Advisory Committee for Aeronautics (NACA). The NACA, created in 1915 by an act of Congress "To supervise and direct the scientific study of the problems of flight," did much to place American flight testing on a firm scientific basis. The NACA began its flight research activities in 1919 at the Langley Memorial Aeronautical Laboratory (now NASA Langley Research Center) using Curtiss JN-4H Jenny trainers.[8] Two years earlier, the American military services recognized the unique importance of flight testing by creating specialized test centers, beginning with the establishment of Anacostia Naval Air Station in 1917. The next year, 1918, the Army established McCook Field in Ohio as the Signal Corps' experimental laboratory. In fact, both military services had a heritage of flight research predating the creation of these two centers, but the creation of these centers marked an important milestone in the evolution of American military aeronautics. Anacostia remained the Navy's flight testing center until the establishment of the Naval Air Test Center at NAS Patuxent River in 1943. Likewise, McCook eventually gave way to Wright Field, and Wright, in turn, to Edwards Air Force Base, as the air service's premier flight testing center. These early centers were no less professional in their approach to flight testing and flight research than their successors are in the present day. For example, McCook pilots and engineers submitted detailed reports on new aircraft, with remarks on control forces, control effectiveness, stability, handling characteristics, and the efficiency of cockpit instrument displays.[9]

Professional flight researchers and aircraft designers recognized that the devil-may-care "show me the stick and I'll fly it" test pilot was a dangerous anachronism, who was disappearing in fact if not in fiction. In 1920, Edward P. Warner and F. H. Norton of the NACA wrote that "Test flying is a highly specialized branch of work, the difficulties of which are not generally appreciated, and there is no type of flying in which a difference between the abilities of pilots thoroughly competent in ordinary flying becomes more quickly apparent."[10] Warner, an individual who greatly influenced the subsequent history of American flight testing, stressed giving the test pilot training in analytical methods. During this time, a number of Federal and private organizations started issuing formal flight testing handbooks and instruction guides for prospective flight test crews. One such, by Army Captain George Patterson, quaintly warned that test crews should not use pens to record information, "as the ink will freeze at high altitudes."[11] In 1927, Lawrence V. Kerber (the former director of flight testing at Wright Field) and W. F. Gerhardt issued a pioneering guide, A Manual of Flight Test Procedure, leaving no doubt as to how exacting the flight testing process was becoming.[12]

During the 1920's, flight research and flight testing advanced rapidly, keeping pace with major changes affecting the development of aeronautical technology. Among the revolutions transforming aviation at this time were the streamline doctrine, which led to greater emphasis on aerodynamics and efficient high-speed high-altitude flight; the development of more powerful and lightweight engines, particularly the high-performance radial engine; and the transformation of the aircraft structure from wood to wood-and-metal, and eventually to all-metal. Flight researchers examined problem areas relating to all of these subjects, and many others as well. They studied how aircraft behaved in accelerated flight, flew to increasingly higher altitudes, evaluated new devices for aircraft, made the first landings on sea-going aircraft carriers, and developed new flying techniques and design criteria. In March 1924, for example, Army test pilot James H. "Jimmy" Doolittle test-flew a Fokker PW-7 experimental fighter biplane so that aeronautical engineers could better design future pursuit biplanes to withstand the forces of high-speed abrupt maneuvers. During one 7.8g dive pullout, the rear wooden surface of the Fokker's upper wing cracked, indicating that the PW-7 had reached its limit load. This, incidentally, refuted the then-commonly held belief that wings fail in a pullout by shearing backwards towards the tail.[13] Doolittle himself is a good example of the engineering test pilot. He was not an unthinking daredevil but, rather, a shrewdly calculating professional who eventually earned an M.S. and Sc.D. in aeronautical engineering (based upon his flight testing studies) from the Massachusetts Institute of Technology. Meticulous and thorough test reports became a Doolittle hallmark, and a model for future test pilots to follow.[14]

Flight researchers examined new aircraft equipment and design theories by testing such developments on aircraft in actual flight. The variable-pitch propeller, the Handley Page wing flap and slat, the smaller ethylene glycol (vs. water) radiator, and the exhaust driven turbosupercharger were all thus experimentally verified.[15] In 1928, NACA engineers tested an experimental radial engine cowling on a modified Curtiss AT-5A Hawk. As designed, the uncowled Hawk had a maximum speed of 118 mph. With the cowling, its speed jumped to 137 mph (equivalent to the addition of 83 hp). Such results could not fail to impress the aircraft industry, and the "NACA cowling" soon became a standard installation on radial engine aircraft.[16] The NACA also pioneered in instrumenting aircraft to record flight conditions. This early work eventually led to a document of major significance, NACA Report 369, by C. H. Dearborn and H. W. Kirschbaum, entitled "Maneuverability Investigation of the F6C-3 Airplane with Special Flight Instruments." This study, which the NACA completed in 1930, was the first detailed examination of aircraft handling qualities ever done in the United States. Onboard recording instrumentation had provided a precise record of the airplane's behavior during loops, pullups, pushdowns, and abrupt rudder maneuvers. The results were then reduced to easily understood data that aircraft designers around the world could put to use in engineering new aircraft.[17] The Daniel Guggenheim Fund for the Promotion of Aeronautics made a number of significant contributions to aeronautical education, technology, and flight safety during its short four-year existence from 1926 to 1930. In particular, two of its more important activities involved flight testing and flight research in the fields of so-called "blind" (instrument) flying and short-takeoff-and-landing (STOL) aircraft design.[18] The Fund created a special "Full Flight Laboratory" at Mitchel Field, Long Island, in conjunction with the Army Air Corps and the Bureau of Standards (which was deeply committed to studying the technology of radio navigation and blind landing aids). On September 24, 1929, Fund test pilot Jimmy Doolittle (on loan from the Army) completed the world's first blind flight from takeoff to landing, using three new aviation instruments developed at the behest of the Fund: the Kollsman precision altimeter, the Sperry gyrocompass, and the Sperry artificial horizon.[19] In 1930, the experimental Curtiss Tanager STOL biplane won the Guggenheim International Safe Aircraft Competition (a British design, the Handley Page Gugnunc, finished a very close second), in a competition notable for the wide and exciting diversity of technological approaches taken to achieve the goal of a truly STOL aircraft. Some of these approaches included variable incidence and variable camber wings, wing slats or slots, flaps, spoilers, long-stroke landing gears, adjustable horizontal stabilizers, and so-called "floating" ailerons. The information acquired during this noteworthy competition provided valuable design criteria for subsequent STOL aircraft.[20]

The 1930's witnessed an expansion of work undertaken in the 'twenties, notably in the areas of high-altitude flight (eventually marked by notable developments in pilot protective suits, pressure cabins, and the turbojet engine), rotary-wing research (particularly the transition from the autogiro to the genuine helicopter), and in the increasing professionalization of the test pilot. To further the desires of aircraft designers to develop fast and efficient long-range high-altitude aircraft, various aeronautical research establishments around the world supported extensive studies of the upper atmosphere. Consequently, engineers and inventors sought ways to provide pilots and flight crews with adequate physical protection. This experimentation took the form of high-altitude balloon flights (some marred by tragedy), and experiments with pressure suits and pressure cabins. In the United States, Wiley Post climaxed a long pressure suit development program in March 1935 by flying a modified Lockheed Vega monoplane, the Winnie Mae, at over 30,000 feet from Burbank, California, to Cleveland, Ohio. Post's experimental full-pressure suit, though awkward and uncomfortable, worked very well, and can be considered the ancestor of the modern full-pressure spacesuit.[21] Later that year, Army pilots Albert Stevens and Orvil Anderson reached 72,395 feet while piloting the balloon Explorer II.[22] Finally, in 1937, the Army flew the XC-35, a derivative of the twin-engine Lockheed Electra transport, equipped with turbosupercharged engines and a pressurized cabin. During extensive testing at Wright Field, the XC-35 validated the pressure cabin concept, anticipating all modern pressurized civilian and military aircraft.[23] By the end of the decade, the world's first airliner designed for pressurized operation, the Boeing Model 307 Stratoliner, was already flying. Flight research had produced results enabling aircraft designers to build high altitude aircraft with confidence. Subsequent research, particularly by then-Colonel W. Randolph Lovelace of the Aero-Medical Laboratory at Wright Field and the Boeing company, confirmed the practicality of developing aircraft and training aircrews to operate routinely above 30,000 feet. This work had an important impact on post-World War II commercial aircraft operations in addition to its obvious military significance.[24]

Development of rotary-wing aircraft made major strides in the 1930's, building upon earlier development of the autogiro by Spanish engineer Juan de la Cierva in the 1920's. With its unpowered rotor system, the autogiro could not accomplish the true vertical takeoff and landings possible with the genuine helicopter, and thus, despite its remarkable STOL performance, remained more an indication of what still needed to be done than as a fulfillment of promise itself. Rudimentary helicopters appeared during the 1930's, typified by the coaxial Gyroplane Laboratoire of Louis Bréguet and René Dorand and the twin-rotor Focke-Achgelis Fw 61 of Heinrich Focke, but it remained for expatriate Russian designer Igor Sikorsky to develop the helicopter into a practical reality, beginning with his VS-300 testbed. The VS-300 made its first tethered ascent in 1939, and months of thorough exploratory and developmental flight testing were required before it completed its first successful free flight on May 13, 1940. Further experimentation to improve its controllability resulted in Sikorsky developing a totally satisfactory control system that subsequently appeared on the Sikorsky R-4, the world's first production helicopter. Once again, flight testing had refined a good concept into a workable production system.[25] Complementing this developmental work were analytic studies by the NACA using a variety of early autogiros and helicopters. The NACA had begun its rotary-wing flight testing research with a Pitcairn PCA-2 purchased in 1931, averaging two flights per week over the next five years. The NACA trained a coterie of Army personnel in rotary-wing flight testing methods who then returned to the service and established an acceptance testing program, freeing the NACA to concentrate on research. Subsequent NACA work emphasized research on controllability, rigid rotors, blade motion, and rotor dynamics and loads. NACA's work in this field accelerated the development of more sophisticated helicopters after the Second World War, and presaged the extensive work undertaken by the NASA on helicopters and prop-rotor V/STOL technology, perhaps best exemplified by the recent Bell XV-15 tilt-rotor program.[26]

A series of sensational motion pictures caused the 1930's public to regard the test pilot as a short-lived wild character with no concern for personal safety, who had overdeveloped qualities of foolhardiness and underdeveloped qualities of common sense, what professional flight testers scorned as a "high guts to brains ratio." The worst and most influential of these films was Test Pilot, starring Clark Gable, Spencer Tracy, and Myrna Loy. This unfair and demeaning portrait enjoyed then--and still enjoys, to some extent--wide acceptance. Serious test pilots, such as Eddie Allen, wrote refutations that never quite succeeded in catching up to the myth. Commenting on the demise of the "here goes nothing" approach to test flying, Allen stated that "Under the changing conditions of increasing knowledge, fatalistic risk taking becomes ignorant recklessness."[27] Unfortunately, occasionally a test pilot did take a foolish chance, and endow the myth with the trappings of veracity. Contract test pilot Jimmy Collins had popularized the myth in his own best-selling book Test Pilot, and tragically fulfilled it shortly thereafter by crashing the experimental Grumman XF3F-1 biplane fighter during a rash and ill-judged dive pullout that overstressed the airplane. Such actions demonstrated only too well that a test pilot must be thoroughly acquainted with the potential dangers stemming from his actions.[28] By the early 1940's, the contract test pilot, a free-lancer who flew for a variety of companies, was increasingly an anachronism, though some of these individuals--such as Eddie Allen himself, and Vance Breese--were truly outstanding airmen. The future belonged to the careerists--the test pilots who flew for the government or for private industry. Increasingly, then, a need arose to train and furnish such men to the aeronautical community, endowing them with standardized training and strong technical backgrounds. Out of this need, and particularly from the urgency of Second World War demands, emerged the first test pilot schools. Great Britain created the Empire Test Pilots' School (ETPS), where prospective test pilots could receive a thorough grounding in flight test procedures. In the United States, the Navy followed with creation of the Naval Test Pilot School at Patuxent River, and the Army Air Forces started a similar school at Vandalia Airport, near Wright Field. After the Second World War, the USAF moved this latter school to Edwards Air Force Base. (Other such schools followed, notably in France, and recently a new civilian school, the National Test Pilot School, has begun operations at Mojave, California, in the midst of the Antelope Valley's flight testing nexus). Neatly fitting into this manifestation of interest in ensuring the professional standards of flight test pilots was the work of the NACA. Late in the 1930's, Hartley A. Soulé and Robert R. Gilruth of the Langley laboratory began an extensive investigation aimed at deriving a standard set of guidelines within the field of aircraft stability and control so that test pilots, flight test engineers, and designers would all speak a common "language." This resulted in the issuance of a landmark 1941 report (NACA Report 755) by Gilruth entitled "Requirements for Satisfactory Flying Qualities of Airplanes," issued in 1943. After the Second World War, the NACA continued its efforts to define standard pilot rating criteria. Test pilot George E. Cooper of the NACA Ames Aeronautical Laboratory derived a ten point pilot opinion scale which (in its original form) rated aircraft performance as "Satisfactory" (1-3), "Unsatisfactory" (4-6), "Unacceptable" (7-9), and "Unprintable" (10). In expanded and refined form, of course, this became the justly famed Cooper-Harper scale, used world-wide for the evaluation of new aircraft.[29]

Aside from service testing of military airplanes for wartime duty, the major challenges of aeronautical development in the 1940's were building upon the turbojet revolution and "breaking" the so-called sound barrier. The turbojet revolution was the product not of the aero-propulsion community but, rather, from individual inventors who built small demonstrator powerplants and flew them in rudimentary research airplanes. Only after demonstrating the potentialities of the gas turbine in such fashion were these inventors and their backers able to convince unenthusiastic engine manufacturers and governments to support further turbojet development. The two major figures in gas turbine research were the British test pilot and engineer Frank Whittle, and the German physicist Hans von Ohain. Shrewdly aligning himself with German industrialist Ernst Heinkel, von Ohain was able to undertake development of the Heinkel He 178 technology demonstrator, which completed the world's first jet flight on August 27, 1939.[30] Whittle's work resulted in the Gloster E.28/39, which flew in May 1941.[31] Surprisingly, in light of outstanding early turbosupercharger work, the United States came in third in the jet engine race, behind Germany and England. On October 1, 1942, Bell test pilot Robert M. Stanley ushered in the American jet era with a brief flight in the experimental Bell XP-59A Airacomet. So secret was this program (essentially a blending of an American airframe with Whittle engine technology imported from Great Britain) that all tests were conducted in the remote, barren surroundings of Muroc and Harper Dry Lakes. Indeed, at one point, security personnel disguised the plane with a bogus propeller.[32] The XP-59A soon gave way to the Lockheed XP-80 Shooting Star, first tested at Muroc in early 1944. This latter aircraft, of course, eventually spawned one of the most successful families of jet airplanes, the P/F-80 fighter, T-33 trainer, and F-94 interceptor family. Development of the P-80 occurred too late to permit its introduction into combat, though both Germany and Great Britain placed jet aircraft into operational service (the Me 262, Ar 234, and He 162, and the British Gloster Meteor) before war's end.[33] The U.S. Navy also pursued jet development, and on July 20, 1946, Lt. Cmdr. James Davidson landed the prototype McDonnell XFD-1 Phantom, a twin-jet design, aboard the carrier U.S.S. Franklin D. Roosevelt, sending the Navy into the jet age.[34] With 500+ mph jet speeds, the time available to pilots to make critical decisions decreased markedly, and flight test personnel, accustomed to testing piston-powered aircraft, had to institute special procedures for use with jets.[35] Bell engineer Benson Hamlin prepared the first guide to gas turbine aircraft testing, Flight Testing Conventional and Jet Propelled Airplanes, in 1946.[36]

The inability of 1940's wind tunnels to furnish reliable transonic aerodynamic information, together with the well-publicised loss of several test aircraft from so-called "compressibility" (most notably Ralph Virden in a Lockheed P-38), led the NACA and the military services (in conjunction with private industry) to undertake joint transonic and supersonic research aircraft development programs, generating the famed postwar "X-series" of experimental airplanes. The plane itself now became a unique research tool, using the sky as a laboratory. These American efforts mirrored equivalent efforts abroad, notably by Great Britain and Germany. Along the way, various stop-gap methods of research were attempted, particularly use of modified bombers to drop falling body aerodynamic shapes, rocket-propelled models, and experiments with diving fighters. One of the most interesting interim test methods involved the so-called "wing flow" method of research, using a small model mounted on a balance mechanism installed in the gun bay of a modified P-51 Mustang. NACA pilots would dive the Mustang to over Mach 0.7, and the behavior of the small model in the resulting accelerated transonic flow would be recorded by onboard instrumentation for subsequent analysis. American researchers dived P-38, P-47, and P-51 fighters as high as Mach 0.82 [37], and German investigators took Bf 109, FW 190, Me 163, and Me 262 fighters to as high as Mach 0.85. [38] Test pilot A. F. Martindale of the Royal Aircraft Establishment, however, achieved Mach 0.9 (±.01) in a Spitfire Mk. XI during carefully conducted flight testing at Farnborough in 1943. Starting at an altitude of 40,000 feet, he attained an airspeed of 610 mph at 29,000 feet before initiating a 2.2g pullout.[39] All of this work, with models, falling bodies, and diving fighters, encouraged proponents of piloted research aircraft that could undertake research missions in level flight, without the time constraints and associated hazards present when diving towards the earth in a buffeting and marginally controllable aircraft.

The evident great interest of German aerodynamicists in high-speed flight planforms such as the swept, delta, and tailless configurations vindicated the work of Allied investigators who had studied such designs, and also stimulated postwar development of new ones. But the problem of transonic flight remained. Geoffrey de Havilland Jr. perished in the crash of the tailless De Havilland D.H. 108 Swallow research airplane when it broke up in the midst of a violent longitudinal pitching oscillation at approximately Mach 0.87; his death coincided with a British governmental decision to abandon construction of specialized transonic research aircraft, notably the Miles M.52, on grounds of safety and economy.[40] The first manned supersonic flight occurred on October 14, 1947, when Air Force test pilot Charles E. "Chuck" Yeager reached Mach 1.06 (approximately 700 mph) at 43,000 feet, over the Mojave Desert near Muroc, flying the rocket-propelled and air-launched Bell XS-1. The significance of this accomplishment, considered at the time the most important flight since that of the Wrights at Kitty Hawk, cannot be overemphasized. Aviation science had crossed the invisible threshold to flight faster than sound, and the notion of a "sound barrier" crumbled into ruin. The test pilot subsequently wrote that: [41]

> "With the stabilizer setting at 2° the speed was allowed to increase to approximately .98 to .99 Mach number where elevator and rudder effectiveness were retained and the airplane seemed to smooth out to normal flying characteristics. This development lent added confidence and the airplane was allowed to continue to accelerate until an indication of 1.02 on the cockpit Mach meter was obtained. At this indication the meter momentarily stopped and then jumped to 1.06 and this hesitation was assumed to be caused by the effect of shock waves on the static source. At this time the power units [the four-chamber XLR-11 rocket engine-ed.] were cut and the airplane allowed to decelerate back to the subsonic flight condition. When decelerating through approximately .98 Mach number a single sharp impulse was experienced which can best be described by comparing it to a sharp turbulence bump."

Yeager's matter-of-fact report belied the very real drama that had attended the flight.

Companies now set out to exploit the supersonic breakthrough, and many new configurations underwent flight verification before being applied to new supersonic aircraft. Researchers examined the sweptwing concept with the low-speed Bell L-39 testbed before placing the North American F-86A Sabre into production, and later used sweptwings on the high-speed Douglas D-558-2 Skyrocket (the first Mach 2 airplane, in 1953) and the ill-fated Bell X-2 (the first Mach 3 airplane, in 1956). They explored and rejected the semi-tailless configuration with the Northrop X-4 (which manifested the same kind of disturbing stability and controllability problems encountered by the earlier Messerschmitt Me 163 and the tragic D.H. 108, though fortunately not to the same disastrous degree). The variable sweep wing underwent proof of concept testing with the Bell X-5. The thin low-aspect-ratio wing first flew on the Douglas X-3 and encouraged Lockheed to pursue a similar planform with the XF-104 Starfighter, first flown in 1954. The delta wing first flew on the Convair XF-92A testbed of 1948, inspiring the subsequent F-102/F-106/B-58 family of aircraft. Foreign delta research aircraft included the Avro 707 and Fairey F.D.2. The French pioneered ramjet aircraft studies with the Leduc 010 of 1949, and the Mach 2+ Nord Griffon. But flight research continued to extract a heavy price for the technical gains. All three D.H.108's crashed, killing their pilots. Three X-1's, two X-2's, one X-5, and one D-558-1 Skystreak were lost, killing five airmen. The Air Force renamed Muroc as Edwards AFB after the loss of a Northrop YB-49A Flying Wing and its pilot, Capt. Glen Edwards. The YB-49A Flying Wing represented an elegant attempt to achieve what has been one of flight's oldest and most seductive visions--a pure streamline all-wing craft unencumbered by the addition of tail surfaces or fuselage. Yet there are also elements of pathos as well, for the Flying Wing was, at the time, too challenging a concept for the state of aviation technology. The all-jet YB-49A represented a poor compromise: an aircraft originally designed as a piston-engine propeller-driven bomber (the XB-35) hastily modified as a jet aircraft, and lacking the aerodynamic design and structure necessary to take advantage of the gas turbine's performance. More serious were longitudinal and directional instabilities, which posed major challenges at a time when the state of stability augmentation technology was nowhere near as sophisticated as it is today. Though futuristic in concept, the Northrop XB-35 and YB-49--with aging aerodynamics, and afflicted by abysmal stability problems--were simply the wrong aircraft at the wrong time. Today, with the revolutionary advances of electronic flight control technology and the development of sophisticated composite structures, the opportunity for a successful, practical long-range flying wing bomber has never looked brighter--as is evident with the ongoing development of the Northrop B-2, the Advanced Technology Bomber.[42]

By mid-century, flight testing and flight research had firmly established itself in aeronautical science, and research centers such as America's Edwards, Great Britain's Boscombe Down, or France's Istres were known world-wide; the very mention of the names implied uncompromising standards, exactitude, and precision. Graduates of the test pilot schools located at these centers served governments and industry around the world, examining new concepts and problems. (The pilots recognized their professionalism by forming, in 1955, The Society of Experimental Test Pilots; SETP membership likewise has spread world-wide).[43] Significantly, the flight testing process had become a tightly structured one following precise methodology. An examination of flight test methods in use by the United States Air Force from 1947 through the 1970's offers an opportunity to see how changes in technology, requirements, and test philosophy resulted in evolutionary changes in the methodology of flight testing itself.[44]

Phase Testing (Unofficial), 1947-1951

Phase	Purpose	Tester	Aircraft
I	Basic Airworthiness	Contractor	Prototype
II	Verify Contractual Guarantees (Contractor Compliance)	Air Force	Prototype
III	Correct Deficiences Noted in II (Design Refinement)	Contractor	Pre-Production
IV	Operational Testing	Air Force	Production

Examples: F-84 (straight-wing), F-86, B-47

Phase Testing (Official), 1951-1958

Phase	Purpose	Tester	Aircraft
I	Basic Airworthiness	Contractor	Prototype
II	Contractor Compliance	Air Force	Prototype
III	Design Refinement	Contractor	Pre-Production
IV	Performance and Stability	Air Force	Production
V	All-Weather Operation	Air Force	Production
VI	Functional Development (to correct previously undiscovered deficiences)	Air Force	Production
VII	Operational Suitability	Air Force	Production
VIII (added 1956)	Unit Operational Employment Testing (at initial operating base)	Air Force	Production

Examples: F-84 (sweptwing), F-100, F-104, B-52, KC-135

Category Testing, 1958-1972

Category	Purpose	Tester	Comments
I	Subsystem Development Test & Evaluation	Primarily Contractor but some USAF also	First use of JTF
II	System Development Test & Evaluation	Mixed Contractor and USAF testing	Gradually increasing USAF control
III	System Operation Test & Evaluation	Primarily USAF; grad. decrease of contractor role	OT&E under control of using command

Examples: F-105, F-106, C-5, F-111

Each of these particular systems had weaknesses or characteristics that resulted in their being replaced by subsequent ones. Early unofficial phase testing--unofficial in that it was not governed by any official document or directive--occasionally resulted in the Air Force receiving aircraft that were not operationally suitable. "Official" phase testing attempted to rectify this, but without success. Instead, the process grew increasingly long, and operational suitability issues remained--chiefly because the time period that this process operated in coincided with the rapid proliferation of aircraft subsystems such as search radars and fire control systems. For example, nearly half of the Convair F-102A Delta Dagger interceptors delivered to the Air Force lacked functioning fire control systems, necessitating time-and-cost-consuming retrofit.[44] Category testing likewise attempted to address these problems, introducing the concept of the Joint Test Force. The "jointness" concept worked well, persisting to the present day, but some problems still remained, such as duplicative testing, insufficient Air Force involvement in the Category I testing (and, hence, little "heads up" warning of potentially serious deficiencies), and an unsatisfactory Category III test process that, in some cases, tended to actually delay and hinder the introduction of an aircraft into operational use. The "concurrency" philosophy of the 1960's likewise impacted upon successful category testing, resulted in production of aircraft having deficiences that had to be worked out or lived with by the using command. The B-58, C-5, C-141, and (above all) the F-111 are classic examples of aircraft that encountered serious delays and difficulties because of weaknesses in the category testing process.[44] Following a Presidential "Blue Ribbon" defense study in 1970 that acknowledged the generally satisfactory nature of developmental testing but the persistent problems with operational test and evaluation, the Air Force dropped category testing in mid-1972, replacing it with the notion of DT&E/OT&E that, in a general sense, persists to the present day; time phasing became less significant, and passing critical milestone points assumed much greater significance. Continued interest in operational testing by a structure separate from both the R&D and user commands led to the creation of the Air Force Test and Evaluation Center (AFTEC) at the beginning of 1974. These developments occurred concurrently with a general return to the "fly-before-buy" acquisition philosopher that had previously governed military aircraft development, but which had been dispensed with in the 1960's. The return to competitive fly-offs was signaled by the AX competition (won by the Fairchild A-10 as opposed to the Northrop A-9), and the Lightweight Fighter (YF-16 vs. YF-17) competition.[44]

The 1950's were a particularly fruitful period in flight testing, as evidenced by the profusion of testbed and demonstrator aircraft developed world-wide for a variety of purposes. They offered engineers and designers the chance to experimentally verify or refute numerous ideas such as variable wing sweeping (with the X-5 and Grumman XF10F-1), tail-sitting VTOL aircraft (the Convair XFY-1 and Ryan X-13), the Whitcomb area-rule (with the YF-102A and Grumman F11F-1 Tiger), rubber-mat landing pads, and other schemes. Flight testing unveiled new problems, such as sweptwing pitchup (discovered on the Douglas D-558-2 Skyrocket), and coupled motion instability (revealed by the Douglas X-3 and the North American F-100A Super Sabre). Coupled motion instability led to the loss of the Bell X-2 at Mach 3.196 on September 27, 1956, and the death of test pilot Milburn Apt. Earlier in the month, Iven Kincheloe had flown the plane to an altitude of 126,200 feet, the first near-space manned flight, revealing the difficulties of flying a plane with conventional control surfaces in a low dynamic pressure environment, and helping spur the development of reaction control thruster for the subsequent hypersonic North American X-15.[45]

The North American X-15 program constituted a milestone effort on the road to winged lifting reentry from space. It also forced major advances in high-speed flight technology and flight test techniques and facilities. In January 1945, an experimental winged Nazi A-4b missile (essentially a V-2 with a low aspect ratio sweptwing) had been successfully boosted into the upper atmosphere, transitioning from a ballistic flight path to a Mach 4 supersonic glide before experiencing structural failure and breaking up. This was the fastest flight by a winged vehicle until the advent of X-15 testing in 1959.[46] The A-4b experiment followed on the heels of speculative work by the team of Eugen Sänger and Irene Sänger-Bredt regarding the possibility of long-range hypersonic flight involving global and even orbital operations.[47] These earlier efforts, as well as the baseline of experience from the "Round One" rocket research airplanes (the X-1, X-2, and Douglas D-558-2 families), all influenced the subsequent development of the X-15. Three of the X-15 aircraft were built, designed for air-launching from a modified Boeing B-52 mothership. The X-15's were the first aircraft designed to have a structure capable of withstanding the thermal gradients experienced in hypersonic flight, as well as the first to be propelled by throttleable "man-rated" liquid-rocket engines. They were true "aero-space" craft, and required a complex flight control system consisting of aerodynamic control surfaces (including a rolling tail and cruciform tail configuration), and reaction control thrusters. The pilot had no less than three control systems: a center stick for approach and landing, a sidestick for high speed flight, and a reaction controller. Likewise, the X-15 required that the pilot have the benefit for full-time full-pressure protection, necessitating the development of a genuine spacesuit by the David M. Clark company; this suit subsequently greatly influenced the development of American suits used in the space program. It also required creating a special 485 mile test corridor with three tracking stations located at Ely and Beatty, Nevada, and at Edwards. Nothing this extensive had previously existed for flight research, and it foreshadowed the worldwide tracking network developed for the later manned spacecraft program.[48] Thus, the three X-15 aircraft bridged the gap between flight within the atmosphere and flight into space. A modified X-15, the X-15A-2 (which was slightly lengthened and had provisions for jettisonable external fuel tanks) reached Mach 6.72 (4,520 mph), while another attained an altitude of 354,200 feet (67 miles). The information output of this highly successful and ambitious flight research program boiled down to 700 technical documents, equivalent to the output of a typical 4,000 person Federal research center for more than two years. [49] Unfortunately, the loss of test pilot Michael Adams in the crash of the third X-15 in late 1967 put a high price on this effort. The X-15 flight research program consisted of two phases: an aerodynamic and structural heating investigation phase and, increasingly after 1963, a follow-on program whereby the X-15 aircraft were utilized to carry experiments into the upper atmosphere or to high Mach. Through this applications research program, the X-15 aided space science research and, in particular, the Apollo program. This ten-year effort, which lasted from 1959 to 1969 and involved 199 research flights, constituted the most productive and exciting research aircraft program.[50]

The X-15 did not constitute the only noteworthy effort in the high speed flight domain during the late 1950's and into the 1960's. Of particular significance were the Lockheed Blackbird, North American XB-70A, and Anglo-French Concorde SST development efforts. Under conditions of great secrecy, Lockheed test crews evaluated a radical Mach 3+/80,000 ft.+ supersonic aircraft, the blended wing-body A-12, which served as the basis for the subsequent YF-12A experimental interceptor and the SR-71A strategic reconnaissance aircraft. The Mach 3+ A-12 posed tremendous challenges in both design and in flight testing; designer "Kelly" Johnson remarked subsequently that virtually everything about the titanium aircraft "had to be invented from scratch."[51] Flight testing such a craft of necessity occupied large blocks of airspace; some flights covered the whole southern United States, from California to Florida and back.[52] Though some aircraft and crews were lost, by and large the flight testing program went smoothly, paving the way for introduction of the SR-71 into service in the mid-1960's. The Blackbird subsequently went on to a long and distinguished career as a national security asset. NASA, in conjunction with the Air Force, undertook a comprehensive supersonic investigation program with two YF-12's from 1969 through 1979, materially aiding understanding of the problems of large sustained supersonic aircraft, particularly those of structural loading due to thermodynamic effects, aerodynamic and propulsion interactions, inlet unstart, flight path management, and development of adequate stability augmentation systems and integrated propulsion and flight controls.[53] In contrast to the YF-12/SR-71 effort, the experience with the gigantic North American XB-70A Valkyrie proved less sanguine. This shapely canard delta represented a very different design approach than the seductive Blackbird. Like the Blackbird, the XB-70A was a Mach 3 design, though it rarely actually attained this speed in part because of difficulties with its bonded stainless steel honeycomb structure shedding external skin. Operational problems with the landing gear, hydraulic system, and engines posed continuing headaches, and the potentially most useful of the two prototypes was lost in a well-publicized collision with a chase plane during a publicity flight. Though the XB-70A furnished some useful information on the problems and characteristics of large supersonic aircraft, it was not as useful as the ever-productive Blackbirds.[54] The Anglo-French Concorde, on the other hand, was a remarkably successful program in terms of its research and development, though airline reluctance to purchase and operate supersonic transports limited the type's service to Britain's and France's national flag-carriers. The most elegant and shapely aircraft ever flown, the Concorde built logically upon earlier flight research experience with the Griffon, Mirage, and F.D.2 families. Beginning in 1969, the prototypes of the Concorde, each equipped with twelve tons of recording instrumentation, begain a cautious and incremental flight test program leading to Mach 2. They passed Mach 1 seven months after the type's first flight, and Mach 2 over a year later. The two prototypes and five production aircraft assigned to te test program accumulated nearly 4,000 hours of flight testing, making the Concorde the most exhaustively tested jetliner by the time of its introduction into scheduled service in 1975.[55] At the complete opposite end of the performance spectrum, of course, were the numerous STOL and V/STOL testbeds and demonstrators flown during the 1960's, such as the Bell X-14B, Curtiss-Wright X-19, Bell X-22, and LTV XC-142A, various rigid rotor and variable-stability helicopters, and the vectored-thrust P.1127 and its successor the Hawker Kestrel (from which evolved the combat-proven Harrier). While all of these demonstrated the basic feasibility of V/STOL flight, only the P.1127 proved worthy of further development. The kind of capability now sought in the Bell V-22 program (and presaged by flight testing with the XV-15 in the early 1980's) required a technology base not yet available to researchers in the 1960's.[56]

There were, of course, numerous other developments and concepts evaluated in the 1960's and 1970's, such as advanced laminar-flow research (with the much-heralded but disappointing Northrop X-21A), the supercritical wing (evaluated on modified F-8 and F-111 testbeds in joint NASA-Air Force programs), and electronic flight control systems (with modified B-47, F-4, F-8, Hawker Hunter, and F-104 testbeds). While the supercritical wing work greatly benefitted the commercial air transport market, that of the electronic flight control revolution benefitted the development of a new generation of advanced military aircraft, typified by the F-16. The "Electric Jet" established new standards of agility and maneuverability for fighter aircraft, but also exhibited numerous teething difficulties caused by its radical flight control system. That these problems were worked out (particularly those dealing with controllability during high angle-of-attack and departure conditions) is due to the dedicated work of the F-16 CTF, which labored long and hard to make the promise of this exciting airplane a reality. In hundreds of hours of exceptionally hazardous testing, the flight test crews of the F-16 CTF thoroughly proved out the aircraft without loss of a single plane or life--a far cry from the days of testing high-performance aircraft during the 1950's.[57] To address some of the risks inherent in the developmental testing of new designs such as the F-15 and F-16, researchers turned to a variety of other "tools." Variable-stability aircraft gave test teams greater flexibility in predicting aircraft performance before the critical "first flight." A particularly useful variable-stability aircraft has been the Calspan/Lockheed T-33, used in studies ranging from the X-15 to the F-15 Eagle.[58] Variable-stability trainers such as Calspan's modified Douglas A-26 and, subsequently, a modified Gates Learjet, have enabled student test pilots to fly numerous types of "different" aircraft during their training. Advances in testing methods, notably automated real-time telemetry analysis, permitted preliminary data evaluation while a test flight was in progress; researchers thus gained more information per flight, cutting test time and costs roughly in half. Grumman's Automated Telemetry System (ATS), one such early system, proved especially useful during the F-14A Tomcat testing program, producing time savings of 67%.[59] The RPV--the Remotely Piloted Vehicle--demonstrated its usefulness (but also its limitations) as a cheap low-risk unmanned method of testing new designs and aircraft systems. For example, NASA undertook a spin-test program using a scale RPV of the McDonnell-Douglas F-15 Eagle that lent encouragement and confidence to Air Force and contractor plans for manned F-15 prototype spin trials. However, the more sophisticated and challenging Rockwell HiMAT (Highly Maneuverable Aircraft Technology) proved much less successful. This control configured canard design had electronic flight and propulsion controls and a composite structure. It attempted to do far too much, essentially an effort to generate a ground-controlled test vehicle having virtually all the capabilities of a full-sized piloted research aircraft. Its disappointing results serve as a reminder that design must always be appropriate, bound up with the establishment and recognition of clear, realistic, and attainable research goals. Despite the disappointment of the HiMat experience, however, it is clear that the RPV (or, more specifically, the Remotely Piloted Research Vehicle--RPRV) is a flight testing tool that is here to stay, much as the military RPV, despite its critics, has managed to secure for itself a notable place in the skies over contemporary battlefields.[60]

By the time of the initiation of the first manned spacecraft ventures, flight testing had so ingrained itself into the aeronautical research and development consciousness that NASA, virtually without thinking, automatically selected its astronauts from the ranks of test pilots (endowing them with as much command and control capability over their vehicles as was practicable, and selecting key programmatic personnel for the Mercury, Gemini, and Apollo efforts from the ranks of proven flight test managers. Such an approach certainly paid off, as, for example, in the Apollo 13 mission, where the astronaut crew (reflecting their flight testing backgrounds) made critical life-or-death decisions without having to rely for their survival completely on mission control back on earth.[61] Test pilot/astronaut crews demonstrated their ability to deal with unusual circumstances and trying conditions on many of the space flights, notably on Gemini 8 (David Scott and Neil Armstrong)[62], and on the descent of the Lunar Module to Tranquillity Base on Apollo 11 (Armstrong and Edwin Aldrin)[63]. Dating to the work of Sänger-Bredt, there had always been a strong interest in flying a winged vehicle into space and then returning it through the atmosphere. The X-15 program, of course, made significant contributions to this hypersonic technology, but other significant work was undertaken as well, notably the attempt at developing the Boeing X-20 Dyna-Soar (for Dynamic Soaring) boost-glider, and a variety of tailored lifting body shapes. Dyna-Soar fell victim to the Kennedy Administration's disenchantment with winged spacecraft and a prediliction for a ballistic approach to spaceflight; at the time of its cancellation in December 1963 it was approximately 2½ years and $373 million away from its first flight. Despite termination, it was a generally useful technological exercise.[64] The technological void left by the cancellation of Dyna-Soar was filled, to a limited extent, by tests of pilotless reentry shapes, notably a winged "hot structure" ASSET (for Aerothermodynamic elastic Structural Systems Environmental Tests) vehicle, and an ablatively cooled blunt-lifting body spacecraft, the PRIME (for Precision Recovery Including Maneuvering Entry). The slender delta ASSET program consisted of 6 flights (one of which was marred by booster failure) down the Eastern Test Range, using Thor and Thor-Delta boosters. ASSET offered the fir practical experience the aerospace community had with an actual lifting reentry vehicle returning from space at near-orbital velocities.[65] PRIME involved firings over the Western Test Range using Atlas boosters. The Martin SV-5D PRIME vehicle was the first to demonstrate hypersonic maneuvering and significant cross-range excursions during an actual entry from space, and also inspired development of the piloted SV-5P (X-24A) low-speed lifting body demonstrator.[66] Both ASSET and PRIME were extraordinarily productive programs conducted "on the cheap" and in a national research environment that emphasized non-lifting or minimal lifting reentry vehicle development as opposed to the genuine lifting capability that they demonstrated.

This interest in piloted lifting reentry eventually spawned the national Space Shuttle program. The roots of the Space Shuttle include winged boost-glider concepts, concepts involving aerodynamically tailored lifting body shapes, and concepts for complex fully reusable combinations of these. While ground tests using hypersonic tunnels, shock tubes, ballistic ranges, and the like generally confirmed the acceptability of such designs for hypersonic flight, considerable doubt existed--particularly concerning lifting body shapes--whether such craft could transit from supersonic flight to a subsonic approach and landing. Accordingly, in the 1960's and 1970's, a family of "low-speed" (in the sense that they were limited to speeds below Mach 2) lifting bodies underwent flight test at Edwards in a joint NASA-Air Force research program. These consisted of the plywood NASA M2-F1, the "heavyweight" rocket-propelled NASA-Northrop M2-F2/3 and HL-10, and the Air Force-Martin X-24A/B. Each represented a variation upon the blunt lifting body theme. For example, the M2 series were modified half-cones, while the HL-10 and X-24A were modified fattened deltas. The X-24B, on the other hand, represented a completely different "flatiron" shape derived by the Air Force Flight Dynamics Laboratory and then "gloved" over the X-24A. These craft demonstrated that hypersonic lifting bodies could be developed that would have satisfactory supersonic and subsonic flying characteristics and handling qualities. In 1975, for example, the X-24B completed a series of flights involving rocket boost into the upper atmosphere, followed by a low L/D approach to a precision landing on the Edwards 15,000 ft. concrete runway, an important demonstration that a Shuttle craft could indeed complete a powerless descent to a precision recovery on a conventional runway.[67]

The subsequent development of the Space Shuttle did not proceed smoothly. Problems with its thermal protection system "tiles" and the Shuttle's liquid-fueled main engines delayed its first flight into orbit until April 1981, more than four years after the first Shuttle, the Enterprise, completed its approach and landing tests (ALT) at Edwards. The ALT program had proceeded rapidly, with the Shuttle being launched from a modified Boeing 747 carrier aircraft. (The Soviet Union has recently undertaken a smiliar series of flights of their own Shuttle design, from the back of a modified Myasischev Mya-4 Bison bomber). The Enterprise was flown "captive inactive" (without an astronaut crew) and "captive active" (with an astronaut crew) while mated to the 747, and then launched "tailcone on" (to give a higher L/D and longer descent time) and, finally, "tailcone off" (simulating the low L/D of a "mission configuration" Shuttle from orbit). The last tailcone-off flight included a landing on the Edwards 15,000 ft. runway, and during this flight the Shuttle manifested a potentially serious pilot-induced oscillation (PIO) aggravated by the inherent characteristics of its fly-by-wire flight control system. As a result, NASA undertook a brief but highly significant research program, modeling the Shuttle's control characteristics on a modified Vought F-8 Crusader (the NASA Digital Fly-by-Wire--DFBW--testbed).[68] Selected changes were incorporated as a result of this testing into the control system of the Shuttle. Shuttle's actual orbital flight tests confirmed the basic value of a Shuttle system, and the generally good design of the orbital transportation system itself. Shuttle's formal test program consisted of missions one (launched April 12, 1981) through four (launched June 27, 1982), but many areas of interest and concern remained after this time. For example, despite the large effort expended in wind tunnel testing--indeed, Shuttle was the most exhaustively tested aerospace vehicle developed to that time--discrepencies between actual and predicted results pointed yet again to the deficiencies of purely ground-based test facilities and predictive techniques. Brake failures and landing problems plagued flights, calling into question the wisdom of "routine" landings on the concrete runway at the Kennedy Space Center, where such a problem could be catastrophic. These and other considerations demonstrated that the Shuttle, far from being an "operational" system in the sense of an airliner or military aircraft (something its more ardent supporters had envisioned) was, as astronaut Vance Brand subsequently remarked, a system that "always will require special care and feeding." [69] NASA accumulated a vast wealth of experience and "lessons learned" on early Shuttle operations, assembling a major summary document of the program for use in developing other aerospace systems.[70] Then came the tragedy of the Challenger.

Challenger's loss will forever haunt the American space program, but it needs to be placed in the proper historical perspective. The loss of the Titanic did not signal the end of transatlantic oceanic voyages, and the loss of the early De Havilland D.H. 106 Comet jetliners did not result in the abandonment of the turbojet airliner. Technology, once invented and developed, cannot be "disinvented" however much critics might wish to do so. What Challenger's loss did illustrate, unfortunately, was the price paid for technological complacency and poor management.[71] Since the days of Lilienthal, in the late pre-history of flight, researchers--flight researchers in particular--have recognized that advances in capability are only rarely without cost. Writing after Challenger, former Gemini and Apollo astronaut Michael Collins stated that ". . . if someone had suggested to me in 1963, when I first became as astronaut, that for the next 23 years none of us would get killed riding a rocket, I would have said that person was a hopeless optimist, and naive beyond words."[72] It is this blend of pragmatism and managerial discipline that must be kept in mind as the aerospace community--worldwide--advances the frontiers of flight. Already new aerospace systems are contemplated--such as Europe's Hermes, Sänger II, and America's proposed X-30 National Aero-Space Plane (NASP) that will demand excellence of design and testing at a level not often reached in the past.[73] The record of past accomplishment--with systems such as the first turbojets, the first supersonic aircraft, the X-15, Concorde, the X-29 and, yes, the Shuttle--is a comforting one as these futuristic projects edge onwards.

It is obvious, then, that despite the tremendous advances that have been made in computational techniques, ground test facilities, and predictive methods, the need for exhaustive flight testing, together with the demand for professionally trained and highly motivated flight researchers, will always exist. In 1967, NASA Administrator James E. Webb testified before Congress that:[74]

> "Flight testing of new concepts, designs, and systems is fundamental to aeronautics. Laboratory data alone, and theories based on these data, cannot give all the important answers. . . . Each time a new aircraft flies, a 'moment of truth' arrives for the designer as he discovers whether a group of individually satisfactory elements add together to make a satisfactory whole or whether their unexpected interactions result in a major deficiency. Flight research plays the essential role in assuring that all the elements of an aircraft can be integrated into a satisfactory system."

Those words are as true for the future of aerospace as they have been for its past.

REFERENCES

1. Lynn White, "Eilmer of Malmesbury: An Eleventh Century Aviator," Technology and Culture, (Spring 1961), passim.

2. Constance Babington-Smith, Testing Time: The Story of British Test Pilots and Their Aircraft, New York: Harper & Brothers, 1961, p. 43.

3. Otto Lilienthal, "Practical Experiments," and "At Rhinow," in James Means, ed., Epitome of the Aeronautical Annual, Boston: W. B. Clarke Co., 1910, pp. 8, 32-33.

4. Wilbur Wright, address to the Western Society of Engineers, September 1901, in Charles H. Gibbs-Smith, Aviation: An Historical Survey From Its Origins to the End of World War II, London: Her Majesty's Stationary Office, 1970, pp. 222-223.

5. Orville Wright, "How We Made the First Flight," Flying and The Aero Club of America Bulletin,(Dec. 1913), passim.

6. Mary Busk, E. T. Busk: A Pioneer of Flight, With a Short Memoir of H. A. Busk, Flight Commander, RNAS, London: John Murray, 1925, passim.

7. Ronald W. Clark, Tizard, Cambridge, MA: MIT Press, 1965, passim.

8. Jerome C. Hunsaker, "Forty Years of Aeronautical Research," Smithsonian Report for 1955, Washington, D.C.: Smithsonian Institution, 1956, pp. 246-250.

9. Lois E. Walker and Shelby E. Wickam, From Huffman Prairie to the Moon: The History of Wright-Patterson Air Force Base, Washington, D.C.: GPO, 1986, pp. 140-141, 190-191.

10. Edward P. Warner and F. H. Norton, Preliminary Report on Free Flight Tests, NACA Report No. 70, 1919, p. 575.

11. George B. Patterson, Notes on Practial Airplane Performance Testing, Dayton, OH: Air Service Engineering Division, June 1919, pp. 29, 35-36.

12. W. F. Gerhardt and L. V. Kerber, A Manual of Flight-Test Procedure, Ann Arbor, MI: University of Michigan Department of Engineering Research, 1927.

13. James H. Doolittle, Accelerations in Flight, NACA Report No. 203, 1925, pp. 371-388.

14. Lowell Thomas and Edward Jablonski, Doolittle: A Biography, Garden City, N.Y. Doubleday & Company, 1976, passim.

15. Ronald Miller and David Sawers, The Technical Development of Modern Aviation, New York: Praeger, 1970, passim.

16. James R. Hanson, Engineer in Charge: A History of the Langley Aeronautical Laboratory, 1917-1958, Washington, D.C.: GPO, 1986, pp. 123-139.

17. C. H. Dearborn and H. W. Kirschbaum, Maneuverability Investigation of the F6C-3 Airplane with Special Flight Instruments, NACA Report No. 369, 1930.

18. Richard P. Hallion, Legacy of Flight: The Guggenheim Contribution to American Aviation, Seattle, WA: University of Washington Press, 1977, passim.

19. Daniel Guggenheim Fund for the Promotion of Aeronautics, Solving the Problem of Fog Flying: A Record of the Fund's Full-Flight Laboratory to Date, New York: Daniel Guggenheim Fund, 1929, passim.

20. Daniel Guggenheim Fund for the Promotion of Aeronautics, The Daniel Guggenheim International Safe Aircraft Competition: Final Report, New York: Daniel Guggenheim Fund, 1930, passim.

21. Stanley R. Mohler, and Bobby H. Johnson, Wiley Post, His Winnie Mae, and the World's First Pressure Suit, a volume in the Smithsonian Annals of Flight series, Washington, D.C.: Smithsonian Institution Press, 1971, passim.

22. Albert W. Stevens, "Man's Farthest Aloft," The National Geographic Magazine, (January 1936), passim.

23. XC-35 Aircraft Data File, Library of the National Air and Space Museum, Smithsonian Institution, Washington, D.C.

24. N. D. Showalter, "High Altitude Flight Test Procedures," Boeing High-Altitude Flight Symposium, August 25-27, 1947, Seattle, Washington, pp. 1-11.

25. H. F. Gregory, Anything a Horse Can Do: The Story of the Helicopter, New York: Reynal & Hitchcock, 1944, passim.

26. Frederic B. Gustafson, "History of NACA/NASA Rotating-Wing Aircraft Research, 1915-1970," a limited-edition reprint (VF-70) from VertiFlite Magazine, 1971, pp. 4-26.

27. Edmund T. Allen, "Here Goes Nothing: Flight Testing From Daredeviltry to Scientific Research," Aviation, (December 1937), passim.

28. Jimmy Collins, Test Pilot, Garden City, N.Y.: Doubleday, Doran & Co., 1935, p. 174.

29. George E. Cooper, "Understanding and Interpreting Pilot Opinion," Aeronautical Engineering Review, (March 1957), passim.

30. Ernst Heinkel, Stormy Life: Memoirs of a Pioneer of the Air Age, New York: E. P. Dutton & Co., 1956, p. 224.

31. Frank Whittle, Jet: The Story of a Pioneer, New York: Philosophical Library, 1954, passim.

32. Ronald D. Neal, "The Bell XP-59A Airacomet: The United States' First Jet Aircraft," Journal of the American Aviation Historical Society, Fall 1966, passim.

33. Bastian Hello, Final Report of Development, Procurement, Performance and Acceptance--XP-80 Airplane, Dayton, OH: Army Air Forces Air Technical Service Command, 28 June, 1945.

34. Grover Heiman, Jet Pioneers, New York: Duell, Sloan and Pearce, 1963, pp. 85-100.

35. Frederick Trapnell, "A Test Pilot Looks at the Jets," Aeronautical Engineering Review, (May 1950), passim.

36. Benson Hamlin, Flight Testing Conventional and Jet-Propelled Airplanes, New York: The Macmillan Company, 1946, passim.

37. G. S. Welch, "Effects of Compressibility Dives on Fighter Aircraft," Project No. (M-1)82, (a report of The Army Air Forces Board), Orlando, FL: The Army Air Forces Board, 19 June 1944, pp. 5-9.

38. R. Smelt, "Notes on Discussion With the Staff of Focke-Wulf Ltd.[sic]," Technical Note No. Aero 1644, Farnborough, Eng.: Royal Aircraft Establishment, May 1945, pp. 6-7.

39. Charles Burnet, Three Centuries to Concorde (London: Mechanical Engineering Publications Ltd., 1979), p. 38.

40. Richard Hallion, Supersonic Flight: The Story of the Bell X-1 and Douglas D-558, New York: The Macmillan Company in Association with The Smithsonian Institution, 1972, pp. 1-3.

41. Charles E. Yeager, "The Operation of the XS-1 Airplane," in USAF, Air Force Supersonic Research Airplane XS-1 Report No. 1, 9 January 1948, p. 19.

42. Ardath M. Morrow, Case History of the YB-35,YB-49 Airplane (Supplement to Case History of the XB-35 Airplane), Wright-Patterson AFB, OH: Historical Office, Executive Secretariat, Air Materiel Command, February 1950, passim.

43. Society of Experimental Test Pilots, The Society of Experimental Test Pilots: History of the First Twenty-Five Years, Covina, CA: Taylor, 1978, passim.

44. Larry G. Van Pelt, The Evolution of Flight Test Concepts, Edwards AFB, CA: History Office, June 1982, pp. 6-36.

45. Iven C. Kincheloe, "Flight Research at High Altitude, Part II," Proceedings of the Seventh AGARD General Assembly, the Washington AGARD conference, 18-26 November 1957.

46. Richard P. Hallion, "In the Beginning Was the Dream. . .", in Richard P. Hallion, ed., The Hypersonic Revolution: Eight Case Studies in the History of Hypersonic Technology, v. I: From Max Valier to Project Prime, 1924-1967, Wright-Patterson AFB, OH: Special Staff Office, Aeronautical Systems Division, 1987, p. xviii.

47. Eugen Sänger and Irene Bredt, Über einen Raketenantrieb für Fernbomber, Deutsche Luftfahrtforschung UM 3538, Ainring: DFS, August 1944 (translated by the Naval Technical Information Branch, Bureau of Aeronautics, U.S. Navy, as Translation CGD-32, A Rocket Drive for Long-Range Bombers, 1952), passim.

48. Wendell H. Stillwell, X-15 Research Results, Washington, D.C.: GPO, 1965, pp. 26-78.

49. Walter C. Williams, "The Role of the Pilot in the Mercury and X-15 Flights," Proceedings of the Fourteenth AGARD General Assembly, Lisbon, Portugal, 16-17 September 1965.

50. John V. Becker, "The X-15 Program in Retrospect," Raumfahrtforschung, March-April 1969, passim.

51. Clarence L. Johnson, "Some Development Aspects of the YF-12A Interceptor Aircraft," Journal of Aircraft (AIAA), July-Aug. 1970, passim.

52. Richmond L. Miller, "Flight Testing the F-12 Series Aircraft," Journal of Aircraft (AIAA), September 1975, passim.

53. Richard P. Hallion, On the Frontier: Flight Research at Dryden, 1946-1981, NASA SP-4303 (a volume in the NASA History Series), Washington, D.C.: GPO, 1984, pp. 189-199.

54. Thomas G. Foxworth, "North American XB-70A Valkyrie," Historical Aviation Album[s], nos. 7 & 8, 1969-1970, pp. 76-87, 164-175.

55. Kenneth Owen, Concorde: New Shape in the Sky, London: Jane's Publishing Co., 1982, passim.

56. Jay Miller, The X-Planes: X-1 to X-29, Marine-on St. Croix: Specialty Press, 1983, passim.

57. Philip F. Oestricher and Robert C. Ettinger, "F-16 High Angle of Attack Testing," XXIInd Symposium, The Society of Experimental Test Pilots, Beverly Hills, CA, 28 September 1978.

58. G. Warren Hall, "Research and Development History of USAF Stability T-33," Journal of the American Aviation Historical Society, Winter 1974, passim.

59. Conversation with Charles A. Sewell, Grumman Corporation, 1985.

60. R. Dale Reed, "RPRV's: The First and Future Flights," Astronautics & Aeronautics (AIAA), April 1974, pp. 26-42.

61. Henry S. F. Cooper, 13: The Flight That Failed, New York: The Dial Press, 1973, passim.

62. Barton C. Hacker and James M. Grimwood, On the Shoulders of Titans: A History of Project Gemini (a volume in the NASA History Series), Washington, D.C.: GPO, 1977.

63. Coutney G. Brooks, James M. Grimwood, and Loyd S. Swenson, Chariots for Apollo: A History of Manned Lunar Spacecraft (a volume in the NASA History Series), Washington, D.C.: GPO, 1979.

64. Clarence J. Geiger, "Strangled Infant: The Boeing X-20A Dyna-Soar," in Hallion, ed., Hypersonic Revolution, I, pp. 185-377.

65. Richard P. Hallion, "ASSET: Pioneer of Lifting Reentry," in Hallion, ed., Hypersonic Revolution, I, pp. 449-527.

66. John L. Vitelli, and Richard P. Hallion, "Project PRIME: Hypersonic Reentry From Space," in Hallion, ed., Hypersonic Revolution, I, pp.529-745.

67. Johnny G. Armstrong, Flight Planning and Conduct of the X-24B Research Aircraft Flight Test Program, AFFTC-TR-76-11, Edwards AFB, CA: Air Force Flight Test Center, 1977, passim.

68. M. J. Sugano and T. R. Brice, "The Approach and Landing Test Program of the Space Shuttle Orbiter 101," AIAA Aircraft Systems Conference, Los Angeles, CA 21-23 Aug.1978

69. Vance D. Brand, "Space Shuttle Redesign Status," in The Society of Experimental Test Pilots, 1986 Report to the Aerospace Profession, Beverly Hills, CA: SETP, 24-27 Sept. 1986, pp. 224-235.

70. Advanced Programs Office, NASA Lyndon B. Johnson Space Center, Space Shuttle Directions, JSC-20939, Houston, TX: NASA JSC, June 1986, passim.

71. Presidential Commission on the Space Shuttle Challenger Accident, Report to the President, Washington, D.C.: GPO, 6 June 1986, pp. 72, 104, 117-118, 120-148, 161, 176, 198-201, 236.

72. Michael Collins, Liftoff: The Story of America's Adventure in Space, New York: Grove Press, 1988, p. 224.

73. Stuart O. Schmitt, T. J. Wierzbanowski, and Johnny Johnson, "The Challenge of X-30 Flight Test," XXXI Symposium, The Society of Experimental Test Pilots, Beverly Hills, CA, 26 Sept. 1987.

74. Statement of James E. Webb, in U.S. Congress, Committee on Aeronautical and Space Sciences, Senate, Aeronautical Research and Development Policy, Washington, D.C.: GPO, 1967, p. 7.

AIR COMBAT ENVIRONMENT TEST AND EVALUATION FACILITY (ACETEF)

R. R. Smullen and S. D. Harris, LCDR, USN
Naval Air Test Center
Patuxent River, Maryland 20670, USA

SUMMARY

Recent combat experience has brought to light a critical need to assess the mission effectiveness of Naval aviation weapons systems and vehicles against a myriad of new threats. Flight testing is the primary source of data on the effectiveness of our aircraft and weapons, but flight testing is costly, and limited to only a few of the most crucial questions. Flight testing is also inherently a public event; it can be observed with impunity. Hence, flight testing is patently unsatisfactory for dealing with some issues related to national security interests. Senior decision-makers in the Department of Defense (DOD) require objective, quantitative assessments of the effectiveness of our weapons and people against literally thousands of possible combinations of threat and contingency planning conditions. We need a revolutionary approach to obtaining the requisite data. The Naval Air Test Center (NAVAIRTESTCEN) has begun to develop an innovative system, known as the Air Combat Environment Test and Evaluation Facility (ACETEF), to meet this need.

OVERVIEW

ACETEF will permit ground testing of fully integrated avionics systems in an environment that closely parallels actual combat. The data generated by ACETEF will augment that available from conventional flight testing in three ways. First, ACETEF will reproduce almost exactly the conditions encountered during a test flight, allowing systems engineers to study a problem under controlled conditions. Second, and perhaps more important, ACETEF will subject the weapons and crew to conditions that cannot be reproduced in actual flight, short of real combat. Finally, ACETEF testing will be covert; it cannot be witnessed by uninvited observers. The data that results from ACETEF will be unique, and invaluable to decision-makers at all levels of both the systems acquisition and operational command hierarchies.

Development of ACETEF entails no known technological risks; the elements already exist in various stages of maturity. ACETEF adds value to existing NAVAIRTESTCEN assets by integrating them into a complete system. This article describes some of the current capabilities, and the problems that have led NAVAIRTESTCEN to conceive and begin development of the ACETEF approach to testing.

REQUIREMENTS

The development of the ACETEF concept by NAVAIRTESTCEN was motivated by a number of interrelated factors. First and foremost is the unprecedented escalation in the complexity and effectiveness of a multitude of threats to Naval Aviation. Recent combat experience has been both encouraging and sobering. We haven't miscalculated the threat yet, but we can envision a future scenario with less desirable outcomes. Recent intelligence reports depict an increasingly sophisticated combat capability in areas previously considered relatively unimportant. New missiles with advanced guidance systems, controlled by complex and robust command and control networks present a serious challenge to our ability to project power. Weapons of entirely new type are being tested by potential adversaries, and will be fielded in the next few years. At present, we have precious few national assets that can generate objective data on the effectiveness of our systems against those currently and probable future threats. Our instrumented test ranges cannot come close to providing test conditions that approximate the threat density we can expect in real combat. Conventional flight testing is inadequate in the face of the rapid increase in system complexity necessary to counter the threats and achieve military and political objectives. We simply cannot fly enough to evaluate all of the threat situations that analysts can "dream up".

A second factor that led to the ACETEF concept is the rapidly increasing sophistication of foreign intelligence measures. Foreign surveillance systems (e.g. SIGINT, PHOTINT or ELINT) can easily monitor flight tests, gathering data on the performance capabilities and limitations of vehicles and weapons. Both visual and electronic data can be obtained readily from even rudimentary monitoring systems and technology. Although some issues can only be resolved with confidence during flight testing (such as basic vehicle performance limitations), there is the constant risk of compromising classified signals if electronics emissions are not held to a minimum. When the signals are limited, the value of the flight test is limited. The ACETEF system incorporates a secure approach that minimizes the likelihood that a foreign technology can penetrate and monitor tests.

The third factor is cost. Current developmental testing technology, methodology, and instrumentation characterizes baseline system performance and verifies specification compliance. Operational test and evaluation relies on expensive flight testing to provide estimates of mission effectiveness. ACETEF will fill an ever-widening gap between the two. ACETEF will allow the Navy, for the first time, to begin to evaluate objectively the performance of aircraft, weapons, and crew, in covert analyses of total system combat performance. The ACETEF methodology will significantly enhance the quality of developmental test and evaluation, and will become the foundation from which operational flight testing moves toward more thorough assessment of crucial operational issues. ACETEF testing will be integrated with conventional flight test methodologies to validate the results of the simulations. Flight testing will become more cost-effective and will be focused only on those questions that require flight testing. With exponentially increasing costs of sensors and weapons is formidable. To achieve a reasonable level of confidence that the investment in those systems has been and is wise, we need hard facts: objective data that show how effective our systems are against those threats. ACETEF will provide those data.

Finally, the need to evaluate the interoperability of our systems has been demonstrated repeatedly during the last few years in limited combat situations. Just counting all the numbers of ways that our systems must interact is challenging. Navy aircrews must talk to Army, Marine, and even NATO parties on the ground. Third-party targeting is a reality, representing an incredible advance in mission effectiveness, but at the cost of immense burden on command, control, and communications systems. Air Force aircraft must be integrated into Navy airborne strike units under certain combat scenarios. Testing all combinations of systems in flight test is simply not possible. Waiting to test them in combat is unthinkable. Testing in simulation is both cost-effective, and provides the requisite data.

ACETEF CONCEPT

A typical system integration test of today entails a fully flight-worthy aircraft with complete weapons systems suspended in an anechoic chamber that provides greater than 100dB isolation for electromagnetic (EM) signals from the outside world. The aircraft is illuminated with radar and other signals as if it were flying through a real combat mission. In a high-fidelity simulation facility next door, a flight crew straps into a cockpit that duplicates the aircraft in the chamber. The crew observe displays and control systems much as they would during a flight test. In an adjacent shielded hangar, other aircraft are connected to external power and their systems are subjected to stimulation and analysis. Such tests are conducted as a matter of course at the NAVAIRTESTCEN. However, the various elements, aircraft in the chamber, and in the hangar, and pilot in simulator are independent. The data that they yield are not correlated; few data on interoperability are available. Whether the systems will work together effectively is still uncertain. Whether they will work at all in some advanced high-density threat environment is even less clear. We can test the systems to specification, but we still do not know if they will do the job. Many questions remain unanswered. Can a given jammer be integrated into a new platform and provide increased survivability? Can a new air-to-air self-defense missile be added to a ground-attack aircraft and provide the crew with increased likelihood of mission success? Or will the new capability simply increase crew workload and result in decreased mission performance?

The ACETEF approach is to exploit simulation technology to provide a testing environment that reproduces the conditions of actual combat. In ACETEF, a fully integrated weapons system, incorporating vehicle, avionics, weapons, crew, other aircraft in the strike group, and critical elements of the command/control hierarchy will be immersed in a stimulation/simulation environment that literally fools both aircraft and crew into believing that they are in actual combat. Aircraft systems are fooled through a combination of simulation and stimulation. Simulation is accomplished by digital computers; stimulation is accomplished by computer controlled environment generators that provide radio frequency, electro-optical, or laser stimuli which duplicate, as closely as possible, real signals (including war-reserve modes, when appropriate). The crew is fooled by providing very high fidelity sensory cueing (visual, aural, and motion) while simultaneously providing realistic workload conditions (threats, mission objectives and constraints, communications channels, etc.). The total EM environment that an aircraft/aircrew experiences during combat is simulated, encompassing EM interference among subsystems on-board the aircraft, threat-generated signals, communication and data link signals, radar returns, jamming and other electronic countermeasures, counter-countermeasures, visual stimuli, and other signals through the EM spectrum. The connections among flight crew, aircraft, and other players is real-time and closed-loop. When the pilot presses a button in the simulator cockpit, the appropriate system is activated in the aircraft suspended in the anechoic chamber. When a jammer is activated in the chamber, the pilot sees his simulator radar display become distorted. This type of integration between simulator and aircraft has already been demonstrated in a limited test. A serious problem in the stores management system of the F/A-18 aircraft was diagnosed and resolved using this approach in 1980. ACETEF simply builds on that experience.

In ACETEF, the combat scenario may be hypothetical, devised to address some possible future threat, or it may be real, constructed quickly to address some critical question that may arise during system development, contingency planning, or training exercise. As a result, the aircrew and aircraft will perform in a manner that very closely duplicated operations under battle conditions. And hard facts can be accumulated on just how well they do the job.

As ACETEF evolves, it will provide a unique testing facility for new applications of computing technology in air warfare: in communications, sensor fusion, and tactical and strategic decision-aid applications, for example. The central role of the computer in managing combat has only recently begun to emerge. ACETEF promises to be a unique facility for testing new concepts for system integration. Rapid prototyping of new systems will be supported by the availability of the data-rich testing environment. Of special interest is the role that computing technology can play in upgrading the effectiveness of current systems by retrofitting enhanced computers and new systems architectures. The effectiveness of the A-6 and F-14, and hence their life-expectancy, are being improved by orders-of-magnitude by the expedient of adding new computing capabilities. For example, a major performance gain in the the F-14D radar over its predecessor is achieved in the computing capability embedded in the radar. With a few orders-of-magnitude increase in computational power, new capabilities can be realized.

Sensor fusion, the long-sought algorithm to correlate pre-mission planning data, and late-breaking intelligence data, with on-board sensors, such as the radar warning receiver, radar or IR sensors, etc., would be practical with conventional programming techniques if the computer power were available on-board the aircraft. The algorithm to achieve sensor fusion can be demonstrated on conventional ground-based computers. A reliable algorithm has been demonstrated to identify the path of least threat to or away from a target zone. Other programs perform automated assignment of weapons to threats to achieve a mathematically optimal kill-ratio. These programs work in theory, but will they work in combat? These enhancements may be extremely cost-effective, obviating substantial investments in new weapons systems until breakthroughs, such as stealth, propulsion, structures, aerodynamics, and other technologies are mature enough to warrant development of new aircraft.

Consider the implications of increasing the on-board computing capabilities of tactical aircraft by hundreds or thousands of times. Such is the objective of DARPA's Strategic Computing Initiative, the Very High Speed Integrated Circuit technology program, and other substantial DOD-sponsored efforts. How can that power be exploited to increase survivability and mission effectiveness? How can we evaluate the tradeoffs among the hundreds of alternative possible systems configurations that are emerging as practical?

In today's intensely competitive systems acquisition process, there exists no real incentive for various vendors to develop common solutions and foster commonality among aircraft. That same vital process that provides many diverse competitive solutions to Defense problems, fails to provide cost-effective means for evaluating the tradeoffs among the solutions. The responsibility for discovering potential advantages of specifying commonality falls on the government facilities. The ACETEF will provide substantial impetus for development of common solutions to common problems among various aviation communities.

The software that controls ACETEF simulations, and the basic systems architecture of our facilities can be exported to other government and industry simulation laboratories. In a joint effort with the Naval Training Systems Center, involving their Visual Technology Research Simulator and advanced multiprocessor computer facilities, we are developing an acquisition strategy that exploits our initial investment in aerodynamics, avionics, and threat models in the test and evaluation arena by transitioning them to the development of training systems, here-to-fore an extremely difficult thing to accomplish. If we succeed, and we expect to demonstrate the approach in FY88, we will have opened the door to substantial (multi-million dollar) savings in training simulator acquisition and life-cycle costs. ACETEF will magnify those savings.

Other benefits can be foreseen in recently demonstrated applications. The threat environment generators used in ACETEF could be tapped to provide realistic scenarios to support battle force in-port training exercises. New tactics would undoubtedly emerge as a clearer understanding of the effectiveness of our systems evolves. Innovative ideas could be tested in the crucible of near-reality before committing to full-scale development. We expect that ACETEF will provide an especially fruitful test site for development of reusable software modules in the programming language Ada. We anticipate that ACETEF will confront and develop new ways of integrating large real-time process control computers in tightly-coupled networks. All of these advantages arise through our systematic, build-a-little, test-a-little, evolutionary approach to implementation of the ACETEF concept.

CONCLUSIONS

The ACETEF is an evolutionary/modular system that provides a systematic building block on which the T&E answers of today and tomorrow can be found. We utilize some of the elements of the ACETEF routinely today. It was the immense sucess of our experiments with aircraft/simulator integration that led us to conceive, and eventually to plan for, the Air Combat Environment Test and Evaluation Facility. In ACETEF, the encounter between our systems (aircrew and aircraft) and their projected combat environments, reproduced as closely as simulation technology and our understanding of the threats will allow, will provide data of incalculable value. Data that cannot be obtained any other way.

FLIGHT TESTING IN THE NETHERLANDS, AN OVERVIEW

by
J.T.M. van Doorn and R.L. van der Velde
National Aerospace Laboratory NLR
Anthony Fokkerweg 2
1059 CM Amsterdam
The Netherlands

SUMMARY

This report gives an overview of the flight test activities and capabilities in The Netherlands. A general description of the flight test programs of the last decades with civil and military aircraft, helicopters and research aircraft will be given. Some of the highlights of the more recent programs will be presented, i.e. the type certification of the Fokker 50 and Fokker 100 civil transport aircraft, evaluation and certification trials with the military F16 fighter aircraft, helicopter-ship compatibility testing and the determination of the mathematical model of the Cessna Citation 500 for a Phase II flight simulator. Furthermore a short description will be given of the flight test instrumentation and flight test techniques that have become available in The Netherlands during the last decade.

1 INTRODUCTION

Only a few years after the historical flight of the Wright brothers in 1903 several aeronautical organisations were founded in The Netherlands that still exist. The Royal Netherlands Air Force was founded in 1913. In 1919 Anthony Fokker of The Netherlands founded what is now Fokker Aircraft Company. Also in that same year both KLM Royal Dutch Airlines and what is now the National Aerospace Laboratory NLR were founded. At a somewhat later stage the faculty of Aerospace Engineering of the Delft University of Technology joined this group of organisations in which aircraft operators, the national aircraft industry and the aerospace research institutes were represented. These organisations have always cooperated very closely and the lines of communication have always been very short. Before World War 2 both civil and military aircraft were being developed in The Netherlands. After the war Fokker directed its main development activities towards civil aircraft types like the F27 Friendship and the F28 Fellowship and more recently, the Fokker 50 and Fokker 100. In the military segment Fokker produced several military fighter aircraft types under license from British and American companies. As a consequence the greater part of the flight test activities in The Netherlands was with civil aircraft. Especially the flight testing of the Fokker 50 and Fokker 100 prototype aircraft formed a great percentage of the total flight test activities in the last few years. The flight tests with military aircraft were limited to evaluation and certification of subsystems and general support to air force and navy.

Speaking of flight testing in The Netherlands is hardly possible without mentioning the National Aerospace Laboratory NLR. As a result of its central position NLR plays an important role in virtually all flight test activities in The Netherlands. NLR is the central institute in The Netherlands for aerospace research. Its principal mission is to render scientific support and technical assistance on a non-profit basis to Dutch and foreign aerospace industries and organisations, civil and military aircraft operators and government agencies concerned with aviation and space flight.
NLR closely cooperates with Fokker in aircraft development projects under contract with The Netherlands Agency for Aerospace Programs (NIVR); it assists aircraft operators (KLM Royal Dutch Airlines, Royal Netherlands Air Force, Royal Netherlands Navy) concerning the evaluation of aircraft and equipment, and concerning technical problems in aircraft operation. Furthermore NLR assists The Netherlands Department of Civil Aviation (RLD) in its supervisory task with respect to air traffic control, airworthiness, flight safety, accident investigation, environmental control, etc.
NLR is a customer-oriented research organisation, deriving its income for 70 percent from research-, development-, test- and evaluation-contracts and for 30 percent from subsidies from the Dutch government for its basic research programs. NLR is headed by a Board, composed of representatives from the Dutch government, aerospace industry and aircraft operators. NLR employs a staff of about 780, and is organised in five major research divisions: Fluid Dynamics, Flight, Structures and Materials, Space and Informatics. (Fig.1.1)

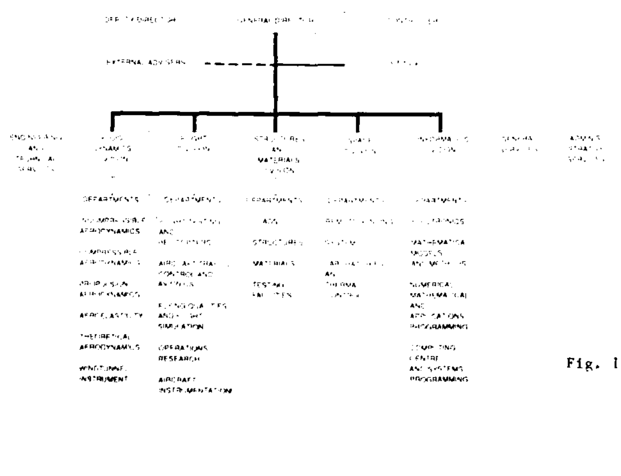


Fig. 1.1 NLR's Organisation.

In the Flight Division three departments can be found that play an important role in flight testing: the Flight Testing and Helicopters Department, the Operations Research Department and the Aircraft Instrumentation Department.

This report gives an overview of the flight test activities and capabilities in The Netherlands. A general description of the flight test programs of the last decades with civil and military aircraft, helicopters and research aircraft will be given. Some of the highlights of the more recent programs will be presented, i.e. the type certification of the Fokker 50 and Fokker 100 civil transport aircraft, store certification trials with military fighter aircraft, helicopter-ship compatibility testing and the determination of the mathematical model of the Cessna Citation 500 for a Phase II flight simulator. Furthermore a short description will be given of the flight test instrumentation and flight test techniques that have become available in The Netherlands during the last decade.

2 FLIGHT TEST PROGRAM OVERVIEW

2.1 Civil transport aircraft

Fokker aircraft

The first civil air transport aircraft developed in The Netherlands since World War 2 was the Fokker F27. Following a program of subcontract work for the Allied Air Forces which culminated in military aircraft construction contracts, such as the production of 330 Gloster Meteor, 400 Hawker Hunter and 350 Lockheed F104G Starfighter aircraft, Fokker embarked in 1953 upon its biggest postwar venture: the design, development and construction of a twin turboprop, short-haul airliner with a seating capacity of 36. Shortly after its introduction a version with a stretched fuselage was introduced for 46 to 50 passengers. The aircraft would be known as the F27 Friendship. This aircraft was Fokker's bestseller for 25 years, 786 aircraft were sold, 205 of which were manufactured under license by Fairchild in the USA. There it was known as the Fairchild F227.

In the early sixties Fokker initiated the development of a short-haul twin-jet with tail-mounted engines and a seating capacity for 65 passengers: the F28 Fellowship. It had a take-off weight of 25,700 kg and the range was about 1000 nautical miles. The F28 flight test program started with the maiden flight of the first prototype on 9 May 1967. Three aircraft were scheduled to be involved in the development and certification flying. After two months the second prototype joined the program and two months after that a third aircraft was added. The first two aircraft were identically equipped with extensive test- and recording equipment. The third aircraft was intended to serve as "stand-by" and to perform trials requiring limited or very specialised instrumentation, such as for icing tests. The flight test program took 20 months to complete, in this period approximately 900 flight hours were logged. The aircraft was certified in 1969. In later years several versions were developed by Fokker, the original version of the F28 was then called the F28 Mk 1000.

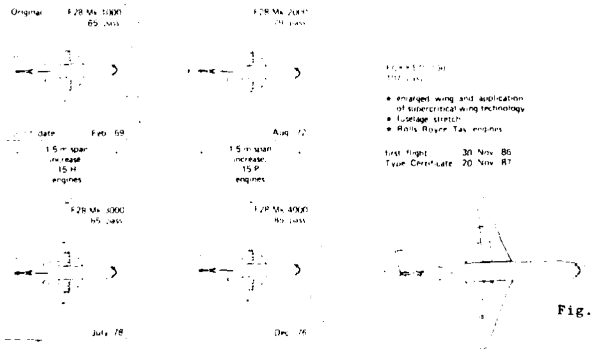


Fig. 2.1.1 The F28-family and how it evolved into the Fokker 100.

Fig.2.1.1 gives an overview of this F28-family and how it eventually evolved into the Fokker 100. The first new version was the F28 Mk 2000, which was certified in 1972. It had an extended fuselage and could carry up to 85 passengers. From this version the Mk 4000 was developed. It had an increase of wingspan of 1.5 m, aerodynamic refinements, a new look interior and uprated engines with improved noise attenuation. This version was certified in 1976. Hereafter a Mk 3000 was developed from the original Mk 1000, it had the original fuselage, but otherwise the same features and improvements of the Mk 4000. It was certified in 1978.
All versions were subjected to flight test programs, in which the consequences of the alterations were evaluated and certified. These tests were all done on the first prototype, the A1, the flight test instrumentation of which was extensively updated in the course of the years to keep pace with the increasing "user requirements". This same aircraft was later, at an age of 20 years, used as the "Avionics Test Bed" (ATB) in the Fokker 100 development program. Of the F28 Fellowship-family 241 aircraft were sold, most of which were Mk 1000, Mk 2000 and Mk 4000 versions.

Production of both the F27 and F28 was terminated in 1987 as the production of the Fokker 50 and Fokker 100 aircraft fully occupied Fokker's production resources. The development of these new aircraft types was initiated in 1982. In 1983 Fokker officially launched the Fokker 50 and Fokker 100 programs simultaneously.
The flight test programs started in late 1985 for the Fokker 50 and late 1986 for the Fokker 100. The test programs included full scale prototype certification according to JAR and FAR regulations, they are described in more detail in section 3.1 and in Reference 1.

The Fokker 50 (Fig 2.1.2) is a twin turboprop for 50 passengers and it is a much improved and modernised derivative of its predecessor, the F27 Friendship. The main differences can be found in the flight controls, landing gear, engines (P&W 124), propellors (Dowty Rotol 6 blades), avionics, electrical system, hydraulic system.
The first flight with the first Fokker 50 prototype was on 28 December 1985. The type certificate was awarded on 15 May 1987. In the test program two prototypes and one production aircraft were used. Until the type certification 985 flight hours were logged.

The Fokker 100 (Fig.2.1.3) is a short-haul twin-jet with tail-mounted, high by-pass ratio engines with seating capacity for 108 passengers. The Fokker 100 is a derivative of the F28 Fellowship Mk 4000, as can be seen in Figure 2.1.1. From a structural point of view it is a very drastic adaptation. The main differences are: an extended fuselage (+5.9 m), a newly designed wing with larger span (+3 m) and chord, based on the application of the supercritical wing technology and a horizontal stabilizer of increased span (+1.4 m). Furthermore the Fokker 100 has completely new engines, Rolls Royce Tay 620 and 650 with a thrust of 13,850 lbs, resp. 15,000 lbs, with a much lower specific fuel consumption, and a very low noise level which meets the latest (Stage 3) regulations. Other major differences are: thrust reversers, second generation digital avionics with full cat. 3B Autoland capability and a "glass cockpit".

The first flight with the first Fokker 100 prototype was on 30 November 1986. The type certificate was awarded on 20 November 1987. In the test program two prototypes and one production aircraft were used. Until type certification 1052 flight hours were logged. The flight test programs of the Fokker 50 and Fokker 100 will be discussed in more detail in Chapter 3.1.

Fig. 2.1.2 Water ingestion trials with the Fokker 50 at Cranfield, U.K.

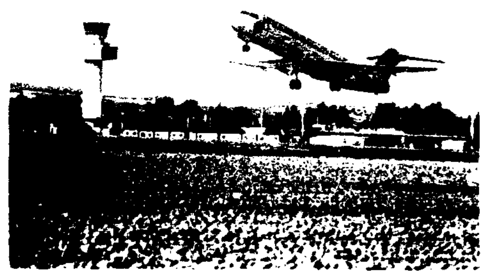


Fig. 2.1.3 Take-off and landing performance and fly-over noise were measured at Granada, Spain (photograph Fokker).

During the execution of a flight test program the relation between Fokker and NLR is as follows. The program management and the test conductance are performed by the Fokker Flight Department. Flight test instrumentation operations and data preprocessing operations are a shared effort of Fokker Electronics Laboratory and NLR's Aircraft Instrumentation Department. Each department developed certain instrumentation subsystems which will be further described in Chapter 4 and consequently operated them during the tests. Both departments operate their own data preprocessing stations which interface with their data acquisition systems. The data preprocessing stations interface with the respective Fokker- and NLR mainframe computers.

Cessna Citation 500.

In the spring of 1986 the Dutch Government Civil Aviation Flying School (RLS) decided to purchase a phase II flight simulator for the Cessna Citation 500 business jet, used in the final part of the pilot training. The objective was to reduce the costs of the training by transferring part of the type training from the aircraft to the simulator, enabling the RLS to reduce its fleet from six to three aircraft.
For the Citation 500, which was developed in the late sixties, no adequate mathematical model and data packages were available. Therefore the Faculty of Aerospace Engineering of the Delft University of Technology (DUT) and NLR were selected to execute a flight test program, identify mathematical models of aerodynamic forces and moments, engine performance characteristics, flight control system and landing gear and to evaluate the models with off-line and pilot-in-the-loop-real-time simulations. One of the aircraft was equipped with an advanced high accuracy flight test instrumentation system from NLR's MRVS-family (see Chapter 4) and a flight test program was carried out consisting of 52 hours in 23 flights. A data base was formed from which a mathematical model could be developed. Different types of system identification techniques were applied to develop models of the aerodynamic forces and moments, the static and dynamic engine performance, the flight control system and the landing gear. The time between program go-ahead until first flight was 3 months, the flight test program was virtually completed in 2 months. Delivery of all mathematical models was completed within five months after the first flight. This program is discussed in more detail in Chapter 3.4.

2.2 Military aircraft

Fighter aircraft

The Netherlands did not develop a capability to design military fighter aircraft and as a consequence the scope of flight test programs with these aircraft is rather small compared to countries which do have such a capability. However, a constant stream of flight tests has been going on during the last decades. The first post war military fighter programs were executed on the Lockheed F104G Starfighter of the Royal Netherlands Air Force (RNLAF).
Later, in 1970, the Northrop NF5 made its appearance. This version of the well known F-5 fighter line had been developed to the specification of the Dutch and Canadian Air Forces. The first aircraft which had been extensively used in the test program by Northrop came to The Netherlands provided with a digital data acquisition system and NLR took care of the instrumentation. This aircraft was in use as a general purpose test aircraft for the RNLAF for many years. During this period NLR managed the instrumentation, adapted it to specific requirements and trained Air Force personnel to operate it.The NF5 was succeeded by the General Dynamics F16 (Fig.2.2.1), which is the subject of today's military flight testing. The nature of these tests could best be described as improvement programs, improvement in the sense of improving or expanding the weapon employment capabilities and improving the operational capabilities of the man-machine combination. During the operational life of military aircraft it is often necessary to equip the aircraft with new systems or external stores, to expand the operational envelope or to certify new configurations.

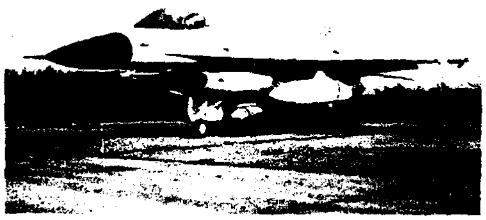


Fig. 2.2.1 NLR plays an essential role in numerous flight test programs for the qualification of weapon systems for aircraft of the Royal Netherlands Air Force.

In executing these complex tasks, NLR has given support to the Royal Netherlands Air Force (RNLAF) ever since the first tests with the F 104G, thereby acquiring a great deal of expertise in this field.

Many test programs involve investigations to be performed during actual flight. NLR provides a total flight test package consisting of:
- planning and organization of a flight test program,
- design, installation and operation of all required test equipment,
- data processing and data analysis.

NLR has done certification work for several aircraft-store combinations of the above mentioned aircraft types of the RNLAF, by providing support in flight testing and analysis to investigate:
- flutter
- loads and stresses
- performance
- flying qualities
- store separation

Other military flight test programs in The Netherlands comprise the development, test, evaluation and/or certification of new subsystems, such as a reconnaissance system, a gunsight, a terrain profile matching system, the test and evaluation of modifications to the aircraft, both hard- and software, such as flap scheduling, Fire Control Computer weapon delivery software and radar software and the development of new operational tactics.

The unprecedented flying qualities of the F16 and the extreme demands on the human physique raised the question whether the pilot would become the limiting factor in F16 operations. Pilot's complaints led to some research programs to investigate the accelerations and the resulting loads in the spine and neck of the pilot during high-G manoeuvres and during low altitude, high speed missions.

At this moment the RNLAF has at its disposal two F16 aircraft, an F16A and an F16B, which have been equipped with electrical and mechanical provisions for the rapid installation of a powerful and versatile digital flight test data acquisition system. The aircraft can be converted from squadron service to flight test service and vice versa in two days. The data acquisition system, designed by NLR, is described in detail in Reference 2. Both instrumented aircraft have been used in several of the mentioned certification, evaluation and research programs.

The relation between the Air Force and NLR during the execution of a flight test program is as follows: the program management is done by the Air Force and, in most cases, the test conductance is managed by NLR's Flight Test Department. Flight test instrumentation and data preprocessing is provided by NLR's

Aircraft Instrumentation Department. Data reduction and analysis is done by NLR specialists. The final reporting is done by NLR and/or Air Force specialists. Some programs are entirely conducted by the Air Force, sometimes using NLR instrumentation, sometimes only using the test pilot's judgement.

Lockheed Orion P-3C

In 1983 the Royal Netherlands Navy ordered a training simulator for the Lockheed Orion P-3C anti-submarine patrol aircraft.
The mathematical models for this simulator were extracted from rather old and scanty flight test information and theoretical and windtunnel data. In order to be able to adjust and validate the models more accurately, better and more complete information on the aircraft had to be obtained.
Therefore it was decided to install a data acquisition system in one of the RNLN's operational aircraft and to execute an extensive flight test program to acquire the so-called proof-of-match data.

2.3 Helicopters

With regular intervals during the last 25 years NLR's Flight Testing and Helicopters Department has been involved in helicopter-ship qualification testing for not only the Royal Netherlands Navy (RNLN) but also for several foreign navies. The objective of these tests is to determine the take-off and landing flight envelope for helicopters on board ships with a small flight deck to ensure an optimal operational availability.
NLR has developed a method to assess the possibility of a safe take-off and landing under specified adverse conditions, with as high a payload as possible. The operational envelope can be greatly expanded by utilizing the optimized take-off and landing techniques designed by NLR. The final result could mean a considerable increase in the operational availability of the helicopter on board the ship. Because of the unique character of each helicopter-ship combination and the innumerable possible combinations it is understandable that usually no extensive testing has been carried out by the manufacturer for the combination of interest. Helicopters tested in the programs are, a.o., several versions of the Westland Lynx (Fig.2.3.1), Agusta Bell AB 212 and Dauphin HH 65A. This subject is further discussed in more detail in Chapter 3.3.

Fig. 2.3.1
Helicopter-ship qualification testing for the Royal Netherlands Navy, Westland Lynx helicopter.

Other helicopter flight test programs in The Netherlands have rather an ad-hoc character, some examples are given:
- Determination of cross wind limits for the Westland Lynx
- Evaluation of an engine health monitoring system
- Performance evaluation of the Agusta A 129
- Evaluation of night vision goggles during night flights with the Bo 105
- Evaluation of an enhanced Stability Augmentation System of the Bo 105
- Determination of the navigation accuracy of several Doppler navigation systems with the Bo 105
- Determination of the radar cross section of the Alouette III and Bo 105 as a function of terrain cover
- Pilot's workload assessment

2.4 Research aircraft

There are three research aircraft in The Netherlands, two of them belong to NLR and the third one belongs to the Delft University of Technology (DUT). NLR operates a Fairchild Metro II (Fig.2.4.1) and a Beechcraft Queen Air 80 (Fig.2.4.2), DUT operates a De Havilland Beaver.

Fig. 2.4.1 NLR's Fairchild Metro II aircraft.

Fig. 2.4.2
NLR's Beechcraft Queen Air 80 aircraft equiped with a scatterometer for remote sensing research, developed by the Delft University of Technology.

In relation to the subject of flight testing the research aircraft are being used for research, development, test and evaluation of new:
- Flight test techniques
- Flight test-, aircraft- and cockpit-instrumentation
- Remote Sensing equipment, such as SLAR, SAR, Scatterometer, visible-light scanner
- ILS/MLS-procedures
- Air traffic control procedures
- Advanced avionics systems

Examples of former, recent and present-day flight test programs with the research aircraft are:
- Investigation of liquid-motion during zero-g flights, in preparation for the experiments with fluids in partly filled containers placed in the Fluid Physics Module in Spacelab.
- Investigation of techniques to assess pilot workload. Results were based on the generation of subjective ratings, physiological measurements and task performance parameters.
- Investigations into the optimum presentation formats on programmable Electronic Flight Instrument Systems (EFIS-displays)
- Research into the application of a Head Up Displays and the optimum presentation formats
- Flight tests with a side stick controller, in relation to the development of criteria for handling qualities of transport aircraft with fly-by-wire flight control systems, such as the A320. This development program also incorporated flight tests with the Total In Flight Simulator (TIFS) in the USA.
- Development of the Non-Stationary Measurement technique (NSM) for the determination of lift-drag polars in accelerated flight and the determination of stability and control parameters (see Chapter 4)
- Development of the STALINS-method for the measurement of aircraft trajectories during take-off and landing performance measurements (see Chapter 4)
- Development of the ALAND-method for the measurement of the aircraft trajectory and touch-down point during Fokker 100 Autoland cat 3B certification testing (see Chapter 4).
- Development of a Synthetic Aperture Radar (SAR) for remote sensing research
- Application of a Side-looking Airborne Radar (SLAR) as a research tool in the area of surveying and monitoring of land and sea. The Dutch digital SLAR is a joint development of the Delft University of Technology (DUT), Physics and Electronics Laboratory TNO and NLR. It has a resolution of 7.5 m across track and 16 mrad along track. The SLAR data is recorded on board on a high-density digital recorder. The data can be geometrically corrected for variations in aircraft attitude and motion through simultaneously recorded inertial reference data. Furthermore, radiometric corrections for system parameters are executed to calculate radar backscatter coefficients.
- Development and operation of a 6-band (between 1GHz and 18GHz) Scatterometer made by the Delft University of Technology. Previously the instrument was used in connection with the development of a scatterometer in the ERS1 satellite of the European Space Agency (ESA), which will measure wind force and direction on a global scale from the radar reflection patterns of the oceans. At present this scatterometer is used for research on radar reflection in agricultural applications.

Both NLR aircraft have been equipped with high accuracy digital data acquisition systems and mounting provisions under the fuselage for installation of instrumentation pods, antennas and other equipment. The Metro has a 610x610 mm camera hatch with 40 mm thick optical quality glass. The glass can be replaced with metal plates for mounting special equipment, such as antennas and the trailing static tube. The Queen Air has a tiltable mounting system under the fuselage, at present used for the scatterometer disc antenna which has a diameter of 1 m.

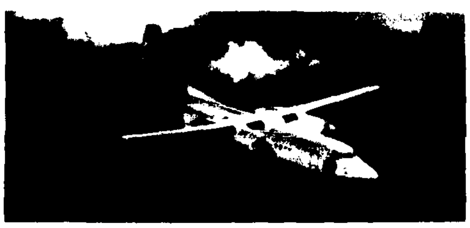
3. FLIGHT TEST PROGRAM HIGHLIGHTS

In this chapter four flight test programs are described in more detail. They are chosen because they clearly demonstrate the quality and versatility of flight testing in The Netherlands.

3.1 Type certification flight testprograms of the Fokker 50 and Fokker 100

In 1983 the Fokker Aircraft Company launched both the Fokker 50 and Fokker 100 programs simultaneously. Obviously that did not mean that flight testing should take place also simultaneously. Based on market demands and engine availability dates it was decided that the Fokker 50 should be ready for flight testing first and the program should be laid out in such a way that no interference with the Fokker 100 flight tests would occur. (Ref. 1).

The Fokker 50 is a derivative of the F27 Friendship, it is a twin turboprop, seating about 50 passengers. (Fig. 3.1.1).



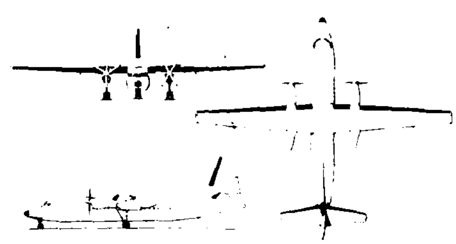


Fig. 3.1.1 The Fokker 50.

The project started in 1983 as a minimum change derivative, retaining the basic structure of the well proven F27, but with new engines. However, after the program go-ahead, mainly due to marketing pressure, the definition of the aircraft progressively changed from a minimum change to a fully modernized derivative.

An overview of the additional changes after program go-ahead is as follows:
- the pneumatic system was replaced by a fully hydraulic system,
- the electrical system became an all AC system,
- NAV/COM equipment was fully modernized,
- electronic flight instruments (EFIS) were introduced,
- the integrated alerting system (IAS) was new,
- the cockpit layout was fully modernized adopting the "dark cockpit" philosophy, as engineered for the Fokker 100,
- the pitot/static system was moved from the wing tips to the forward fuselage,
- the airconditioning system was redesigned,
- the interior layout was fully modernized, including new passenger and service/emergency doors and more, but smaller windows,
- for engine-non-containment reasons the trim control routing was separated from the main control routing.

Apart from the engine-nacelle and air-intake, that had to be changed drastically the aerodynamic changes were minor:
- rudder and ailerons were aerodynamically balanced to reduce control forces,
- wingtips were provided with "foklets", bent-up wingtips to improve lateral stability,

In 1983, when the program was launched, a 500 hrs flight test program was planned for the basic certification of the Fokker 50, to be flown with two prototypes and a first series aircraft in 7 months. By the end of 1985, due to the mentioned additional system changes and general reconsiderations, the program had grown to 700 hrs.

Due to a variety of reasons the Fokker 50 production didn't meet its schedule resulting in a late arrival of the prototypes at the flight department and very high pressure on a first flight date. It was decided to fly before the end of 1985 (28th Dec.). However, the aircraft was incomplete and the instrumentation system not yet operational. So, although this first flight was a success, the aircraft went straight back into the hangar to be completed and readied for the start of the flight test program. Also then, the pressure was tremendous, and when the second flight was finally made just over two months later, the aircraft and its instrumentation system were still not complete. It was only just sufficient to start with the first few weeks of the program.
It was hoped that there would be sufficient opportunity to complete the aircraft and its instrumentation system in the first phase of the flight test program, during the down-times in the nights and weekends, but this did not work out too well.
It became clear that the absence of a large certification flight test program for a number of years had seriously eroded the ability to estimate the necessary steps leading to a successful flight test program.

Due to the originally assumed limited Fokker 50 program, the instrumentation systems (see chapter 4 for more details) of both prototypes were designed to be dedicated only to the planned tests on each aircraft, thus accepting a very limited flexibility in program planning; both aircraft were not fully interchangeable!
Also, the instrumentation systems were build to match the early specifications, leaving little space for later additional requests. Too many of those came in a later stage causing significant problems and basic reconfigurations during the program.

The Fokker 50 flight test program was expected to be a fairly smooth program. In aerodynamic respect the aircraft was not much different from the F27 and most aerodynamic changes, like the foklets and horn-balanced rudder, had been flight tested on a modified F27, Fokker's Maritime testbed, the year before. The new pitot-static system had also been evaluated on that aircraft.
Although Fokker was the first customer for the Pratt & Whitney 124 turboprop engine and its engine and propellor electronic control systems, not many troubles were expected, neither with the engine nor with the propellor. The engine had been flight tested together with the six bladed Dowty propellor on a Viscount, P & W of Canada's flying testbed. Besides, the maximum power ratings of this engine, being 10% higher than those of the RR-Dart engine, had been flown on the Fokker F27 Maritime test bed by temporarily boosting the Dart engines to the required power level. Limited windtunnel testing had been considered adequate. The general optimistic view was that the program should be smooth.
The reality was different. The aerodynamic changes introduced by engine/propellor combination proved to have considerably more effect on the flying properties of the aircraft than expected.

Due to various problems with the engine and its electronic control system, as well as the propellor electronic control system, much single engine testing was postponed for several months.
A few changes in the longitudinal stability of the aircraft proved to be necessary, requiring time consuming hardware changes. This also affected the definition of the autoflight control laws, so final AFCS testing could not start before the longitudinal problems had been solved.
Those set-backs caused considerable delay, affecting the runway performance and noise program that should have been finished before the summer season in Granada, Spain, generally an ideal flight test location all the year through, except for the hot summer with its turbulent atmosphere. For the runway performance program Manching, the MBB flight test centre in the south of Germany, was chosen. With their excellent facilities MBB provided good support to complete the runway performance program within the planned period.
The noise program had to be delayed until September, to be flown in Granada after all.
Due to the engine problems, it was too late for the hot weather program to find the required "hot" conditions (40 deg. C plus) on this side of our planet and those tests were postponed till March 1987. Senegal in Africa was then selected and there the right temperature was found. In the meantime the cold-weather conditions in Scandinavia had nicely lined up with the delays and those tests were done in the winter period.

Due to the delays and the growth of the flight test program, the Fokker 50 progress had not been sufficient to avoid, as originally planned, interference with the Fokker 100 flight test program that started on the 30th of November 1986. Also because of this interference, the flight test production rate of the Fokker 50 went back from the maximum average of 14 hrs/ac/wk in the 4th and 5th month of the program to 7 hrs/ac/wk in the second half of the program. This was acceptable because delivery of the Fokker 50 was by then delayed to July 1987 with basic type certification in May 1987. The Fokker 100 got priority consuming some capacity from the Fokker 50, which got worse when the second Fokker 100 started flying in February 1987.
Fortunately by then the bulk of the Fokker 50 program had been finished.

One of the major goals of the Fokker 50 flight test program was to certify the aircraft with as few concessions as achievable: although the first customer would get his aircraft late, he should get what he had ordered!
It was quite obvious that, as soon as the Fokker 50 would be certified, the Fokker 100 would absorb all available flight test supporting capacity, leaving hardly any capacity for product improvement activities on the Fokker 50 for at least the first six months of operation. So the first customer should get a good aircraft without any operational restrictions right from the start.
This goal was achieved: the basic certificate of the Fokker 50 was very complete, leaving hardly any concessions with respect to the original design specification. Virtually all problems had been solved. However, this approach was partly responsible for the program overrun, both in hours and in time.
On the other hand, during the first six months, after the basic type certification of the Fokker 50 only some 70 hrs of additional test flying were necessary for product improvements and customer options.

The resulting progress of the Fokker 50 flight test program as compared with the planning, can be presented as follows:

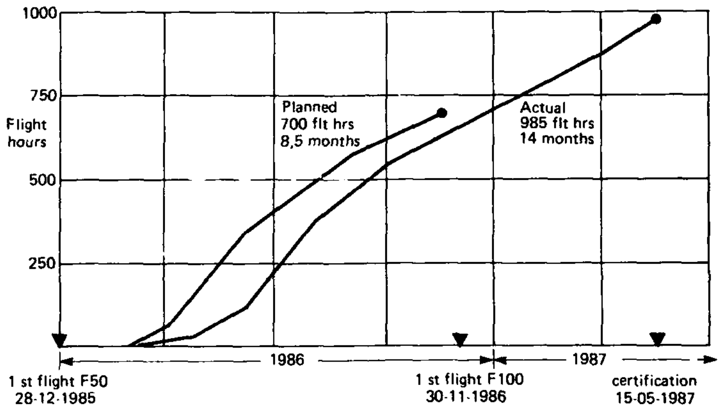


Fig. 3.1.2 The Fokker 50 flight test program, planning and reality.

The figure clearly illustrates the delay and growth of the program.

The corresponding program split-up in flight hours looks as follows:

	planned		actual
	(1983)	(1985)	(15-5-87)
flight envelope	10	20	18
flutter/aeroelasticity	10	10	6
flight handling	60	110	230
performance	190	200	265
propulsion	40	60	176
avionics	60	140	145
hydr/mech systems	5	20	23
airco & anti-icing	20	55	52
noise & vibration	100	75	63
loads	5	10	7
total	500	700	985

The most significant exceedances from the 1985 planning are obvious: flight handling and propulsion, which agrees with the major problem areas encountered in the program. The exceedance in performance test hours is a secondary effect of the same.

The Fokker-100 is a derivative of the F28 Fellowship.
It is a tail-mounted twin-engined short-haul jet aircraft with seating capacity for 108 passengers. (Fig. 3.1.3).

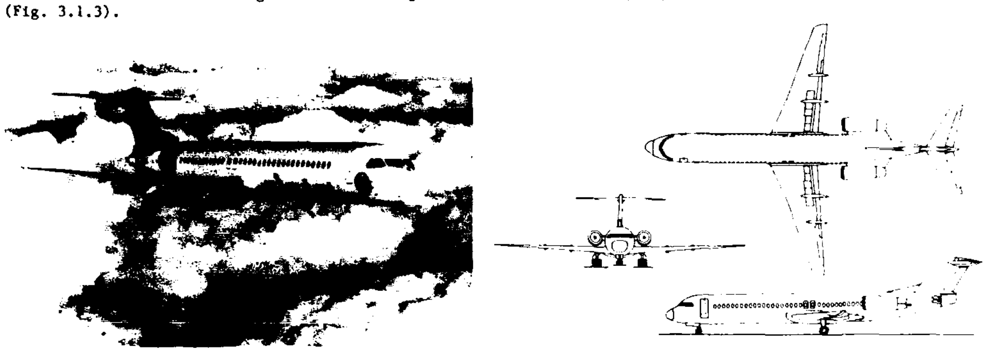


Fig. 3.1.3 The Fokker 100.

From the start of the program, the Fokker 100 was designed as a fully modernised derivative of the F28. The major aerodynamic changes were:
- the fuselage was stretched with 5.9 m,
- the wing was modified extensively: only the basic wingbox of the F28 was retained, but the planform was changed and the span was increased; the forward and aft wing sections were changed to achieve a super-critical wing profile; the ailerons and flaps were redesigned,
- the horizontal tailplane was enlarged.

The major system changes are limited in number, but are very significant:
- the RR Spey engines of the F28 were replaced by RR Tay engines, giving some 50% more thrust than the Spey at considerably increased fuel efficiency,
- All avionics were replaced by Arinc 700 digital equipment; the aircraft got a new automatic flight control and augmentation system (AFCAS) capable of CAT 3b landings, a flight management system (FMS) and autothrottle system were introduced, an EFIS and multi-function display system (MFDS) incl. flight warning system (FWS) were installed,
- a glass cockpit was introduced, incorporating 6 CRT's (4 for EFIS and 2 for MFDS) replacing conventional instruments, designed in accordance with the "dark cockpit philosophy", adopted by Fokker for both the Fokker 50 and Fokker 100,
- carbon brakes were introduced.

All other systems were copied from the F28 and only changed to a limited extent.

In 1983, when the Fokker 100 program was launched together with the Fokker 50, the flight test program was planned for 800 hrs, to be flown in 10 months with 2 prototypes.
In 1986, before the start of the flight test program, due to further development and revised opinions the program up to basic type certification had increased to 1084 hrs in 11 months, to be flown with two prototypes and, during the last few months,one series aircraft. As a result, the required flight hours production rate had increased from 9 to an average of 10 hrs/ac/wk, expected to be an acceptable average.

For preliminary evaluation of AFCAS hard- and software, the original prototype of the F28 was extensively modified to serve as an avionics flying testbed, supporting the Fokker 100 AFCAS program in an early stage of the program (Fig. 3.1.4)

Fig. 3.1.4 The F28 Avionics Test Bed.

Thus, early AFCAS software was tested in flight about a year before the first flight of the Fokker 100. This resulted in a true automatic landing on that first flight. The avionics flying testbed was also used as an instrumentation testbed, testing various new MRVS units (the Fokker/NLR instrumentation system, see chapter 4).

Early in the planning of the Fokker 100 flight tests, it was understood that maximum flexibility was required. Thus, both test aircraft were identically instrumented, except for a few extreme cases. This resulted in two virtually similar prototypes. This made it possible to shift programs from one aircraft to the other. Disadvantage of this approach was the fact that both aircraft were relatively heavy. Some tests, to be flown on low weight, had to be repeated on the first series aircraft. Experience with the Fokker 50 had taught to reserve adequate spare capacity in the system to cope with additional recording requests.

Despite significant delays in the production phase, considerable effort and time were spent to achieve a first flight with a virtually complete aircraft and complete instrumentation system, allowing good flexibility and thus progress, right from the start of the program. This lesson was learned from the Fokker 50 program.
In the first few months the flight test production rate was 12 hrs/ac/wk, while 16 hrs/ac/wk was achieved as a program maximum average over the 5th and 6th month of the program, considerably above the planned average of 10 hrs/ac/wk.

Because the Fokker 100 was considered an almost entirely new aircraft from the beginning, with major aerodynamic and system changes, even more precautions had been taken to assure a smooth flight test program.
Extensive windtunnel testing had been done for the aircraft and especially the new wing. The new engine, the Rolls Royce Tay, had been tested minutely by Rolls-Royce and was flown by Gulfstream on their new Gulfstream VI aircraft about a year before the Fokker 100.
The all new digital avionics system had been flight tested a year before the first flight of the Fokker 100 on the avionics flying testbed, the F28 prototype, and although not fully representative for the Fokker 100, it proved to be a very useful exercise. Apart from that, engineering simulators were available to test and fly all AFCAS-, EFIS- and FMS-software versions.
Right from the start of the flight test program, all went quite smooth. The aerodynamics of the aircraft agreed very well with the windtunnel results; only a few minor changes were required allowing completion of the flight handling certification program 4 months before certification date.
The engines behaved superbly, giving a lot of confidence to the crew in all flight conditions.

The first flight of the Fokker 100 ended with a hands-off Autoland, and although many AFCAS problems were still to be solved, it all looked very promising right from the start.
Despite the 6 weeks late arrival of the second prototype, the progress with proto-1 had been so good that within a few months this draw-back had been more than compensated.
In March 1987, proto-1 went to Granada on schedule to stay there for 3.5 months to complete the runway performance and noise program in time. The hot weather program was flown on time in August in Tunis, North-Africa. Obviously there were problems, but most of them could be solved and none of them disrupted the flight test program seriously.
A set-back occurred in August 1987. During a high-speed landing to simulate an aborted take-off and investigate the behaviour of the thrust reversers at that high speed, heavy shimmy of the right hand main undercarriage occurred. The torque-link appeared too weak and was torn loose whereupon the sliding member with the wheels broke off. The aircraft came to a stop sliding on the remaining stub of the gear and on the wingtip. Repair of the wing (the undercarriage could be replaced easily) and strengthening the torque links took about 6 weeks. An extensive program to investigate the shimmy problem took another few weeks. Certification therefore slipped a few weeks to 20 November 1987.

In general, the Fokker 100 flight test program went quite well as is clearly shown by the progress line as compared with the planning:

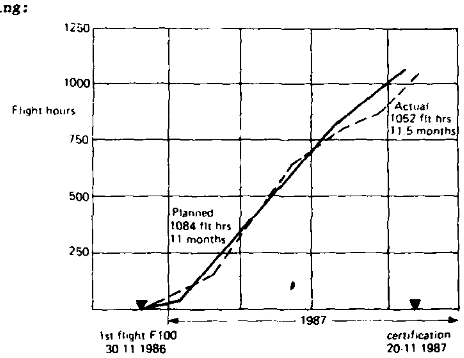


Fig. 3.1.5 The Fokker 100 flight test program, planning and reality.

The temporary set-back in August, following the shimmy incident, can clearly be recognized.
The program split-up in flight hours looks as follows:

	planned		actual
	(1983)	(1986)	(20-11-87)
flight envelope	30	35	20
flutter/aeroelasticity	10	10	6
flight handling	80	253	265
performance	200	228	224
powerplant, incl. APU	100	82	75
avionics	170	291	227
hydr/mech systems	20	35	72
airco & anti-icing	20	42	43
noise & vibration	130	57	80
loads	20	27	29
cockpit design	20	24	11
total	800	1084	1052

Additionally, 41 hours were made for ferries, demo's, etc., adding up to a total number of 1093 flight hours at certification date.
Most of the realized hours are remarkably close to those planned in 1985. The avionics hours are significantly less due to the decision to leave the bulk of the Autoland program till after the certification date.

At this very moment flight testing with the Fokker 50 and Fokker 100 still goes on. The carbon brakes for example, did not perform to specification and had to be modified and tested a second time. The Fokker 100 Autoland program, leading to a cat. 3B certificate, requiring some 300 automatic landings, was flown from November 1987 till March 1988. Higher thrust engines (for USAir) have been installed on the Fokker 100 and a large number of new customer requirements on both Fokker 50 and Fokker 100 have to be tested while production accelerates.

3.2 Store Certification Flight Testing

For more than two decades the National Aerospace Laboratory NLR has been involved in supporting the Royal Netherlands Air Force (RNLAF) in the clearance of new store configurations of the Northrop NF5 and the General Dynamics F16 (Fig. 3.2.1).

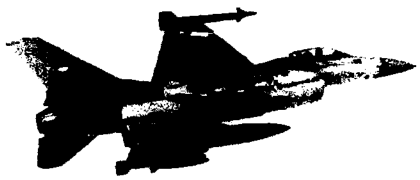


Fig. 3.2.1 Instrumented F-16 aircraft of the Royal Netherlands Air Force prepares for a tank seperation test.

The necessity for these store clearances originated from the requirements of the RNLAF to use weapons from existing inventories or the introduction of new weapon racks and/or new weapons which were not yet cleared by the aircraft manufacturers or other authorities.
The RNLAF could have applied to the aircraft manufacturers for additional clearances of course, but instead the alternative way of involving NLR was chosen mainly for reasons of costs and the need to have expertise "round the corner" readily available .
The clearance activities concern store separation, performance, loads, flutter, stability and control.

A full description of the flutter clearance capability can be found in Reference 3. It treats the following subjects:
- aeroelastic calculation methods,
- unsteady aerodynamic computer codes, including store aerodynamics,
- influence of store aerodynamics and wing transonic effects on flutter characteristics,
- store flutter clearance procedures,
- results of flutter and dynamic response calculations,
- ground resonance tests to validate structure modeling,
- flight tests with and without telemetry.

Aero-elastic analyses (Fig. 3.2.2) and if necessary flutter tests will be carried out to determine the safe envelope with respect to flutter and limit cycle oscillation.
If the results of all analyses show safe separation throughout the flight envelope, flight testing is conducted only as necessary to validate predictions. If results show marginal or unsafe areas of the flight envelope, flight testing is started with an instrumented aircraft.

The remainder of this chapter concentrates on the store separation tests.
When a new certification requirement is received, an analysis is carried out as a first step which compares the new configuration with similar, certified ones. To facilitate this, an extensive aerodynamic data base of already certified stores has been build throughout the years. If a new store fits within the analogy criteria, no further analyses are performed and only limited flight testing or none at all may be conducted. If a store doesn't fit, then potential problem areas are identified and required specific analyses and test are determined.

NLR can predict store trajectories using theoretical, grid, flow angularity and freedrop methods. When wind tunnel testing is required, NLR prefers use of the grid method. This is because grid data can be used off-line to perform trajectory analyses. Trajectories are calculated using a six degree of freedom computer program called VORSEP. VORSEP accepts aerodynamic parameters as inputs. The model can be operated in two ways: (1) to predict store trajectories when aerodynamic coefficients are obtained from theoretical studies, wind tunnel tests, or from tests with the NLR full scale captive store load measuring system (described in the subsequent paragraph), and (2) to determine aerodynamic coefficients from store trajectory data measured in a wind tunnel or from full scale store separation tests. In these cases, the model initially uses predicted coefficients to produce a predicted trajectory and the coefficients are adjusted until the predicted and actual trajectories coincide.

In addition to the above, NLR has developed, and validated, a unique, full scale flight test captive store load measuring system (Ref. 4). This system consists of a support structure suspended from a bomb rack, a five component load measuring balance, and a replaceable store shape (which is made as light as possible to minimize inertial forces). The system is designed so that in-flight airloads may be measured with the store in a captive carriage position and in a displaced position (with a spacer placed between the store and the carriage rack). Figure 3.2.3 shows an NF-5 test aircraft with a fuel tank mounted on the captive store load measuring system in the displaced position.

Fig. 3.2.3 NF-5 test aircraft with fuel tank mounted on NLR captive store load measuring system in the displaced position.

Selection of displacement values is based on a study showing that interference aerodynamic forces decay rapidly to small values by the time one store diameter is reached. Three missions are usually required to gather store airloads data for each configuration, one mission with the store in the captive carriage position and two missions with the store in displaced position. Store airloads are subsequently used in the six degree of freedom computer program VORSEP to predict store separation trajectories. Data from a number of tests show that store separation trajectories based on flight test full scale captive loads are far more accurate than theoretical or wind tunnel based predictions. The system is particularly suited for our use, because in The Netherlands many stores are of the low density, unguided variety, carried on parent pylon and multiple carriage racks.

Besides the prediction of store trajectories to identify the safe, marginal and unsafe areas of the flight envelope, also the physical compatibility of a store to an aircraft has to be established. Loading and fitting trials are executed if compatibility is not obvious. Subjects such as aircraft/store interfaces (hardware and signal exchange) and operational aspects are also considered.
New stores will change inertia as well as aerodynamic loads on the airframe. Ejection forces and dynamic structural responses, especially in case of ripple releases, will also differ from certified cases. Therefore the structural integrity will be investigated.

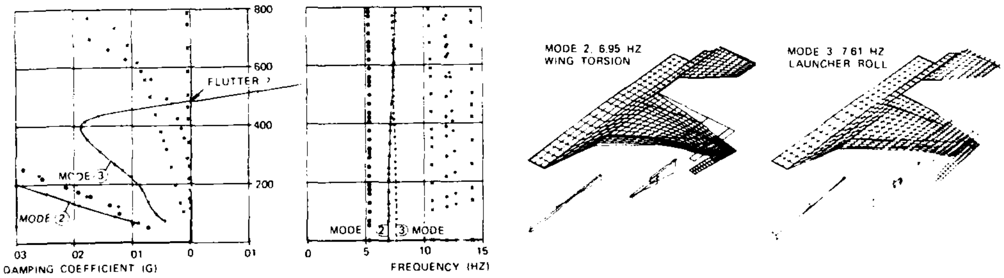


Fig. 3.2.2 Calculated frequencies and damping coefficients as function of airspeed for symmetrical vibration modes. At about 500 knots flutter might occur; the figure shows the two involved mode shapes.

The separation tests start at a point judged to be safe. If the separation as reconstructed from the flight test (fig. 3.2.4) differs significantly from the predicted trajectory, aerodynamic load data are extracted from the actual results and used to update the predictions. This iterative procedure cuts down the number of flight test required. The process is continued until separation envelope goals have been achieved.

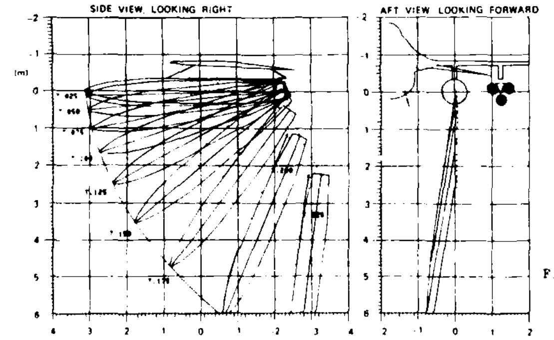


Fig. 3.2.4 Two view representation of a seperation of a fuel tank as reconstructed from a flight test.

3.3 Helicopter-ship qualification testing

The operation of a helicopter is restricted by limitations, as specified in the Flight Manual. Apart from the engineering limitations (weight, center of gravity, airspeed, power, etc.), also some operational limitations such as flight in icing conditions and landings on slopes are given. For successful operations on board ships, additional limitations and special procedures are required. These are not given by the manufacturer, since they depend to a large extent on the ship and its environment (Fig. 3.3.1).

Fig. 3.3.1 The ship, the helicopter and the environment.

During about 25 years NLR has gained much experience in the determination of limits for helicopter operations on board ships. Test programs were carried out successfully for the Royal Netherlands Navy, the Peruvian Navy, the Royal Norwegian Coast Guard/Navy and other foreign navies. (Ref. 5).

Prior to the helicopter flight tests on board the ship, the following investigations are carried out:
- windtunnel tests on a scale model of the ship to determine airflow characteristics near the flight deck;
- airflow tests on board the ship to verify the results of the windtunnel tests;
- shore-based helicopter flight tests to determine and to verify the helicopter limitations as specified by the manufacturer (e.g. cross-wind limitations and power required).

Helicopter operations from ships are affected by various factors, which are investigated in the Low-Speed Windtunnel of NLR (Fig. 3.3.2).

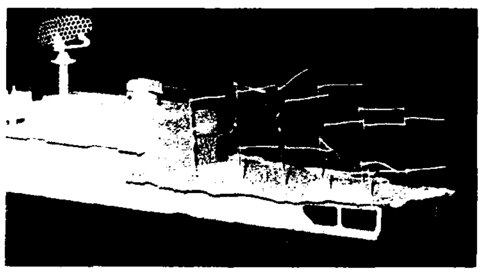


Fig. 3.3.2 Investigation of the airflow near the flight deck of a ship in a wind tunnel. Airflow is visualised by means of tufts.

- Flow deviations and turbulence caused by the ship's superstructure.
  Observations of the airflow in the neighbourhood of and above the flight deck are made and measurements of flow deviations and turbulence with hot-wire anemometers are taken.
- Smoke at or over the flight deck and at the helicopter approach path.
  Exhaust plume trajectories are measured for all relevant relative wind directions and velocities and at various ship power settings. In addition, in case of gas-turbine-powered ships, the shape and location of the high-temperature plume and its border regions are determined.
- The airflow at the (proposed) location of the wind measuring gear of the ship. The quality of the installation, especially with respect to ambiguity, is evaluated and the relationship between the true relative wind and the indications obtained is established.

A benefit of wind-tunnel investigations is the early detection of unfavourable conditions. It also provides an opportunity to investigate the effects of modifications of the ship.

To verify the results of the windtunnel tests for the flight-deck environment, some full-scale airflow measurements on board the ship are carried out. With a special movable mast, equipped with low-inertia anemometers and temperature sensors, wind conditions and plume temperature above the flight deck are measured (Fig. 3.3.3). The relationship between these measured wind conditions and the relative wind indicated by the ship anemometer, is determined.
During these tests the ship motions are measured to determine the dynamic characteristics of the ship, which play an important role in the helicopter-ship operations.
The aim of shore-based helicopter flight tests is to verify and to detail the helicopter limitations concerning, for instance, cross-wind conditions and engine performance, especially during hover (Fig. 3.3.4). Together with the test pilot the results of the flight tests are interpreted, taking into account the control margins needed to counteract the dynamic response of the helicopter due to turbulence in the vicinity of obstacles as present during operations from ships.
This interpretation leads to adjusted helicopter limitations.

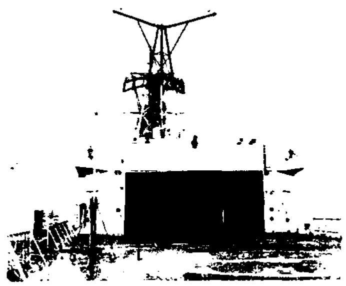


Fig. 3.3.3 Movable mast for full-scale airflow measurements above the flight deck.

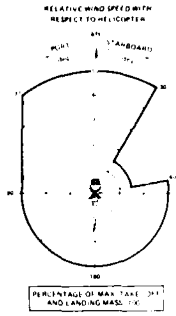


Fig. 3.3.4 Relative wind limitations for helicopter take-off and landing for shore-based operations, as given by the manufacturer.

With the aid of the results of the wind-tunnel tests and the airflow tests on board the ship the wind limitations determined during the shorebased helicopter flight tests are corrected for the readings of the ships anemometer. These limitations are the basis for the flight-test program on board the ship. The flight tests are carried out to determine the effects of the reduced pilot's view over the flight deck, the ship motion, and the turbulence on the pilot's workload, which may result in an adaptation of the limitations.

For the execution of the flight tests, NLR supplies measuring and recording instrumentation (see also chapter 4) for both the ship and the helicopter. NLR generally also provides three members of the test crew, furthermore consisting of members of the contracting agency: the test pilot, a flight-deck officer and a helicopter-directing officer. The NLR test crew is responsible for the set-up and the execution of the flight-test program and handles the test equipment. To this end, an NLR observer is in the helicopter while the others are on the ship.
In general the test crew determines:
- optimal procedures for take-off, approach and landing;
- take-off and landing limitations by day and by night;
- limitations for helicopter handling on the flight deck.

Take-off and landing procedures to be carried out are dependent on the type of helicopter and the flight deck configuration (equipment, dimensions, obstacles). Generally take-offs and landings are executed according to the FORE-AFT procedure, during which the helicopter is always aligned with the ship's centerline and with its nose in the sailing direction. After lift-off, a sidewards flight to a hover position alongside the ship either to port or starboard is executed. From that position a forward flight is initiated. The approach to the ship is terminated at a hover position alongside the ship, followed by a sidewards flight to the hover position over the landing spot on the flight deck from where the landing is carried out.
The take-off is always executed to the windward side of the ship while the landing is preferably carried out from port, because the pilot should have a good view over the flight deck (pilot in command is seated in the right-hand seat).
In case a FORE-AFT procedure cannot be executed because of helicopter cross wind limitations, alternative procedures can be applied (fig. 3.3.5).

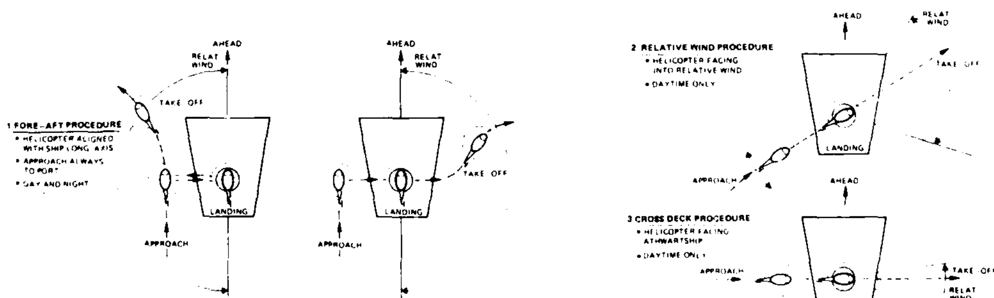
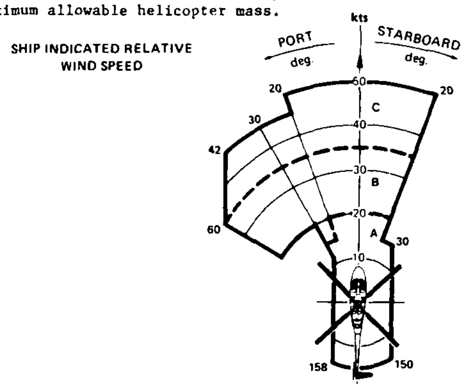


Fig. 3.3.5 Take-off and landing procedures on board the ship.

For safety reasons, the (as yet unknown) limitations are approached gradually. Therefore the flight-test program is divided into several parts with increasing demand on the skill of the pilot and the capabilities of the helicopter.
These parts are:
- tests at low helicopter mass, fair weather, first by day, later on by night;
- tests at maximum helicopter take-off mass, fair weather, day and night;
- tests at low helicopter mass, rough weather and sea conditions, day and night;
- tests at maximum helicopter take-off mass, rough weather and sea conditions, day and night.

The flight tests are carried out up to approximately 50 kts true win: speed.
In the course of the tests situations will be met which prevent continuation of a certain line of investigation. For instance, reaching the power (torque) limit of the helicopter prevents tests at increased weight or at lower relative wind speed. This and other mechanical limits are detected in the helicopter by displaying the value of the relevant parameter and also on the ship by time histories of the parameters obtained on-line through telemetry. In some cases the limits cannot be based on simple measurements, for instance, where turbulence or ship motion makes the take-off/landing too risky. Here the judgement of the pilot is decisive, and adequate experience is required to cope with unexpected situations. In judging the conditions, the pilot must realize that later on pilots with less skill or experience may have to operate under the accepted limits.

At the completion of the take-off and landing tests, a fair idea about the operational limitations has been obtained. For definite results, the measured data together with pilot's comment have to be analyzed in more detail, especially the influence on helicopter performance of turbulence, ship motion, flight-deck configuration and pilot view over the flight deck.
The operational limitations are presented in the form of graphs (Fig. 3.3.6) containing the following aspects:
- take-off and landing procedure;
- limits for relative wind speed and direction;
- maximum allowable ship motion expressed in standard deviation of pitch and roll angle;
- maximum allowable list of the ship;
- maximum allowable helicopter mass.



LIMITS OF HELICOPTER MASS AND SHIP MOTION			
AREA	PERCENTAGE OF MAX TAKE OFF AND LANDING MASS OF HELICOPTER	AMPLITUDE OF SHIP MOTION (STAND DEVIATION)	
		PITCH deg	ROLL deg
A	[95 [98	1 0 0 5	3 5 0 5
B	98	1 5	3 5
C	98	2 0	1 5

Fig. 3.3.6 Take-off and landing limitations for the FORE-AFT procedure by day.

Furthermore relative wind and ship motion limits are given for deck handling operations such as:
- ranging the helicopter from hangar to flight deck and vice versa;
- spreading and folding of rotor blades and
- starting and stopping of rotors.

The results of the qualification testing ensure an optimum operational availability for helicopters on board ships.

3.4 Determination of the Mathematical Model of the Cessna Citation for a Phase II Flight Simulator

The Cessna Citation 500 executive jet aircraft is operated by the Dutch Government Civil Aviation School (RLS) in the final stage of civil aviation pilot training.
In the spring of 1986 the RLS decided to purchase a flight simulator for the aircraft, which should have a Phase II approval. This made it possible to reduce the fleet from six to three aircraft.
Because the Citation 500 was developed in the late sixties no mathematical model and data package was available, which was of such quality that it could be used to obtain Phase II approval. Therefore RLS contracted the National Aerospace Laboratory (NLR) and the Department of Aerospace Engineering of Delft University of Technology (LRT/DUT) to install an accurate instrumentation system in one of the Citation 500 aircraft (PH-CTA) (Fig. 3.4.1), execute a flight test program, analyse the data, evaluate the a priori mathematical models of the aerodynamics, engine, flight control system and landing gear and finally generate the necessary data for these models, based on the results of the flight tests.
A more detailed description of the program and the instrumentation used can be found in Chapter 4 and Reference 6.

Fig. 3.4.1 Cessna Citation (PH-CTA) inflight

The instrumentation system required for flight tests incorporating dynamic manoeuvres must be more accurate than usually is employed with conventional methods. Six more or less independent sensor systems can be distinguished:
1. Inertial reference system (IRS)
2. Air-data measurement system and vanes to measure the aerodynamic angles
3. Transducers for measuring engine parameters
4. Transducers for measuring control surfaces and trim deflections
5. Transducers for the measurement of control forces and deflections and gear parameters such as shock absorber deflections and nose-wheel steering angle.
6. Electrical signals from aircraft systems.

Figure 3.4.2 shows the positions of the various sensors in the aircraft.

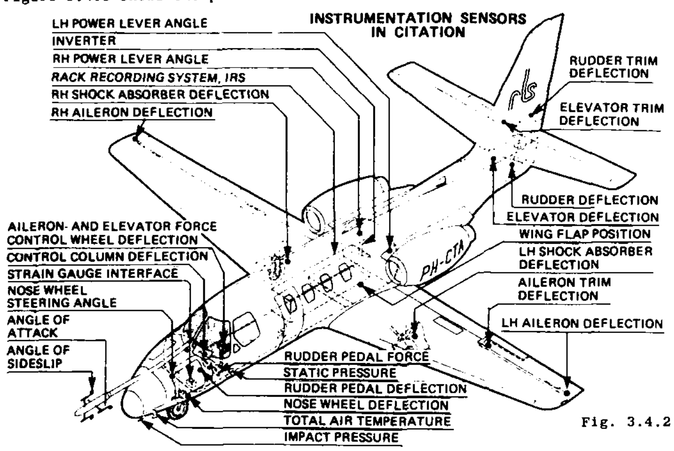


Fig. 3.4.2 Sensors in Citation test aircraft.

The flight test program was drafted with two different objectives in mind:
A. Test flights to obtain data for the evaluation of the mathematical flight simulation model.
B. Test flights in order to fulfill a number of requirements of the simulator manufacturer and the FAA. These are the so-called Acceptance Test Guide (ATG) and Proof of Match (POM) requirements.

Five topics had to be covered by the flight test program with respect to the mathematical modelling viz.:
1. Aerodynamics
2. Engine dynamics
3. Flight control system
4. Aircraft performance and handling on the ground
5. Flight deck cues, such as the levels of sound, vibration and buffeting present in certain conditions.

Obviously, a comprehensive program would be necessary to acquire the data for the modelling of the topics mentioned under label 1 to 4. The only way to perform this challenging task successfully within the limited time available to execute the flight tests, was the ample use of dynamic flight test techniques in combination with a high accuracy instrumentation system and parameter identification analysis techniques. These new techniques have been developed by LRT/DUT and NLR to reduce the valuable test time while maintaining the same fidelity of the results. They are described in detail in References 7, 8, 9, and 10.

Taking into account the configuration considered, a grid of altitudes and speeds was placed upon the flight envelope of interest. This is schematically shown in Figure 3.4.3.

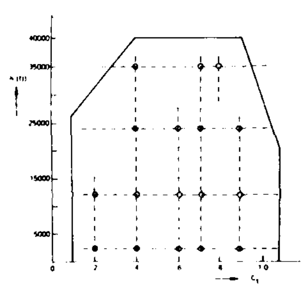


Fig. 3.4.3 Testpoints within flight envelope.

This resulted in a set of testpoints, labelled by a particular configuration, centre of mass, altitude and lift coefficient, at which a train of specific manoeuvres tailored to the various objectives (performance, stability and control, FCS, etc.) is described hereafter. The sequence of the manoeuvres lasted not more than 13 minutes per testpoint. The tests for the aerodynamic modelling can be split up into three parts. Measurements are required with respect to:
1. Performance model
2. Stability and control model
3. Stall, ground effect and buffet model.

The performance model can be split up in a symmetric and asymmetric part. The latter comprises mainly steady flight conditions in which a sideslip angle is present. The following "manoeuvres" can be distinguished:

1. Quasi-steady rectilinear horizontal reference conditions of the test points and the angle of attack excursions of the dynamic manoeuvres, that are initiated from these conditions.
2. Quasi-stationary flight conditions, during which for each axis separately the appropriate trim tab deflection is slowly increased and decreased. At the same time the steady reference condition is maintained by means of the corresponding elevator, aileron or rudder deflection. The manoeuvre lasts as long as the control forces are considered acceptable to the pilot. In fact here an exchange between trim tab and control effectivity takes place.
3. Asymmetric quasi-stationary manoeuvres. These are nominally rectilinear sideslipping flights with "relatively slowly" varying slip angle, roll angle and heading. Positive and negative slip angle excursions are required.
4. Quasi-steady wind-up turn manoeuvre. In nominally horizontal flight and constant airspeed the roll angle is slowly increased and decreased to approximately 60°. Both left and right turns are executed.

When both the longitudinal and lateral performance models are available, the "performance" envelope of the aircraft has been covered. In order to be able to evaluate the stability and control mathematical model the following manoeuvres were selected.

1. Symmetric non-stationary manoeuvres.
   Hereby the aircraft was excited manually by means of rectangularly shaped elevator doublets, varying in amplitude and time.
2. Asymmetric non-stationary manoeuvres.
   Also here rectangularly shaped aileron or rudder doublets, manually applied, were used, varying in amplitude and time.

The arguments to use these type of inputs rather than more "optimal" inputs were of a practical nature: All other solutions were more costly, a block type input is easy to generate manually and is capable of exciting the aircraft over a rather large frequency range.

Ground effect measurements were executed for three configuration settings with landing gear down. For a number of preselected heights varying from 2 to 10 m above the ground rectilinear flights were executed at constant airspeed and height. During the run small excitations were evoked by means of elevator, aileron and rudder. Both the steady parts of the runs and the excitations can be used to evaluate the ground effect.
In the flight test program also landings were included in which the final parts of the landing could be used for the evaluation of the ground effect. Therefore the aircraft was landed hands-off (as far as possible) at a number of configuration settings and at various constant sink rates.
Stalls were performed for four different configurations at approximately 12000 ft using different entry techniques.
Because it is not a Phase II requirement, no specific tests were planned to determine buffet phenomena. However, during stall and manoeuvres, in particular during some of the elevator doublets with large amplitude, these effects were encountered and logged on the test cards.
One test flight was dedicated to special tests with respect to the engine dynamic responses. The tests included throttle chops at several altitudes and speeds, throttle slams from idle to maximum continuous, small throttle steps, in-flight engine shut-downs and starts at several points in the flight envelope. Finally also constant power ratings were recorded on the ground and video recordings of the engine instruments were made of the engine start-up on the ground.
For the evaluation of the flight control model no specific manoeuvres were planned, because during all manoeuvres performed for aerodynamic modelling the control forces and control wheel and pedal displacements were also recorded. However, control column and wheel sweeps were performed with the aircraft at rest on the ground. These measurements give information with respect to the dynamics of the control system. The flight recordings mainly provide the necessary information for the determination of the hinge moments.
Besides tests performed in the air also tests were executed on the ground in order to analyse the undercarriage dynamics. Apart from the use of the ground rolls of take-off and landing for this, also special taxi trials have been done incorporating turns at different speeds and turn rates as well as left/right braking exercises. Finally shock absorber deflections were measured during static tests under various mass and fuel distributions.
A number of flights was performed to generate the Acceptance Test Guide manoeuvres. Thus it was avoided that the data used for the determination of the models had to be used again in the comparison of the model results with flight data.

Data processing and analysis were performed using the well established techniques developed by LRT/DUT and NLR. As stated before these techniques are extensively published in References 7, 8, 9 and 10.

To gain insight in a preliminary stage it was decided to build an a priori model of the Citation 500, based on the available wind tunnel and flight test information, completed with data of comparable aircraft and engineering judgement. At that stage it was not clear, which terms within the models were relevant for the Citation 500. Therefore rather comprehensive models were developed, including all kinds of non-linearities and dependencies on the aircraft state.
If particular parts of the model appeared to be insignificant or could not be identified from the test data, it would be much more simple to set the corresponding coefficients to zero than expanding the model by means of software changes.
The a priori models were implemented on the moving base flight simulator operated by LRT/DUT. As a result a complete a priori model for the Citation 500 was available preceding the flight test program.
After integration of the model within the simulation software and off-line testing, it was possible to fly an "a priori" Citation on-line with the pilot in the loop.
Although, in principle, no on-line simulation is required for the models and data, it was considered as a very valuable option both for the a priori and final models. In case of the a priori model confidence could already be built up with respect to the flyability and/or functioning of the various models. Furthermore pilots can be familiarized more easily with the manoeuvres that are planned in the test flights.

The values of the coefficients in the a priori models were gradually replaced by values stemming from flight data analysis. It appeared that the implementation of the complete final models in the a priori model structure did not cause any difficulty. In fact the final models were in some cases more simple than the a priori model. (A more complete description of model structures and the analyses can be found in Reference 6).

As méntioned in the beginning the FAA requires Proof of Match (POM) data in order to validate the simulator. For that purpose some specific manoeuvres were flown.
Adopting the same initial conditions for aircraft and atmosphere in the simulation, the input signals of the flight tests (time histories of the control system) are now used to drive the simulation models. The resulting model response can then be compared directly with the flight test time histories.
As an example in figures 3.4.4 through 3.4.6 flight test time histories are presented from the POM data base. Characteristic parameters are shown of a level flight acceleration, an all-engine partial climb and a normal take-off. Furthermore in figure 3.4.7 a characteristic motion is depicted viz. the Dutch roll. In the figures also the simulation responses are depicted (dotted lines) as a result of the flight test measured input signals. As can be noticed a good agreement is achieved. Where appropriate the tolerances are indicated on the plots as defined by the FAA.
The flight tests were performed in February and March 1987. All models were analysed and delivered before the end of June 1987. The simulator itself became operational last summer.

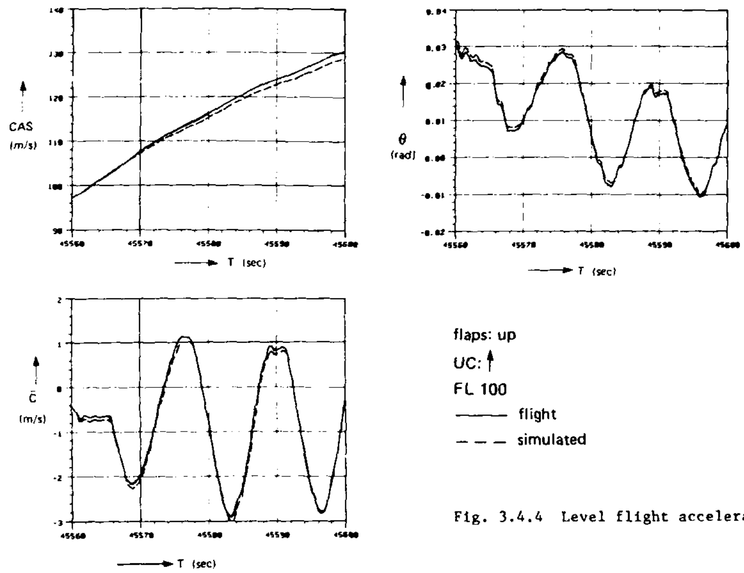


Fig. 3.4.4 Level flight acceleration.

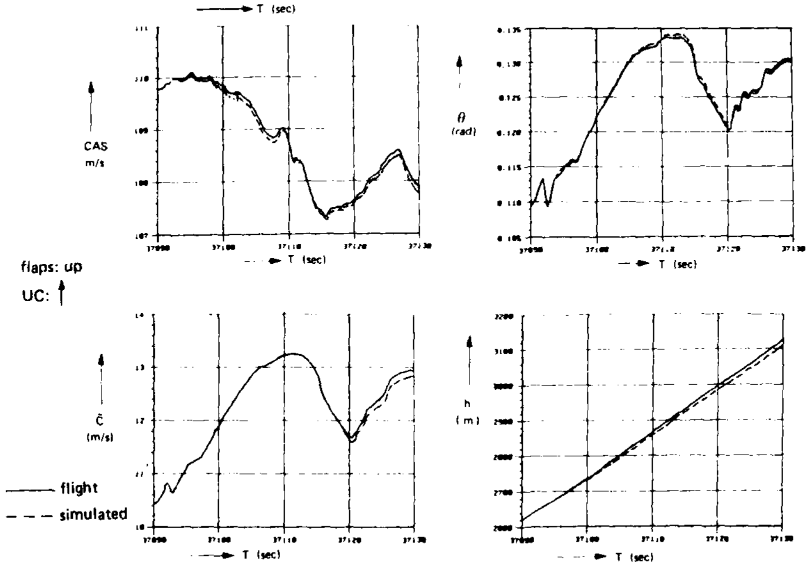


Fig. 3.4.5 All engine climb.

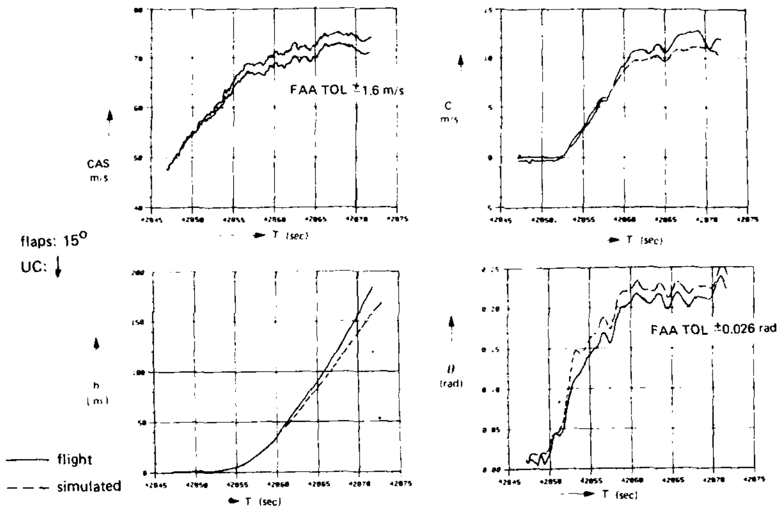


Fig. 3.4.6 Normal take-off.

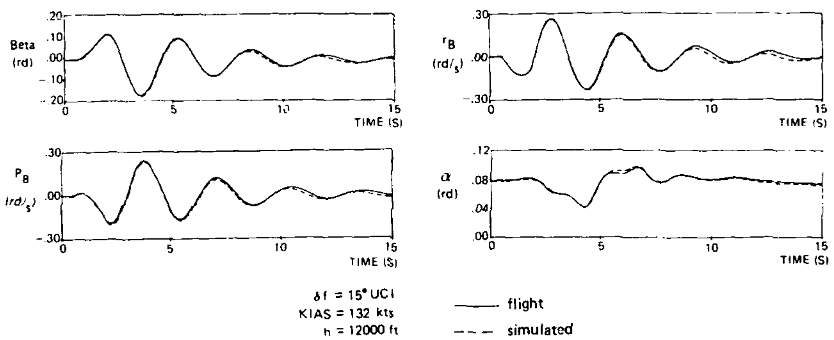


Fig. 3.4.7 Dutch roll dynamics (yaw damper off)
FAA tol: 10% of period
.02 in damping

4. FLIGHT TEST INSTRUMENTATION AND TEST TECHNIQUES

4.1 Civil programs

After World War 2 the flight test instrumentation equipment that was available in The Netherlands was concentrated at NLR. The instrumentation for the flight testing of the F27 Friendship was almost entirely done with NLR-photopanel-recorders. In 1964 instrumentation was required for the F28 Fellowship. This time Fokker Electronics Laboratory provided FM- and UV-recording whereas NLR developed and installed a digital recording systems (50 samples per second!).

In the late seventies Fokker began to think about new generation aircraft and at NLR a new concept for instrumentation emerged. In 1978 Fokker Flight Department initiated a joint NLR-Fokker approach to the integrated development of new instrumentation systems and techniques with data processing capabilities to match. The integrated system to be developed was called MRVS, the Dutch acronym for "Measuring, Recording and Processing System". About 2/3 of the system would be realised by NLR and 1/3 by Fokker Electronics Lab. The primary function of MRVS was to gather and process data from (prototype) test aircraft during flight test programs, aimed at the evaluation and/or certification of new aircraft-types or aircraft systems. MRVS should enable the Fokker Flight Test Department to achieve a short flight test time, both in number of test hours and in duration of the flight test program. From the onset it was clear that MRVS was going to be a powerful, but complex conglomerate of hardware, software, information databases, opera-

ting procedures and last, but not least, people that would make the whole clockwork tick. More specific: the tools that MRVS would provide, are:
- a large quantity of airborne instrumentation, enough to have several flight test programs running at the same time with approximately seven more or less heavily instrumented test aircraft.
- several data reduction stations for digital, analog, vibration and noise data, linked to the Fokker and NLR computer networks.
- central data bases for the storage of flight test data, management-, calibration- and instrumentation configuration data.

The major parts of MRVS are operational since 1983. Although MRVS has been specifically designed for flight testing the Fokker 50 and Fokker 100 prototypes it will be clear that the acquired capabilities can be used in other programs as well. As a matter of fact instrumentation according to the same concept has been used for F16 programs, the described Cessna Citation program and the programs in both NLR's research aircraft.
Figure 4.1.1 gives a general overview of some of the MRVS-features.

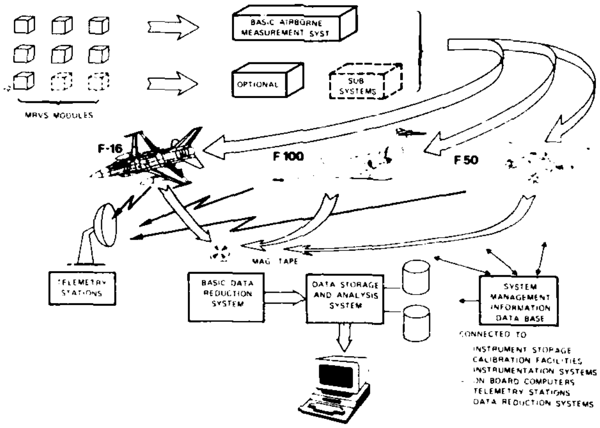


Fig. 4.1.1 MRVS, the Measuring, Recording and Processing System.

Today, all flight test instrumentation and data processing requirements in The Netherlands can be met with MRVS.

MRVS consists of a number of subsystems, some are airborne, others are ground based. Among the airborne subsystems are the digital and analog data acquisition systems, the ARINC 429 avionics multiplexers, the on-board computer, the take-off and landing trajectory measurement system, trajectory measurement systems for Autoland and fly-over noise, multi-sensor aircraft positioning system, telemetry, photo and video systems, interior noise measuring system. More detailed descriptions are given in References 11 and 12.

Among the ground-based subsystems are the telemetry groundstation with tracking L- and S-band antenna, the fly-over noise measuring system, the meteorological station and the video replay stations. Some data processing facilities are also MRVS-subsystems, such as the data preprocessing stations at Fokker and NLR for digital, analog, noise and vibration data and the central data bases (Ref. 13) for flight test data, calibrations, instrumentation configuration and management information. MRVS utilizes the NLR- and Fokker mainframe computer infrastructure for data processing and data transport (Ref. 14).
Both computer networks have been integrated for the rapid transfer of flight test data and the rapid access to flight test and configuration data bases.

4.2 Military programs

Concerning the military programs the following can be added. As touched upon before, the airborne data acquisition system in the military programs (Ref. 2) is, for a great deal, based on the MRVS-concepts. The data acquisition system, capable of recording and telemetering both MIL 1553 A/B mux bus and conventional analog and digital parameters, can be installed in the ammodrum and gun barrel compartment of both modified aircraft (Fig. 4.2.1).
Other instrumentation necessary for the envisaged programs comprises high speed camera installations in a tip launcher with two inward-looking camera's, a centre line pylon with internal camera and a special camera rack with three camera positions (Fig. 4.2.2).

A transportable telemetry ground station with a tracking L- and S-band antenna and real time and quick look capabilities is available for e.g. flutter testing. Expansion of the processing capabilities by incorporation of a powerful Personal Computer is envisaged in the near future. Data reduction and analysis can either be performed in the telemetry ground station at the air base or in NLR's main frame computer.

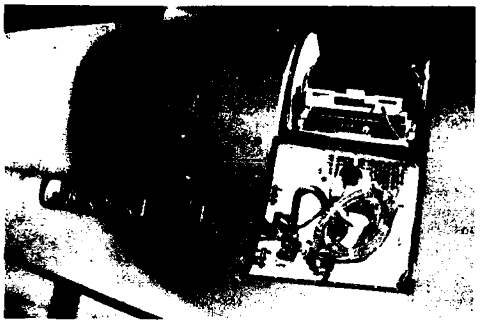


Fig. 4.2.1 Modular instrumentation system in ammo-drum compartment of F-16A with tape recorder on top and "patch panel" in gun barrel space.

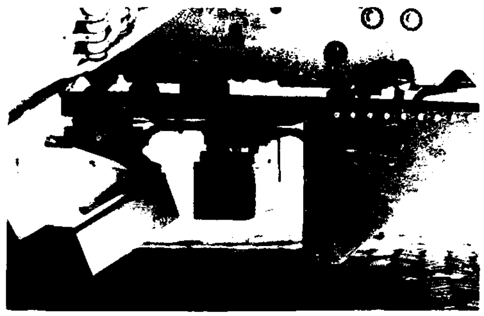


Fig. 4.2.2 High speed film cameras (in protective housing) to record stores separation.

4.3 Take-off landing trajectory measurement system

The first MRVS subsystem to be discussed is the system for the determination of the aircraft trajectory during take-off and landing performance measurements. It was given the Dutch acronym "STALINS" for "Take-off and Landing Measuring System using INS-data".

The design requirements for STALINS were:
- The accuracy of the distance determination during take-off (from standstill until a height of 10.7 m) should be better then 0.1 % (1-sigma) of the distance covered.
- The accuracy of the height measurement during the above mentioned trajectory should be better then 0.15 m.
- It should be possible to extend the measurement of height and distance until a height of 100 m has been reached (with reduced accuracy).
- It should be possible to use the STALINS-method on non-instrumented airfields all over the world. Ground support equipment should be small enough to be carried by the test aircraft itself.
- The measurement results should be available within 24 hours after delivery of the flight tapes to the data processing station.

STALINS' primary sensor is a Litton LTN-76 Inertial Navigation System. This system measures the accelerations in North-South, East-West and vertical directions and it continuously calculates the corresponding velocities and the position of the aircraft in a horizontal plane relative to an earth-bound axes-system. Velocities and distances are obtained by integration of the accelerations.

The horizontal velocity-information from the inertial navigation system itself can very well be used as an input for the STALINS-computation program, provided some corrections are applied in order to obtain the required accuracies for STALINS. This program runs post-flight in NLR's main frame computer. Hence the first integration from acceleration to velocity is left to the inertial system. A number of error sources contributes to the total error in the velocities calculated by the INS. Some are negligibly small, others have to be corrected for. These corrections are applied in the STALINS-computation program, furthermore the corrected North-South and East-West velocities are integrated to obtain distances, covered since the start of the measurement, as a function of time. The integration of the horizontal velocities starts from the point of standstill for take-offs.

Direct calculation of the vertical velocity and height from the INS outputs and the local g-value obtained from outside sources do not provide the required accuracy. This can be solved by calibrating the vertical accelerometer during each measurement. The calibration can be established by comparing the INS-measured height with the actual (geographic) height trajectory of the INS during the ground roll. The latter can be calculated, if (among other things, but the most important of all) the height-profile of the runway (which runway is purely horizontal?) is known with great precision and the aircraft position in the coordinate system of the runway is known. The aircraft position is also required to be able to correctly apply the height-profile of the runway to the aircraft height calculation. The aircraft position can be determined by recording on board the passage of a small radar marker (Ref. 15), placed along the runway at a known point. These markers are small enough to be transported by the test aircraft itself and they can be put into operation within an hour. The runway profile can be measured with standard geodetic survey methods.

The real-time, in-situ calibration of the vertical acceleration makes the STALINS-method unique. It has recently proven its merits many times during the certification program of the Fokker 50 and Fokker 100. The method has been described in detail in Reference 16.

4.4 Non-Steady Measurement method (NSM)

The goal of the NSM method is to determine the performance as well as the stability and control characteristics of an aircraft from dynamic manoeuvres. The method consists of a combination of specially designed flight test manoeuvres, high-accuracy flight test instrumentation and a sophisticated data processing technique, using a Two-Step method. The method was developed by the Department of Aerospace Engineering of Delft University of Technology (LRT/DUT) in the early sixties. In co-operation with LRT/DUT, NLR has developed the method into an operational tool, which has been used in a number of flight test programs.

Conventional methods for the determination of performance require a large number of measurements in carefully controlled steady-state conditions, ranging over all airspeeds, altitudes and aircraft configurations. Even then only a limited number of points on the lift-drag polar curve are obtained. In the NSM method the complete polar curve for a single configuration and altitude is obtained by accelerating the aircraft at a constant power setting from low to high speed. This manoeuvre is designed to accelerate the aircraft very smoothly to ensure that the departure from the steady, trimmed condition is small and the engine is well stabilized. The residual effect of the departure from the steady, trimmed condition is corrected for by using the aerodynamic model of the aircraft determined by the same method, as described below.

The stability and control characteristics are determined from a combination of specially selected control inputs. The reason for this is that in general no single control input sequence allows the determination of the complete nonlinear aerodynamic model. The NSM identification software is designed to allow the easy combination of (parts of) a large number of manoeuvres.

The NSM method requires highly accurate instrumentation in order to allow the accurate reconstruction of the complete aircraft state. Especially the inertial measurements (accelerations and angular rates) are very important, but this requirement is easily met using a modern ring-laser Inertial Reference System (IRS) with special NLR modifications. The accurate measurement of airspeed and altitude is necessary for the performance determination, while accurate measurements of the control surface deflections is required for the stability and control characteristics.

The first step of the data processing starts with the application of the calibrations, the correction for time shifts and the correction for the effect of accelerations and angular rates on the measured variables. In addition, the calculation of the mass, c.g. position and moments of inertia during the actual flight test is very important. Finally the complete state of the aircraft is reconstructed using all available measurements (inertial, air data and vane angles) with a Kalman filter/smoother using a square-root information matrix implementation.

In the second step of the data processing the smoothed recordings are analyzed using a parameter identification program based on linear regression techniques. Since this program is very fast and flexible, it is possible to evaluate a large number of alternative models of differing complexity based on many different (combinations of) recordings. The quality of the determined models can be judged by using the built-in graphical facilities of the program to look at residuals and scatter plots.

The philosophy of the Two-Step method is based on the fact that in actual practice using modern instrumentation systems, the errors in the identification results are mainly caused by the effects of the non-steady atmosphere and by modelling errors. Only a small part of the errors is caused by the errors of the instrumentation system. A consequence of this is that all information needed for the reconstruction of the aircraft state is available in the measured variables and that knowledge of the aerodynamic model will not improve this reconstruction.

In many organizations parameter identification of aircraft aerodynamic models is usually performed using a One-Step method based on output-error techniques. The One-Step method is very time-consuming, because the aircraft state is reconstructed anew for each single parameter identification run. Therefore, it is difficult to evaluate different models and to search for the best parts of the available data in the same way as can be done with the Two-Step method. In addition, because it is based on the theoretical white-noise measurement errors, the calculated parameter confidence bounds are generally very much lower than the actual scatter in the results. In the One-Step method the confidence bounds are calculated using the actual standard deviations of the residuals and are generally much closer to the actual experimental scatter. Nevertheless the One-Step method is used extensively by NLR in those cases where the quality of the measurements or the lack of complete measurements make accurate reconstruction impossible. An example of this is the reconstruction of control force models.

In recent years the NSM-method has been applied successfully in the Fokker 50, Fokker 100 and Cessna Citation flight test programs discussed above. The NSM-method is described in more detail in References 7, 8, 9 and 10.

4.5 Automatic landing flight path measurement system (ALAND)

The Fokker 100 is equipped with a digital Automatic Flight Control and Augmentation System, enabling Cat.3B automatic landings to be performed. In the Fokker 100 flight test program approximately 300 automatic landings had to be carried out, during which the aircraft trajectory and especially the touch-down point had to be determined very accurately. For this purpose NLR developed a trajectory measurement method called ALAND which utilizes a combination of a photogrammetrical method and the inertial method, already in use for take-off and landing trajectory measurements. Figure 4.5.1 gives a schematic representation of the method.

The design requirements were as follows:
- 3-Dimensional aircraft trajectory measurement during approach, landing, roll-out, touch-and-go and go-around.
- Accuracy (standard deviation) from threshold:
  - Position: 0.30 m along-track.
    0.15 m cross-track.
    0.15 m height.
  - Velocities: 0.025 m/s (cross-track and height).
- First trajectory data available within 12 hours.
- Instrumentation transportable by test aircraft.

The ALAND instrumentation consists of a forward looking camera, which takes photographs of the runway lights during the last stage of the landing, an inertial navigation system, a radar altimeter and a pressure encoder. Furthermore the test aircraft has been equipped with its standard flight test instrumentation system. Time correlation between camera and other sensors is accomplished by recording a common IRIG-B time-code signal on film and tape. The number of photographs that have to be read, digitized and processed per landing could be reduced to approximately 7, by making use of the inertial data. The method is described in more detail in Reference 17.

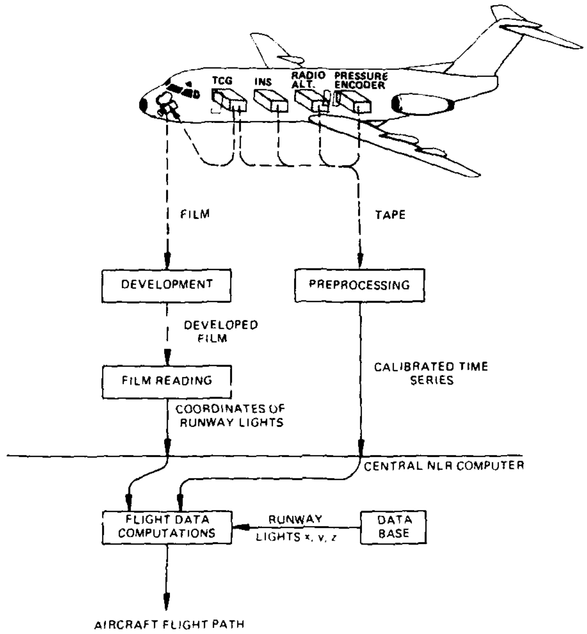


Fig. 4.5.1 General set-up of the Automatic Landing Flight Path Measurement System.

4.6 Multi-DME positioning system (MDP)

The position of an aircraft can be accurately determined by sequentially measuring the distances to a number of DME- or TACAN-stations, if at least two factors are taken into account:

- Since the DME- or TACAN-information supplies the oblique distance between aircraft and groundstation and the desired distance is the projection on the ground of this oblique distance line, a correction for the aircraft height is necessary.
- Between two distance measurements the aircraft has moved to another position and for a correct determination of its position, groundspeed and groundtrack since the previous distance-measurement have to be taken into account.

If this method is combined with an inertial navigation system an accuracy can be obtained of 30 m CEP if at least three DME-stations are being received.

The implementation of this method at NLR is called MDP, the Multi-DME Positioning system. MDP consists of a DME-set and a ROLM 1602B computer. The DME-set on-board the test aircraft is sequentially tuned by the computer to the "visible" DME- or TACAN-stations. After a distance has been read into the computer, it tunes the set to a following station. The position determination algorithm in the computer requires a pressure altitude input from an air data sensor and ground track and ground velocity inputs from an inertial navigation system. In the algorithm the long term drift of the INS can be corrected with DME-data or vice-versa the jittery DME-position-error can be smoothed by the INS-position data. The computer calculates and displays the aircraft position, which is in fact an updated inertial position, and outputs an ARINC-429 data stream with the position information to the Arinc-multiplexer in the data acquisition system.

The MDP-method was used for the performance-evaluation and -certification of the navigation and flight management systems of the Fokker 100 airliner. Besides for flight testing purposes this method is also used in The Netherlands and the USA for calibrating ground based navigation aids. The method is described in more detail in Reference 18.

5. CONCLUDING REMARKS

In this report an overview is given of the flight testing capabilities in The Netherlands and of the central role of the National Aerospace Laboratory NLR in this context. As The Netherlands have their own national aircraft industry Fokker which develops and produces successful civil airliners such as the Fokker 50 and Fokker 100 the accent is more on the civil side of flight testing than on the military side.
Yet there is also a considerable expertise in the relevant disciplines for supporting the military aircraft operators in this country. Generally speaking, the capabilities are much broader than an unprepared observer from abroad might have expected of such a small country.
The development and realisation of completely new systems and methods for data acquisition, data processing and data analysis for the Fokker programs have had and will have a considerable spin-off in the direction of projects for other customers of NLR and Fokker. After the flight testing of these aircraft has been completed the Fokker and NLR flight test organizations will be available, either combined or separately, for future projects.

Acknowledgement

Writing an overview such as this, inevitably involves summarizing and citing original work from others. The authors are therefore indebted to the authors of some of the publications given in the list of References.
In particular they would like to thank Mr. Jan van Twisk, Head of Fokker's Flight Test Department, for his contribution to Section 3.1.

REFERENCES

1. Twisk, J. van — The Fokker 50 and Fokker 100 Flight Test Program. Paper presented at the 18th Annual Symposium of the Society of Flight Test Engineers, Amsterdam (September 1987)

2. Storm van Leeuwen, S. — A small flexible and powerful data acquisition system for the F16 aircraft. NLR MP 85074 (1985)

3. Meijer, J.J. — NLR Contributions to the Flutter Certification of Aircraft with External Stores. NLR MP 86025 U (1986)

4. Alders, G.J. — In Flight Measurements of Aerodynamic Loads on Captive Stores; Equipment and Results. NLR MP 79013 (1979)

5. Hofman, C.F.G.M., Fang, R. — Determination of Limitations for Helicopter Ship-borne Operations. NLR MP 84072 U (1984)

6. Nieuwpoort, A.M.H., Breeman, J.H., Baarspul, M., Mulder, J.A. — Phase II Flight Simulator Mathematical Model and Data-package, Based on Flight Test and Simulation Techniques. NLR MP 88036 U (1988)

7. Wilt, M. v.d. — Flight path reconstruction of symmetric unsteady flights. NLR TR 76133 U (1976)

8. Mulder, J.A., Jonkers, H.L., Horsten, J.J., Breeman, J.H., Simons, J.L. — Analysis of aircraft performance stability and control measurements. AGARD lecture series no. 10[illegible] on parameter identification, 1979

9. Breeman, J.H., Simons, J.L. — Evaluation of a method to extract performance data from dynamic manoeuvres for a jet transport aircraft. NLR MP 78021 U, (1978)

10. Mulder, J.A. — Aircraft flight control system identification. 8th IFAC-IFORS Symposium on Identification and System Parameter Estimation. Beijing, China, (August 1988)

11. Doorn, J.T.M. van, Manders, P.J.H.M., Teunenbroek, O. van, Pascoe, G., Riebeek, H., Voskes, A. — MRVS, a System for Measuring, Recording and Processing Flight Test Data. NLR MP 88034 U (1988)

12. Storm van Leeuwen, S., Voskes, A. — The Data Acquisition System for the Fokker 100 Test Aircraft. NLR MP 87047 U (1988). Paper presented at 18th Annual Symposium of the Society of Flight Test Engineers, Amsterdam (September 1987)

13. Van Teunenbroek, O., Heerema, F.J. — A Database Oriented System for the Support of Flight Tests. NLR MP 87008 U (1987)

14. Pascoe, G. — FDVS, Fokker's Flight Test Data Analysis System. Fokker Report DAS-000-011 (1988)

15. Pietersen, O.B.M. — RASP, a System Generating Very Accurate Position Markers during Flight Testing. NLR MP 86006 U (1986)

16. Kranenburg, C.G., Pool, A., Willekens, A.J.L. — STALINS Method for Take-off and Landing Trajectory Measurements. NLR MP 85057 U (1985)

17. Veen, J.C.T. v.d., Krijn, R. — Trajectory Measurement of the Fokker 100 Aircraft during Autoland Testing. NLR MP 86010 U (1987)

18. Moezel, B. v.d. — Airborne Real-time Multi-sensor Navigation. NLR MP 83013 U (1983)

ON THE DEVELOPMENT OF A BASIC FLIGHT TEST CAPABILITY AND SOME RELATED RESEARCH PROJECTS

by

L.M.B.C Campos, A.A.Fonseca and H.F. Ramos
Instituto Superior Técnico,
Lisbon Technical University,
1096 LISBOA CODEX,
PORTUGAL

SUMMARY

The present paper describes the development of a basic flight test capability in Portugal, through the cooperation between the Portuguese Air Force (FAP) and Lisbon Technical University (UTL), with support from The National Aerospace Laboratory (NLR) in the Netherlands and Braunschweig Technical University (TU-BS) in Germany, under the AGARD support programme for the NATO Southern Flank Nations. After indicating in the introduction (S1) the motivations for this programme, we describe (Part I) the National Flight Test Facility (LNEV), indicating the forms of program management (S2) used in the development of the instrumentation system (S3); we proceed to outline the first research projects (Part II) for which the facility will be used, mentioning in passing application programs, and concentrating on two projects of fundamental reseach on flight in perturbed atmospheres (S4) and non-linear stability (S5). We conclude (S6) by mentioning some of the benefits of the programme.

INDEX

LIST OF ABBREVIATIONS

ADB	Air Data Boom
AFA	Air Force Academy
AGARD	Advisory Group for Aerospace Research & Development
AoA	Angle-of-attack
AT2	Mechanical Hangar
AT3	Electrical Hangar
BA1	Sintra Air Base
C-o-S	Chief-of-Staff
CG	Center of Gravity
DAS	Data Acquisition System
DCU	Digital Control Unit
DE	Electronics Directorate
DMA	Mechanics Directorate
EEZ	Exclusive Economic Zone
FAP	Portuguese Air Force
FMP	Flight Mechanics Panel
FTI	Flight Test Instrumentation
FTTWG	Flight Test Techniques Working Group
[illegible]	Institut fur Flugführung
IST	Instituto Superior Técnico
JNICT	National Fundation for Scientific and Technological Research
LA	Laboratório Aeronáutica
LC	Logistic Command
LNEV	National Flight Test Facility - Portugal
MOU	Memorandum-of-Understanding
MRCP	Main Rack Connector Panel
NATO	North Atlantic Treaty Organization
NDAS	Northrop Data Acquisition System
NDB	National Delegates Board
NLR	National Aerospace Laboratory - The Netherlands
OGMA	Air Force Material Worshops
PCM	Pulse Code Modulation
PDU	Power Distribution Unit
SAU	Straingauge Amplifier Unit
SCU	Signal Conditioning Unit
TUBS	Brawnschweig technical University
UDCP	Under Deck Connector Panel
UTL	Lisbon Technical University

LIST OF SYMBOLS

[illegible]	vertical acceleration
[illegible]	coefficients defined by [illegible]
[illegible]	drag coefficient
[illegible]	drag coefficient due to skin friction
[illegible]	lift coefficient
[illegible]	drag
[illegible]	constants in thrust law [illegible]
[illegible]	disturbance intensity
[illegible]	acceleration of gravity
[illegible]	constant in induced drag
[illegible]	[illegible]
[illegible]	aircraft mass
[illegible]	modified disturbance parameter
[illegible]	radius of turn
[illegible]	[illegible]
[illegible]	reference area
[illegible]	thrust
[illegible]	time
[illegible]	ground speed
[illegible]	horizontal wind velocity
[illegible]	minimum drag speed
[illegible]	airspeed
[illegible]	[illegible] velocity
[illegible]	stalling speed
W	weight
[illegible]	vertical wind velocity
α	angle of attack
α_{eff}	effective angle of attack $\alpha_{eff} = \alpha - \alpha$
α	[illegible]
χ	[illegible]
Γ	[illegible]
γ	[illegible]
Λ	[illegible]
λ	[illegible]
θ	[illegible] $\theta = \alpha - \alpha$
ρ	[illegible]
ω	[illegible]
ζ	[illegible]
[illegible]	[illegible]
[illegible]	[illegible]

S 1 - INTRODUCTION

AGARD has maintained since 1983 a support program to the NATO Southern Flank Nations, intended to develop the scientific and technical capabilities relevant to the defense of these countries within the Alliance. The present paper describes a program (P45) sponsored by the Flight Mechanics Panel (FMP), and intended to give Portugal a basic, but independent flight test capability. Before proceeding to a more detailed description (Part I) of the programme, it may be worthwhile to mention in the introduction the way it was initiated.

This programme of international cooperation arouse, like many others, from an informal exchange of points-of-view, within the auspices of AGARD, in the present case among Portuguese and Dutch members of FMP, namely the first author of the present paper and Mr H. Moelke, who was then the head of the Flight Division at the National Aerospace Laboratory (NLR) in Amesterdam. The NLR is responsable for flight test activities in the Netherlands, both for the Armed Forces and industry. The NLR could transfer to Portugal the flight test instrumentation (FTI), (Table I), which it has used in the certification of civil aircraft (the Fokker F 27 Friendship and F 28 Fellowship) and testing of military aircraft (NF-5, the Dutch version of the Northrop Freedom Flighter), in this period (1980-2) the NLR was starting to develop a new instrumentation system, which would subsequently be used in the certification of a new generation of Dutch civil aircraft (the Fokker 50 and 100) and in the testing of recent military aircraft (the F-16). The Dutch offer was significant in that it gave the opportunity to achieve a basic, but independent flight test capability, which would be self-supporting. There has been contacts with another nation about the transfer of small sets of instrumentation, but this would not give the quantum jump in capability needed to make flight test activities feasable using national ressources.

The utilization of the NLR flight test instrumentation, would require the availability of a suitable aircraft in Portugal. Among the twenty or so types operated by the Portuguese Air Force (FAP), the most suitable was the CASA 212 Aviocar (Figure 1 and Table II) a twin-turboprop light transport, with a 2-ton usable payload, 17,5–m^3 cabin volume and of relatively economical operation. During the same period there were contacts between the FAP and the Lisbon Technical University (UTL), concerning the developement of technological capabilities in Portugal in the field of Aeronautics. As part of these contacts, the Chief-of-Staff (C-o-S) of the FAP General Lemos Ferreira, hosted a visit by the Rector of UTL, Professor Arantes de Oliveira, to the Air Force Academy (AFA), located at the training Air Base in Sintra (BA1), 25 km west of Lisbon. The base also hosts a flight of four Aviocar aircraft, fitted with a comprehensive set of electronics, and used for a wide range of special missions, from aerial photography to surveillance of the Exclusive Economic Zone (EEZ). On this occasion it was suggested that Portugal could acquire a flight test capability by fitting the NLR instrumentation on one of the CASA aircraft. The idea was immediately taken-up by Chief-of-Staff of FAP, who directed the Portuguese National Delegate to AGARD, General Bourbon to support it. General Lemos Ferreira became soon after Chief-of-Staff of the Armed Forces, and the programme has received unfailing support from subsequent Portuguese National Delegates namely General Galvão Borges and Colonel Aldeia Portela.

Thus were paid two basic elements of the programme: (i) the support from NLR, maintained as the head of Flight Division changed from H. Moelke, to H.A. Mooij and J.T.M. van Doorn, all FMP panel members, and acting under instructions from the Chairman and Director of NLR, respectively Professor O.H. Gealach and Ir. J. van der Bliek, both national delegates from the Netherlands, (ii) the continuing cooperation between the UTL and FAP, culminating with the signing of a Memorandum of Understanding on Cooperation in Aeronautics by the Rector, Professor Simões Lopes and the Chief-of-Air-Staff General Brochado de Miranda. The third element was the need to train Portuguese engineers in the use of flight test instrumentation (FTI), involving a significant man-hour cost. Following an interview with Professor Gero Madelung, who is national delegate from Germany, and contacts with Dr. P. Hamel of DFVLR and Professor G. Schenzer of the Braunschweig Technical University (TU-BS), both of whom are FMP panel members, it was agreed that the "Institut fur Flugfuhrung" (IF) of the latter university would train the Portuguese engineers at its own cost. Thus the project P45 to give Portugal a flight test capability was started, with FTI offered by the Netherlands, technical training provided by Germany, an aircraft made available by the FAP, technical direction from the Instituto Superior Técnico (IST) of UTL, and both international support from AGARD and funding from the National Fundation for Scientific and Technological Research (JNICT).

PART I - THE NATIONAL FLIGHT TEST FACILITY (LNEV)

Since the establishement of a flight test facility in Portugal has involved the coordination of several national and international contributions, it may be appropriate to outline the way in which the programme was managed (S2) before proceeding to a description of the instrumentation system (S3).

S 2 - PROGRAM MANAGEMENT

The management of the AGARD program P45 to give Portugal a flight test capability may be described at three interconnected levels: (S2.1) as a sequence of three phases, each consisting of three stages, (S2.2) through the contributions made by each participating institution, (S2.3) by the working arrangements associated with the performance of major tasks.

S 2.1 - PHASES AND STAGES OF THE PROGRAM

The program is organized in three phases, namely: (i) the training of Portuguese engineers on flight test techniques, (ii) the transfer of FTI from NLR in Amsterdam to the Laboratório de Aeronáutica (LA) in Lisbon, (iii) the installation in the aircraft and its verification. Each phase was divided into three stages, and in the following we summarize the activities carried out. We indicate for each stage the date of the relevant report prepared for AGARD by the programme director.

The phase I, concerning the training of Portuguese engineers, involved the following three stages:

- I a (9.2.1986): A two-week introductory course on Flight Test Techniques was given in September 1985 in Lisbon by Dr. P. Vorsmann and Mr. D. Brunner from the IF of TU-BS. The course relied on the experience at TU-BS with a Dornier DO 28 flight test aircraft (since then replaced by a DO 128), and included discussions of which aspects would require adaptation to FAP Aviocar aircraft. The course was attended by 14 mechanical and electrical engineers from IST and FAP, of whom 4 were assigned to the programme.

- I.b (23.2.1986): In the meantime a consultant from NLR, Mr Egter von Wissekerke, came to Portugal in the last week of September 1986, to assess the facilities available to install the NLR instrumentation in the FAP Aviocar aircraft. He visited the Air Force Material Workshops (OGMA), at Alverca, 25 km north of Lisbon, and found the facilities more than adequate for the task. This was hardly surprising since OGMA employs about 3000 people, has facilities to overhaul all of the 20 or so aircraft types operated by the FAP, and derived more than half of its income from work for foreign clients.

- I.c (9.3.1986): The four engineers (2 mechanical and 2 electrical) assigned to the programme at this stage underwent training in greater depth during a 68-day (October to December 1986) mission to the IF at TU-BS. The training included 10 hours flight test time, and a brief visit to NLR in Amsterdam to see the instrumentation due to be transfered to Portugal.

The three stages of phase I were completed in less than a semester.

The phase II concerning the transfer of FTI from the Netherlands to Portugal and its verification also consisted of three phases, each lasting one semester :

- II.a (9.3.1986): The transfer of equipment was prepared by a 1-mounth mission by Prof. J.J.E. Santana of IST and Captain J.A.C. Carvalho of FAP to NLR, where they familiarized with the equipment, in particular the interfaces and documentation which were unique to the FTI instrumentation offered by the NLR. The actual transfer of equipment was somewhat delayed by the need to obtain custom clearances, in spite of the fact that NLR had official permission to offer the equipment to Portugal.

- II.b (9.3.1987): The FTI offered by the NLR was moved to the Laboratório de Aeronáutica at IST, which was created and equiped with support from the JNICT. The offer of equipment by the NLR was substancial (see table I) including a large number of sensors, signal conditioning electronics, tape recorders and quick—look ground stations, to total value when new being in DFL 1 175.700 (roughly US$ 555,000). It took some time to test all this equipment, the conclusion being that it arrived in remarkably good condition, with only very minor, easily replaceable items needing attention. The verification of the equipment and initial planing for the Data Acquisition System were helped by a 1—week consultant mission by Dipl-Ing. V. Brandt of the IF of TU-BS.

- II.c (20.8.1987): the detailed planning for the installation of the FTI in the Aviocar aircraft was prepared by two further training missions by Portuguese technicians : (i) H.F. Ramos and A.A. Fonseca spent 3 weeks in September 1987 at the TU-BS during the period of final check out of the instrumentation fitted to the Dornier Do 128 aircraft (which replaced the Do 28), (ii) A. Arantes spent 2 weeks with Mr. P. Sevenhuysen at NLR in the North Polder training on the selection and use of strain gauges.

These two missions completed the training of Portuguese technicians in Germany and the Netherlands, except for a further 2 weeks of pratical experience on fitting of strain gauges arranged through Mr. R. van der Welde at NLR in September 1988, to complete the training of Mr A. Arantes.

The third phase of installation of equipment in the aircraft, also consisted of three stages of six months each.

- III.a (9.2.1988): The plans for the installation in the aircraft were reviewed during a 2-week consultant mission by Mr. P.M.N. Hollestelle, who had been closely involved with the use at NLR of the instrumentation offered to Portugal.

- III.b (6.9.1988): Three voluminous reports describing and specifing the installation of equipment in the aircraft and its operation were completed by the staff assigned to the project at IST; it received approval by the certification authorities at FAP and passed on to OGMA where construction of assemblies begun.

- III.c (first half of 1989): It is expected that after the installation of FTI in the Aviocar aircraft is complete, a final consultant mission by Mr. P.M.N. Hollestelle from NLR will acompany the last check-outs and first proving flights.

The program P45 was planned in three phases of three stages, each lasting one semester, except for the first three stages which could be carried out in one-semester. It is expected that this 3-1/2 year programme approved by AGARD in 1986 will be completed on schedule in the first half of 1989, with the first flights of the fully instrumented aircraft.

The major tasks are illustrated by a bar chart in table III.

§ 2.2 - CONTRIBUTION OF PARTICIPATING INSTITUTIONS

The preceding summary of the phases and stages of the program has concentrated on the consultant missions supported by AGARD, and mentioned in passing some of the Institutions involved in the programme. A more balanced overview of the main contributions to the programme follows (Figure 2) starting with the international support, and proceding through the national institutions, to the various branches of the FAP involved.

The international and foreign institutions involved were:
- AGARD provided the forum through which the initial proposal for the programme above in contacts at the FMP, and were approved by the National Delevates Board (NDB). AGARD did timely financing of the various consultant missions, and most important, provided a chain of links through which the right people for the job were quickly and efficiently brought to work. The documentation published by AGARD, such as the AG—160 (Flight Test Instrumentation) and AG—300 (Flight Test Techniques) volumes has of course been extensively used, in addition to documentation specific to the equipment offered by NLR and national documentation supplied by members of the Flight Test Techniques Working Group (FTTWG) of AGARD;
- the NLR in the Netherlands offered absolutelly free of charge a comprehensive set of flight test instrumentation, and provided the training of Portuguese technicians and consultant missions in areas specific to this equipment, waiving its man-hours costs on all occasions to date. The support from NLR has been unfailing at all times, even when it had a particularly high workload, during the recent simultaneous flight testing of the Fok r 50 and 100 aircraft. The support from NLR has been extended to areas not originally envisaged, such as the selection and installation of strain gauges,
- the TU-BS in Germany has provided free-of-charge the more extended general training of Portuguese technicians on flight test techniques, and has also supported various consultant missions, these activities complementing very well those of NLR. The TU-BS has given access to its considerable experience on the use of Dornier flight test aircraft, including activities which were latter performed on the Portuguese Aviocar.

The high quality of documentation written by NLR for its FTI, together with the great amount of pratical detail communicated by TU-BS, have been fundamental in planning and carrying out the installation in the FAP Aviocar aircraft.

The three main national institutions involved have been .
- the JNICT has financed the program as a 3-year project for the period 1987–9, under the Program for National Infrastructures for Research and Development. Its funding has allowed the LA to be created, and has covered the costs of some equipment additional to that received from NLR and also some missions not covered by AGARD, as well as aquisition of services other than those of UTL and FAP staff which have worked on the programme without material benefit;
- the UTL, through the IST, has provided the work of staff of the Applied Modelling Group at the Department of Mechanical Engineering and Systems and Electronics Group at the Department of Electrical Engineering. This work includes the testing and checking of instrumentation, the setting-up of a ground simulation bench, the preparation of manuals describing ans specifing the installation in the aircraft and operation of the system and overall management of the program;
- the FAP has supported the program at all levels, from the contribution to management made by the Portuguese national delegates, coordinator and their staff, to the assignements of pilots and technicians from various branches, the allocation of the aircraft and support equipment, and the costs of installation of the FTI in the aircraft at OGMA.

It may be worth while to indicate the departments of the FAP more directly involved in the programme, at technical level
- the certification of modifications to the aircraft and contracts for the execution of work at OGMA is the responsability of the Mechanics (DMA) and Electronics (DE) Directorates;
- the installation work is carried out by OGMA at the electrical (AT3) and mechanics (AT2) hangars, on the basily of detailed drawings prepared at the design office, in accordance with the documentation for the FTI and constraints imposed by the configuration of the aircraft;
- the aircraft is operated by operated by Flight 401 at Sintra Air Base, whose pilots have advised on aspects of the data aquisition system related to flying techniques, such as a double-stick (transducer with strain gauges) used to measure stick forces; the flight line support of the aircraft will be provided by a laboratory at AFA, which is co-located with Air Base 1 at Granja do Marquês.

S 2.3 - TASKS OF THE WORKING GROUP

The investiment of manpower and ressources in this programme, both by the FAP and UTL, reach in 1987 a level where it prompted the signing of a Memorandum of Understanding (MoU), by the Rector of UTL and C-of-S of FAP. This event was widely reported in the press, as the MoU foresees a broad range of cooperative activities in aeronautical education, research and development. In order to implement the MoU the C-of-S of FAP and Rector of UTL each named (Figure 3) a three members to a joint steering commitee, which considers new cooperation projects groups. At the time of signing of the MoU the working group LNEV was already in operation, and it appears that the next research and development projects will be in the areas of structures and electronics. The MoU whose initial text was prepared by Professor J.M.G. Sá da Costa, also foresees activities related to the training of aeronautical engineers, which has been a subject of discussion between UTL and FAP, both within and outside the steering group.

The director of the working group on LNEV coincides with the chairman of the steering group from the side of UTL. The working group consists of ten members, representing the organizations participating in the technical aspects of the programme (Figure 4):
- two electrical enginners (H.F. Ramos and A.A. Fonseca) from IST assigned full time to the program, who have developed most of the FTI and its related documentation;
- two mechanical engineers (Professor J.A.C. Azinheira and Mr. A. Arantes) from IST, responsable for mechanical design aspects;
- a mechanical (Cap. Carvalheira) and an electrical (Cap. Damásio) engineer, from the Logistic Command (LC) in charge of certification and contracting;
- a mechanical (M. Domingues) and an electrical (Cap. R.R. Almeida) engineer from OGMA concerned with building up the installation;
- a pilot (Cap. Almeida) from flight 401 at Air Base 1 who will be the leading crew member for the test flights;
- a member of staff (Lt-Col. A.M.S. Cardoso) of the AFA, and also FMP member who acts as liaison to FAP department other than those listed above.

The method of operation of the working group with respect to each major task is as follows (Figure 5).
- the staff at LA establishes the specifications for each item of the installation;
- the constraints imposed by the configuration of the aircraft are assessed with the help of OGMA, and the effects on flying practice, if any, are taken into account;
- advice from the pilot;
- the preceding inputs are used to design the installation at LA, and the relevant documentation is sent to DMA and DE for certification;
- these authorize the contracts for detailed drawing, assembly and installation at OGMA, with final verification by IST;
- in operation the flight support is provided by BA1, the on-the-spot laboratory check[illegible] at AFA;
- simulations in the ground bench and data processing are centralized at LA.

S 3 - INSTRUMENTATION SYSTEM

The selection of sensors and sources of data (S3.2) and the arquitecture of the data aquisition system (S3.3) were performed in agreement with a set of system design principles which are outlined first (S3.1).

S 3.1 - SYSTEM DESIGN PRINCIPLES

The choice and location of sensors was governed by the following sets of requirements, not always easy to reconcile.
- the aircraft should be quickly convertible from flight test to operational configuration: in order to comply with the standard air base procedures used at BA1 the steps to perform this conversation should be simple and not require any special tooling (some sensors and wiring are left permanently in place; the DAS and other sensors are grouped in removable racks);
- the sensors are, as far as possible, independent and non-interacting with aircraft systems and have the best possible location for signal quality;
- the aircraft is fitted with a comprehensive set of standard instrumentation (eg. Inertial Navigation System, Doppler Radar, Autopilot, Weather radar, and so on): in order to benefit fully from this equipment, the FTI is allowed to extract data from aircraft systems (eg. INS, VHF Nav System) but without its operation - this is done by using pick-ups which cannot corrupt existing signals;
- the instrumentation system is modular and allows installation of new sensors and the disconnection of existing ones with a minimum of reconfiguration: this is done by laying the wires through the aircraft, ending by standard circular MIL connectors

(gathered in connector panels) at a few key locations (in the wings and at the fuselage nose and tail). These connect to a major panel under the cabin floor near the location of the DAS rack (UDCP), as shown in Figure 6 - the DAS is also designed for easy reconfiguration interfaces also through standard circular MIL connectors and having modular Signal Conditioning Units (SCU's);
- interference with flying tasks is minimized to avoid cockpit panel congestion and overwork; the Pilot Control Unit (PCU), has only an amber light to warn that the DAS is operating and a switch to disconnect the whole system in an emergency; all other switches, panels and quick look facilities are located in the main rack at the system's operator position (Operator Control Unit (OCU)); control and monitoring capabilities of each crew member are briefly summarized in Table IV.

As an example of the implementation of these principles, we outline the design of the Air Data Boom:
- the Air Data Boom (ADB) carries windvanes for angle-of-attack and side slip, and a pitot tube; the latter can be callibrated using a trailing cone sensor unreeled from the rear of the aircraft;
- the ADB could not be placed at the nose (to avoid interference with the weather radar), nor at the fuselage side (because of propeller slipstream assymetries), nor at outer wing panels (due to corruption of data by aeroelastic effects), nor at the tail (due to interference from wing/fuselage boundary layer at high angles of attack);
- the location chosen for the ADB is an emergency escape roof hatch, just behind the cockpit; since the ADB is 3,0m long, it is braced by two struts connected to the sides of the fuselage nose (Figure 7);
- the emergency hatch remains operational with the ADB boom on it, because the whole assembly is designed to tilt forward out-of-the-way if the roof hatch is opened; the roof hatch was originally designed in the standard aircraft for emergency exit on the ground only;
- the aircraft is converted from the operational to the flight test role by replacing the standard roof hatch by the one with ADB and struts: the whole operation involves only one electrical connection, four screws and two latches
- the length of the boom was dictated by the avoidance of resonances due to atmospheric excitation and the requirement of structural stiffness for the given weight and distribution of sensors; its length is sufficient to give relatively undisturbed air data, and no significant reinforcement of the roof hatch is needed;
- the boom may be extended or shortened, by the insertion of interchangeable tube sections, to give lengths between 2 and 4 meters, for particular types of flight test work.

S 3.2 - SENSORS AND INSTALLATION IN THE AIRCRAFT

The ADB is one of the sensors packages on the aircraft, chosen here to illustrate the design principles indicated above. Similar principles were used, as applicable in connection with other items, whose distribution around the aircraft is indicated in Figure 8. Besides the ADB these include:
- a rack containing three rate gyros and three linear accelerometers near the C.G., under the floor of the cabin; the location is near to the platform of the INS of the aircraft;
- two sets of strain gauges (and charge amplifiers in the Straingauge Amplifier Unit (SAU) located as near as possible of the gauges) are set up in the engine mountings, whose 8-bar configuration (Figure 9) lends itself to the measurement of forces and torques: this would allow the in-flight measurement of thrust, provided certain corrections (e.g. nacelle drag) are made;
- the stick forces are measured by a dummy stick with strain gauges on the test position (starboard) to measure elevator and aileron forces and strain gauges in the test pilot rudder path to measure the rudder force (Figure 10): the pilots find it acceptable to use even for take-off and landing, with the safety pilot retaining the unmodified control column and rudder pedals;
- the position of the control surfaces (left and right ailerons, rudder and elevator), are measured by synchros, connected by frangible couplings to actuation rods;
- navigation and flight path data is extracted from the VHF Nav. System (VOR/ILS), INS, Radio Altimeter, Flight Director and Autopilot;
- the propulsion data is extracted from the aircraft's instrumentation.

The wiring from the sensors is collected in the four connector panels and make use of the test wires which are ended by standard circular MIL connectors (receptacles) in a connector panel under the cabin floor (UDCP1), near the DAS rack location, a separate connector panel, (UDCP2), is used to route the data extracted from the aircraft propulsion and navigation systems, whose wires also end in standard circular MIL connectors; some sensors, such as the ADB, the trailing cone and the C.G. rack do not use the test wires but have their own receptacles: this means each parameter has a specific connector which allows the easy connection of new sensors and disconnection of existing ones by simply wiring the correspondent connector of the UDCP1/UDCP2/transducer to the DAS input connector, using standard wires ended by two plugs.

The navigation, propulsion, strain gauge, control and air data sets add up to a list of 65 parameters currently measured, whose ranges are indicated in Table V. While the number of parameters is modest by modern standards, it does comes from a variety of sources, has a relatively high sampling rate (139Hz) and exploits most of the available data recording capability.

S 3.3 - THE DATA AQUISITION SYSTEM (DAS)

The data aquisition system interfaces with all the sensors indicated in S3.2, and its block diagram is shown in Figure 11. It is a set of units grouped in a stand-alone removable rack, in the port side of the fuselage cabin. These units perform the tasks of signal conditioning, sampling, coding and data storage for off-line data analysis. The DAS layout is shown in Figure 12.

The rack is surmounted by a large patch panel, the Main Rack Connector Panel (MRCP) with several MIL circular connectors through which individual sensors can be pluged in and control signals are fed to the DAS. The signal conditioning functions are performed by Signal Conditioning Units (SCU1, SCU2 and SCU3) and a Synchro-Digital Converter Unit (SDCU). The actual DAS system input capabilities are despicted in Table VI. The conditioning units output is wired directly to a Northrop Data Acquisition System (NDAS), constituted by two units, a Pulse Code Modulator (PCM) and a Digital Control Unit (DCU) - this system samples, at a fixed rate of aprox. 139Hz, up to 72 analog (±5Vdc) signals with 10-bit resolution or up to 8 digital 20-bit signals (see Table VII).

The output data stream is a parallel 10-bit, NRZ-L code, PCM 10 format plus a frame sync and a serial time history signal which is re-coded to NRZ-M prior to recording in a 16-track Kinelogic Model YE tape recorder, whose specifications appear in Table VIII. Power for the equipment is supplied from the aircraft's +28Vdc secondary (non-essential) bus: inverter, transformers, rectifiers and fuses/circuit breakers are located in the Power Distribution Unit (PDU) and fullfill the power requirements described in table IX.

The major part of the DAS was used by NLR for flight test work on the Dutch Air Force NF-5 Freedom Fighter prior to being offered to Portugal. The documentation prepared by NLR for this program is very comprehensive and allowed the staff on the project to familiarize themselves with the system without difficulty. Most of the sensors offered by the NLR were used in the Fokker F-27 Friendship and F-29 Fellowship civil certification prior to transfer to portugal. The NLR also offered two sets of a quick-look ground system (figure 13 and 14) which allows the plotting of 8 recorded channels at a time. One of the follow-on tasks of the programme will be to develop an interface to feed all 72 channels of information on a standard digital computer for further analysis.

Although the present instrumentation system is intended to give no more than a basic flight test capability, we found it requires over 470 connectors, 3 km of cable, 1000 pages of manuals, 3000 man-hours of design and development work and 4000 man-hours of installation work.

PART II - FUNDAMENTAL AND APPLIED RESEARCH PROJECTS

It is intended that the flight test capability demonstrated with the Aviocar aircraft will be applied to the validation of special equipment pods for fighters operated by the FAP. The FTI for these would consist of a smaller number of sensors with a data aquisition system packed into a smaller volume, by comparison with the Aviocar. Two distinct pod development programmes are under way, but it would be premature to go into detail about these programmes. Instead, we will outline the application of the basic flight test capability to projects of fundamental research in flight mechanics, one concerned with (S4) a severity scale for flight disturbances, and the second a (S5) non-linear model of pitch stability.

S 4 - A SEVERITY SCALE FOR FLIGHT DISTURBANCES

When an aircraft encounters an atmospheric disturbance several flight parameters, e.g., ground or airspeed, angle-of-attack (AoA), and vertical acceleration, change in an interelated way. There is some interest in introducing a single parameter, called the "disturbance intensity", which can be calculated from flight data, and indicates the severity of the perturbation induced by the atmosphere.

S 4.1 - AERODYNAMIC DEFINITION OF DISTURBANCE INTENSITY

The simplest definition of disturbance intensity G would be the relative lift change for flight in a still air L and in a perturbed atmosphere

$$G = \frac{L^*-L}{L} = \frac{L^*}{L} - 1 \qquad (1)$$

The lift in still air is given by

$$L = \frac{1}{2}\,\rho\,U^2\,S\,C_L(\alpha)\,. \qquad (2)$$

where ρ is the mass density, S the reference area, U the groundspeed (coincident with the airspeed in still air) and C_L the lift coefficient, which is a linear function of AoA α away from the stall; the lift in a longitudinal u and transverse w wind is

$$L^* = \frac{1}{2}\,\rho\,\{(U+u)^2 + w^2\}\,S\,C_L(\alpha + \arctan\{\frac{w}{U+u}\})\,. \qquad (3)$$

where the term in the first curly brackets is the airspeed squared and the second curly brackets is the slope of the airspeed relative to the groundspeed. The disturbance intensity is given by:

$$G = \{(1+u/U)^2 + (w/U)^2\}\,\frac{C_L(\alpha + \arctan\{\frac{w}{U+u}\})}{C_L(\alpha)}\,. \qquad (4)$$

for arbitrary wind speeds, horizontal u and vertical w.

If the wind speed does not exceed about 30% of the groundspeed ($u, w, \leq 0.3U$, which is the case of moderate winds ($uw, u^2, w^2 \ll U^2$), the lift change can be obtained by differentiating (2)

$$dL = \rho\,S\,C_L(\alpha)\,U\,dU + \frac{1}{2}\,\rho\,U^2\,S\,\frac{dC_L}{d\alpha}\,d\alpha\,. \qquad (5)$$

with the following interpretation:

-(i) $dL = L^* - L$ is the lift change;

-(ii) $dU = u$ is the change in airspeed, i.e., the longitudinal wind,

-(iii) the lift slope $dC_L/d\alpha = C_L/(\alpha - \alpha_0)$ is, away from the stall, the ratio of lift-coefficient C_L to the effective AoA $\alpha_{eff} = \alpha - \alpha_0$, which is the difference between the AoA α and the angle α_0 of zero lift,

-(iv) the incidence change $\Delta\alpha = d\arctan\{w/U+u\}) \approx w/U$, is due to the vertical wind. The lift change

$$L^* - L = \rho\,S\,C_L\,U\,u + \frac{1}{2}\,\rho\,U\,S\,C_L\,\frac{w}{\alpha - \alpha_0}\,, \qquad (6)$$

divided by (2), specifies the disturbance intensity

$$G = 2\,\frac{u}{U} + \frac{w}{U(\alpha - \alpha_0)}\,. \qquad (7)$$

for moderate winds. For example, for an aircraft landing at a groundspeed U = 60m/s at an effective AoA $\alpha - \alpha_0 = 10°$ (e.g., AoA $\alpha = 7°$ and angle of zero lift $\alpha_0 = -3°$), and encountering a u = -5m/s tail wind and a w = -2m/s downflow, the disturbance intensity is $G = 0.36$, i.e., there is a 36% lift loss. A negative disturbance intensity corresponds to a lift loss, due to a downflow $w<0$ or uncompensated tailwind $u<0$, a positive disturbance intensity corresponds to a gain in lift, due to an upflow $w>0$ or an uncompensated headwind $u>0$.

It is possible to introduce two critical values of the disturbance intensity, viz., the lift losses which would cause an aircraft to stall at take-off speed (G_1) and at approach speed to land (G_2). Let V_S be the stalling of an aircraft:

$$V_S = \sqrt{2W \;/\; \rho \; S \; C_{Lmax}}\,, \tag{8}$$

where the weight coincides with lift $W = L$ in straight and level flight. A change in maximum achievable lift affects the stalling speed

$$\frac{V_S^*}{V_S} = \sqrt{\frac{C_{Lmax}}{C^*_{Lmax}}} = \sqrt{\frac{L}{L^*}} = \frac{1}{\sqrt{1+G}}\,, \tag{9}$$

where the definition of didturbance intensity (1) was used. The stalling speed is increased to unstick speed at take-off $V_S^* = V_2 = 1.1V_S$, for a disturbance intensity G_1 satisfying $\sqrt{1+G_1} = \frac{1}{1.1}$, i.e., $G_1=-0.18$, the stalling speed is increased to the approach speed for landing $V_S^* = V_1 = 1.3V_S$, for a disturbance intensity G_2 satisfying $\sqrt{1+G_2} = \frac{1}{1.3}$, i.e., $G_2=-0.42$. The critical disturbance intensities $G_1 = -0.18$ and $G_2 = -0.42$ represent atmospheric disturbances which, if uncompensated, would cause an aircraft to stall respectively on take-off and landing.

S 4.2 – FLIGHT PERFORMANCE IN PERTURBED ATMOSPHERES

The disturbance intensity, which was defined in terms of aircraft aerodynamics (1), (4) and (7), can be related to flight mechanics (9), in several ways. Consider an aircraft flying straight and level in still air, for which weight equals lift:

$$m\,g = \frac{1}{2}\,\rho\; S\; U^2\; C_L(\alpha), \tag{10}$$

if the aircraft encounters a disturbance of intensity G, the groundspeed changes from U to U*, the AoA from α to $\Delta\alpha$, and there may be a vertical acceleration A

$$m\,(g+A) = \frac{1}{2}\,\rho\; S\; U^2\; C_L^*(\alpha+\Delta\alpha), \tag{11}$$

where the lift coefficient is affected both by the disturbance intensity G and change of incidence:

$$C_L^*(\alpha+\Delta\alpha) = (1+G)\; C_L(\alpha+\Delta\alpha) = (1+G)\; C_L\left\{1+\frac{\Delta\alpha}{(\alpha-\alpha_0)}\right\}, \tag{12}$$

Substituting (12) into (11) and dividing by (10) we obtain the formulas for the disturbance intensity:

$$1+\frac{A}{g} = (U_0/U)^2\;\left\{1+\frac{\Delta\alpha}{(\alpha-\alpha_0)}\right\}\;(1+G), \tag{13}$$

for perturbations or straight and level flight, involving simultaneously changes in groundspeed U to U_0, AoA $\Delta\alpha$ and vertical acceleration A.

The formula (13) has simple interpretation in the cases where only one flight parameter changes at a time:
-(a) if the aircraft flies through the disturbance at constant groundspeed $U_0=U$ and constant AoA $\Delta\alpha = 0$, then it experiences a vertical acceleration:

$$U_0 = U,\;\; \Delta\alpha = 0 \qquad A = G\,g\,, \tag{14}$$

whose value in g's is the disturbance intensity G, this could be used to define the disturbance intensity;
-(b) if the aircraft flies through the disturbance at constant AoA $\Delta\alpha=0$ and without vertical acceleration $A = 0$, there is a change in groundspeed:

$$\Delta\alpha = 0 = A \qquad U_0 = \frac{U}{\sqrt{1+G}}\,, \tag{15}$$

i.e. a decrease in groundspeed $U_0<U$ for positive disturbance $G>0$ which increase lift, and an increase in groundspeed $U_0>0$ is needed to compensate a negative disturbance $G<0$ which decreases lift, the formula (15) is the same as for the stalling speed (9), and is plotted in Figure 15;
-(c) if the aircraft flies through the disturbance at constant groundspeed $U_0=U$ without vertical acceleration $A>0$, the AoA changes:

$$U_0=U,\;\; A=0 \quad \Delta\alpha = (\alpha-\alpha_0)\;\left\{\frac{1}{1+G}-1\right\} = -G\;(\alpha-\alpha_0), \tag{16}$$

the change is proportional to the initial effective AoA, and to the disturbance intensity, as shown in Figure 16, but with minus sign, e.g. a negative disturbance $G<0$ or lift loss requires higher AoA $\Delta\alpha>0$ for compensation.

There are also three cases in which only one flight parameter is kept constant:
-(ab) at constant AoA:

$$\Delta\alpha=0 \quad G=(U/U_0)^2(1+A/g)-1, \tag{17}$$

the groundspeed and vertical acceleration specify the disturbance intensity by (17),
-(ac) at constant groundspeed:

$$U_0=U: \quad G=\frac{1+A/g}{1+\frac{\Delta\alpha}{\alpha-\alpha_0}}-1, \tag{18}$$

the vertical acceleration and change of AoA specify the disturbance intensity by (18);
-(bc) at zero vertical acceleration:

$$A=0: \quad G=\frac{U/U_0^2}{1+\frac{\Delta\alpha}{\alpha-\alpha_0}}-1, \tag{19}$$

the groundspeed and change of incidence determine the disturbance intensity.

In general, a disturbance in straight and level flight causes simultaneous changes in groundspeed and angle—of—attack and a vertical acceleration related by (13), which is plotted in Figure 17 in the form:

$$(U/U_0)^2(1+A/g) = p \equiv (1+G)\{1+\frac{\Delta\alpha}{\alpha-\alpha_0}\}, \tag{20}$$

for four values of the modified disturbance parameter p. The formulas and plots above can be used to calculate the disturbance intensity for perturbations of straight and level flight.

S 4.3 - APPLICATION TO STRAIGHT AND TURNING FLIGHT

As a further example, we indicate how the disturbance intensity may be calcutated in level turning flight. In the absence of disturbances, in still air, in a turn at a bank angle ϕ. The radial component of lift:

$$L\sin\phi = m\Gamma = \frac{m\,V^2}{R} = m\,\omega^2 R, \tag{21}$$

balances the centrifugal force, which may be expressed in terms of acceleration Γ, turn rate ω, tangential velocity V and radius R.

In the presence of a disturbance of intensity G, all quantities may change, e.g., the lift L^*, bank angle ϕ^*, turn radius R^*, instantaneous turn ω^*, and tangential velocity V^*:

$$L^*\sin\phi^* = m\Gamma^* = \frac{m\,V^{*2}}{R} = \frac{m\,\omega^{*2}}{R^*}. \tag{22}$$

Dividing (22) by (21) and using (1) we obtain a set of formulas:

$$(1+G)\frac{\sin\phi^*}{\sin\phi} = \frac{\Gamma^*}{\Gamma} = \frac{R}{R^*}\left(\frac{V^*}{V}\right)^2 = \frac{R^*}{R}\left(\frac{\omega^*}{\omega}\right)^2, \tag{23}$$

which specify the disturbance intensity G for perturbation of turning flight. For example, the formula relating accelerations and bank angles:

$$G = \frac{\Gamma^*}{\Gamma}\sin\phi\csc\phi^* - 1, \tag{24}$$

is plotted in Figure 18, and is valid whether the turn is taken at constant rate, constant velocity or constant radius.

The latter three cases affect the remaining formulas:
-(I) For a turn taken at constant tangential velocity and bank angle:

$$V^*=V: \quad R^*=\frac{R}{1+G}, \quad \omega^*=\omega(1+G), \tag{25a,b}$$

the turn radius and instantaneous turn rate are related to the disturbance intensity by (25a,b),
-(II) For a turn taken at constant radius and bank angle:

$$R^*=R \quad V^*=V\sqrt{1+G}, \quad \omega^*=\omega\sqrt{1+G}, \tag{26a,b}$$

the tangential velocity and instantaneous turn rate are given by (26a,b);
-(III) For a constant turn rate and bank angle

$$\omega^*=\omega \quad R^*=R(1+G), \quad V^*=V(1+G), \tag{27a,b}$$

the radius and tangential velocity are given by (27a,b)
The three cases (I, II, III) are plotted in Figure 19.

If an atmospheric perturbation affects a region of airspace through which the aircraft flies, it can deform its flight path. As a simple example consider an aircraft performing a turn, in the presence of a wind of constant direction. The track angle χ between the flight path and wind changes the disturbance intensity ($G \cos\chi$) between $\pm G$, and the circular turn is transformed into:

-(a) a down wind oval of equation:

$$R(X) = r\,(1 + G \cos\chi), \tag{28}$$

if the turn is performed (27a) at constant rate and bank;
-(b) an upwind oval of equation:

$$R(X) = r\,(1 + G \cos\chi), \tag{29}$$

if the turn is taken (25a) at constant tangential velocity and bank angle.
The ovalized flight path are drawn, in Figure 20, for the two cases, in comparison with the circle $R(\chi) = r$ in still air, for two non-zero values of the disturbance intensity.

We have indicated how the disturbance intensity is defined [1] and calculated in various flight conditions [2], we intend to use flight test data from our aircraft to assess whether:
- the disturbance intensity varies more slowly than other flight parameters, e.g., groundspeed, AoA; vertical acceleration changes are related so as to case a smooth or slow evolution of disturbance intensity;
- the disturbance intensity can be related to subjective assessments of "ride confort", or to the ability to perform certain tasks on board, such as "reading", precise hand movements, etc.

This research complements other work on aircraft flight in perturbed atmospheres, for which some references can be found in the review paper [3].

S 5 - A NON-LINEAR MODEL OF PITCH STABILITY

The preceding account (S4) on flight mechanics in perturbed atmospheres is concerned with comparisons of performance [4] with the case of still air. It is not a substitute for a response calculation [5], which is necessary to calculate flight path changes [6] due to atmospheric disturbances like windshears [7]. The excitation of the phugoid (and short period) mode can be due either to atmospheric disturbances [8] perturbing an aircraft in steady flight, or to [9] initial conditions away fom equilibrium. We will consider a problem in the latter class [10,11,12], i.e. the determination of the pitch control law which compensates the phugoid motion due to initial conditions far away from equilibrium.

S5.1 - EXACT COMPENSATION OF THE PHUGOID MODE

Consider an aircraft in straight and level flight, which starts a dive. The reason may be to track a target of opportunity, or to land in a clearing in an emergency. The dive is started at an initial velocity which is generally distinct, and may be far removed, from the steady flight speeds. If the stick is kept fixed the aircraft will start a phugoid motion [13,14]. In order to compensate completely the phugoid mode, and keep the aircraft on a constant glide slope, the pitch control has to be used appropriately. Our aim is to determine the AoA schedule (or groundspeed schedule) as a function of distance (or time) so as to cancel exactly the phugoid more. In practice it would be necessary to cancel the short-period mode [15] too, in order to keep on a constant glide slope. In the present work we omit the short-period mode, by neglecting the rotational inertia of the aircraft. Thus we are solving "an inverse phugoid problem", of exact cancellation of the phugoid mode.

The problem of compensation of the phugoid mode is expressed by three equations stating that:
-(i) lift balances the component of weight transverse to the flight path of inclination γ (Figure 21):

$$W \cos\gamma = L = \frac{1}{2}\,\rho\, U^2\, C_L(\theta), \tag{30}$$

where the incidence θ is defined $\theta = \alpha - \alpha_0$. As the difference between the AoA α and the angle of zero lift α_0;
-(ii) since the rotational inertia of the aircraft is neglected, it is possible to achieve instantaneously any combination of speed U (in still air) and incidence θ (defined above), consistent with constant lift:

$$U^2\, C_L(\theta) = \text{const} = \cos\gamma\, U_0^2\, C_L(\theta_0), \tag{31}$$

where the constant in (31) is evaluated for the same aircraft in level flight;
-(iii) the balance of forces along the flight path, including inertia, weight, drag and thrust leads to:

$$(W/g)\,\frac{dV}{dt} = -W \sin\gamma + T - D, \tag{32}$$

it being assumed, both in (30) and (32), that the thrust axis coincides with the drag axis. Eliminating between (30-32) a single differential equation for the speed is obtained.

In order to obtain it explicitly we need an aircraft model, i.e. lift, thrust and drag laws. The drag is given by an expression similar to lift (30):

$$D = \frac{1}{2}\,\rho\, S\, U^2\, C_D(\theta), \tag{33}$$

where the drag coefficient in sub-sonic flight is the sum of three components:

$$C_D(\theta) = C_{Df} + k\,\{C_L(\theta)\}^2 + \lambda\, C_L(\theta), \tag{34}$$

namely :
-(i) the form drag coefficient C_{Df}, due to skin friction,
-(ii) the induced drag, proportional to the square of lift coefficient, through the constant k,
-(iii) the term linear on the lift coefficient accounts for a non-symmetric lift-drag polar. The expressions (34) implies that the drag-to-lift ratio is given by :

$$\frac{D}{L} = \frac{C_D(\theta)}{C_L(\theta)} = \frac{C_{Df}}{C_L(\theta)} + \lambda + k\,C_L(\theta), \tag{35}$$

which may be expressed in terms of speed using (31) :

$$\cos\gamma\,\frac{C_D(\theta)}{C_L(\theta)} = \frac{C_{Df}}{C_L(\theta_0)}\,(U/U_0)^2 + \lambda\cos\gamma + k\,C_L(\theta_0)\,\cos^2\gamma\,(U/U_0)^2. \tag{36}$$

We complete the aircraft model by assuming that the dependence of thrust on speed is similar to (36), viz. :

$$\frac{T(U)}{W} = f_0 - f_1 U^2 - f_2/U^2, \tag{37}$$

where the coefficients f_0, f_1, f_2 are all constant.

§ 5.2 - CALCULATION OF SPEED AND INCIDENCE SCHEDULE

The "inverse phugoid problem" (30-32) for the aircraft model (35-37) can be reduced to the solution of a single non-linear differential equation for speed, which is obtained as follows: dividing (32) by the weight and using (30) we obtain :

$$g^{-1}\frac{dU}{dt} = -\sin\gamma + \frac{T}{W} - \csc\gamma\,\frac{C_D}{C_L}, \tag{38}$$

where we may substitute (36) and (37) :

$$g^{-1}\frac{dU}{dt} = F(U) \equiv a - bU^2 - c/U^2, \tag{39}$$

where the coefficients in F(U) are given by :

$$a \equiv f_0 - \sin\gamma - \lambda\cos\gamma, \tag{40a}$$

$$b \equiv f_1 + \frac{C_{Df}}{C_L(\theta_0)}\,U_o^2, \tag{40b}$$

$$c \equiv f_2 + k\,C_L(\theta_0)\,U_o^2\cos^2\gamma \tag{40c}$$

Before we proceed to the response problem, it is worth while to consider the case of steady flight. Steady flight corresponds to zero acceleration $\frac{dU}{dt}=0$, and is possible only for speeds $U_\pm$ which are roots of (39) :

$$0 = F(U_\pm) = \frac{aU^2 - bU^4 - c}{U^2} = -(c/U^2)\,(U^2-U_+^2)\,(U^2-U_-^2), \tag{41}$$

i.e. there are two steady flight speeds (Figure 22) for which drag is balanced by thrust and weight component along the flight path. The steady flight speeds are the roots of the bi-quadratil expression in curly brackets in (41) :

$$U_\pm^2 = \frac{a \pm \sqrt{a^2-4bc}}{2b}, \tag{42}$$

the steady flight is possible only if :

$$\Delta \equiv a^2 - 4bc \geq 0 \tag{43}$$

The case $\Delta = 0$ corresponds to the minimum drag speed

$$\Delta = 0 \quad U_{md}^2 = \frac{a}{2b} = \frac{1}{2}U_o^2\,\frac{f_0 - \sin\gamma - \lambda\cos\gamma}{f_2 U_o^2 - \frac{C_{Df}}{C_L(\theta_0)}}, \tag{44}$$

the minimum thrust required to sustain steady flight is given by the condition $\Delta = 0$, viz :

$$f_1 = 0 = f_2 \quad f_0 = a > 2\sqrt{bc} = 2\cos\gamma\sqrt{k\ C_{Df}} \tag{45}$$

From (42) and (44) it follows that the square of the minimum drag speed is the arithmetic mean of the squares of the steady flight speeds.

$$U_{md}^2 = \frac{1}{2}(U_+^2 + U_-^2) \,. \tag{46}$$

this implies that the upper steady flight speed cannot exceed the minimum drag speed $U_+ < U_{md}^2\sqrt{2}$ by more than $41\% = \sqrt{2} - 1$.

If the initial dive velocity U_0 happens to be to one of the steady flight speeds ($U_0 = U_+$ or $U_0 = U_-$) the aircraft will remain at that velocity, provided it be stable. To asses stability we consider a speed distinct from the steady values, and check the sign of the acceleration (39,41)

$$\frac{dU}{dt} = -(cg/U^2)\,(U^2 - U_+^2)\,(U^2 - U_-^2), \tag{47}$$

viz it is positive between the steady flight and negative outside:

$$\begin{array}{llll} \text{case} & U < U_- & U_- < U < U_+ & U > U_+ \,, \\ \text{acceleration} & \dfrac{dU}{dt} < 0 & \dfrac{dU}{dt} > 0 & \dfrac{dU}{dt} < 0 \,, \end{array} \tag{48}$$

as illustrated in Figure 23, from which it is clear that:
-(i) the upper steady flight speed is stable, since an aircraft at a higher (lower) speed will deccelerate (accelerate) towards U_+,
-(ii) the lower steady flight speed is unstable, since an aircraft at a lower (higher) speed will deccelerate (accelerate) away from it. If the initial velocity is far removed from the upper steady flight speed we have a non-linear stability problem, if the initial velocity is close to the lower steady flight speed then the instability may grow into a non-linear regime.

S 5.3 - STABILITY CURVES FOR CONVERGENCE AND DIVERGENCE

In order to obtain the stability or instability curves we have to integrate the equation of motion, e.g., for velocity U as a function of distance ξ along the flight path

$$\frac{dU}{dt} = \frac{dU}{d\xi}\frac{d\xi}{dt} = U\,\frac{dU}{d\xi} \tag{49}$$

substituting (49) into (47) we have

$$\frac{d(U^2)}{d\xi} = -\frac{2\,c\,g}{U^2}(\,U^2 - U_+^2)\,(\,U^2 - U_-^2), \tag{50}$$

which may be integrated using the change of variable $\xi \equiv U^2$

$$\begin{aligned} -2cg &= \int_{U_0}^{U^2} \frac{\xi}{(\xi - U_-^2)\,(\xi - U_+^2)}\,d\xi \\ &= \frac{1}{(U_+^2 - U_-^2)}\left[U_+^2 \log(\xi - U_+) - U_-^2 \log(\xi - U_-)^2\right]_{U_0}^{U} \end{aligned} \tag{51}$$

We introduce the lengthscale:

$$\Lambda \equiv \frac{1}{2bg} = \frac{U_0^2\ /2g}{f2\ U_0^2 - \dfrac{C_{Df}}{C_L(\theta_0)}} \,. \tag{52}$$

and the distance along the flight path ξ, when divided by the lengthscale, is given as a function of velocity by:

$$e^{-\xi/\Lambda} = \left[\frac{U(\xi)^2 - U_+^2}{U_0^2 - U_+^2}\right]^{U_+^2/(U_+^2+U_-^2)} \left[\frac{U_0^2 - U_-^2}{U(\xi)^2 - U_+^2}\right]^{U_-^2/(U_+^2+U_-^2)} \,. \tag{53}$$

where $U(\xi)$ is the speed at distance ξ, $U_0 \equiv U(0)$ is the initial speed, and the steady flight speeds $U_\pm$ appear as parameters. The expression (53) is highly non-linear, and involves dimensionless ratios of speeds, it is not possible to write speed $U(\xi)$ as a function of distance ξ, but the inverse form $\xi(U)$ in (53) can be used to plot the curves in Figure 24.

The expression (53) can be linearized if the speed does not differ much from the initial values

$$\left[U(\xi)^2 - U_0^2\right]^2 \ll \left[U_\pm^2 - U_0^2\right]^2 . \tag{54}$$

in which case (53) simplifies to

$$e^{-\xi/\Lambda} = 1 + \frac{U_0^2 - U(\xi)^2}{U_+^2 - U_-^2}\left[\frac{U_+^2}{U_+^2-U_0^2} + \frac{U_-^2}{U_0^2-U_-^2}\right], \tag{55}$$

which can be inverted to give speed as a function of distance :

$$U(\xi)^2 = U_0^2\left[1 + \left(\frac{U_+^2}{U_0^2 - 1}\right)\left(\frac{1 - U_-^2}{U_0^2}\right)\left(1 - e^{-\xi/\Lambda}\right)\right]. \tag{56}$$

In the linear case the stability curves are exponentials with the lengthscale l given by (52); this result could be obtained by the standard methods [12] of laplace transform and linear stability derivatives. It applies only for small disturbances. For large disturbances the curves (53) are not exponentials, as can be seen in Figure 24.

In Figure 24 we plot the speed normalized to the minimum drag speed (57a) :

$$V \equiv \frac{U(\xi)}{U_{md}}, \qquad X \equiv \frac{\xi}{\Lambda}, \tag{57a,b}$$

versus distance normalized to the lenghtscale (57b) we consider an aircraft whose stable steady flight speed lies 30% above the minimum drag speed (58a) :

$$V_+ \equiv \frac{U_+}{U_{md}} = 1.3 \qquad V_- \equiv \sqrt{2 - V_+^2} = 0.56, \tag{58a,b}$$

implying by (46) that the unstable steady speed lies 44% below (58b). If the initial speed equals one of the steady flight speeds the aircraft flies steadily at that speed on a constant glide slope (horizontal lines $V = V_\pm$). If the initial velocity lies above the upper steady flight speed $V_0 > V_+$, in order to keep a constant glide slope, the aircraft must deccelerate towards V_+ along the curves shown. Between the steady flight speeds $V_- < V < V_+$, the aircraft accelerates towards the upper steady speed V_+ which is stable : there is an inflexion in the stability curve at the minimum drag speed $V = 1$, if the initial velocity is smaller $V_0 < 1$. Below the lower steady speed there is a strong instability, i.e., rapid speed loss towards the stall.

The theoretical stability (and instability) curves can be plotted for speed or incidence as a function of distance or time, and compared with data recorded in flight at a given initial velocity along a given constant glide slope, to :
- check whether averaging over a time scale larger than the short period mode, leads to experimental curves consistent with the theoretical prediction;
- subtracting the phugoid compensation curve from flight data, the short period mode is isolated.
If the latter is of small amplitude, it could be included as a linear perturbation of the present, non-linear theory.

S 6 - CONCLUSIONS

The program to give Portugal a flight test capability should reach within a few months its major milestone : the first flight of the instrumented Aviocar aircraft. It is possible to draw, on the basis of experience to date, the following four conclusions :
- the AGARD support program to the NATO southern flank nations can be effectively used to improve their technological capability in the aerospace field; its success depends on a strong commitment by the supporting nations, as was the case in the present program with the Netherlands through the NLR and Germany throught the TU-BS;
- at the national level there must be close cooperation between developers and the users, in the present case the FAP and UTL; once this cooperation is established in one program, e.g. flight testing, it tends to spread to other fields (i.e electronics, structures);
- the program should provide an independent and self-supporting capability in the supported nation, which it can develop further to suit its own needs, the basic flight test capability gained in Portugal with the Aviocar aircraft may be extended to the qualification of special equipment in high-performance aircraft;
- the pratical work on aircraft should be complemented by scientific modeling, to give a balenced capability to up-date knowledge and experience, in the present case, the instrumented aircraft will be used in connection with fundamental and applied research projects for which it is well suited.

The present program has focused scientific and technological capabilities in Portugal to research and development in the aeronautical field. We hope that the international cooperation which made this program possible may continue to support its further development and applications, at a national, multilateral and NATO level.

ACKNOWLEDGMENT

The authors acknowledge the contribution made to the programme by the following individuals named in the text :

(name)	(pages)
Cap. Almeida	28-9, Fig.4
Cap. R.R. Almeida	28-9, Fig.4
Mr. A. Arantes	28-7, 28-9, Fig.4
Prof. Arantes de Oliveira	28-5, Fig.3
Prof. J.A.C. Azinheira	28-9, Fig.4
Brig. Bermudez	Fig.3
Ir. J. van der Bliek	28-5
General Bourbon	28-5
Dipl-Ing. V. Brandt	28-7
General Brochado Miranda	28-5, Fig.3
Mr. D. Brunner	28-6
Prof. L.M.B.C. Campos	28-1, Fig.3
Lt-Col. A.M.S. Cardoso	28-9, Fig.4
Cap. Carvalheira	28-9, Fig.4
Cap. Carvalho	28-6
Cap. Damásio	28-9, Fig.4
Eng. M. Domingues	28-9, Fig.4
Ir. J.T.M. van Doorn	28-5
Prof. A.F.O. Falcão	Fig.3
Eng A.R.A. Fonseca	28-1, 28-7, 28-9, Fig.4
Prof. T. Freitas	Fig.3
General Galvão Borges	28-5
Prof. O.H. Gealach	28-5
Dr. P. Hamel	28-5
Ir. P.M.N. Hollestelle	28-7
General Lemos Ferreira	28-5, Fig.3
Prof. G. Mandelung	28-5
Ir. H. Moelke	28-5
Dr. H.A. Mooij	28-5
Col A.A. Portela	28-5, Fig.3
Eng. H.F.S. Ramos	28-1, 28-7, 28-9, Fig.4
Prof. J.J.E. Santana	28-6
Prof. J.M.G. Sá da Costa	28-8
Prof. G. Schanzer	28-5
Ing. P. Sevenhuysen	28-7
Prof. Simões Lopes	28-5, Fig.3
Dr. P. Vorsmann	28-6
Mr. R. van der Welde	28-7
Mr. E. von Wissekerke	28-6

REFERENCES

[1] L.M.B.C. CAMPOS On the influence of atmospheric disturbance on aircraft aerodynamics. Aeron. J., June/July 19· 256–264 (Paper 1085).

[2] L.M.B.C. CAMPOS On aircraft flight performance in a perturbed atmosphere. Aeron. J. Oct. 1986, 302–312 (Paper 1305)

[3] B. ETKIN The trubulent wind and its effect on flight. AIAA Wright Brothers Lecture, Univ.Tor. Aerosp. Dep. Rep. 44, 1980

[4] L.M.B.C. CAMPOS On the disturbance intensity as an indicator of aircraft performance degradation in a perturbed atmosphere. 2nd Int. Conf. Aviation Safety, Toulouse 1986.

[5] L.M.B.C. CAMPOS On the compensation of the phugoid mode induced by initial conditions and windshears. 16th Int. Congress of Aeronautical Sciences, Jerusalem 1988.

[6] A.A. WOODFIELD & J.F. WOODS Worlwide experience with windshear during 1981–1982. AGARD CP-347, paper 11

[7] G. SCHANZER Influence of windshear on flight safety. AGARD CP–347, paper 12

[8] L.M.B.C. CAMPOS On a pitch control law for constant glide slope through windshears. Aeron. J. (to appear)

[9] L.M.B.C. CAMPOS On the inverse phugoid problem as an instance of non–linear stability in pitch. Aeron. J.

[10] PAINLEVÉ, P. 1910 Étude sur le régime normal d'un aeroplane. Technique Aéronautique 1, 3-11

[11] PINSKER, W.J.G. 1971 "Glide-path stability of an aircraft under speed constraint" A.R.C. Report 3705.

[12] NEUMARK, S. 1953 "Problems of longitudinal stability below the minimum drag speed, and the theory of stability under constraint" A.R.C. Report 2983

[13] F.W. LANCHESTER Aerodonetics, Constable 1908 London, 2nd ed. 1910, new 1945

[14] VON MISES, R. 1945 Theory of flight. McGraw-Hill, reprinted Dover 1973

[15] B. ETKIN, Dynamics of flight stability and control. Wiley 1974.

[16] BABISTER, A.W. 1980 Aircraft dynamic stability and response. Pergamon Press

Pos. nr	Qty	Description	Purchase year
1	2	PCM data acquisition system	75
2	2	PCM Quick Look system	79
3	1	PCM test rack	78
4	1	Punched tape reader	79
5	2	Kinelogic recorder	71
6	2	PCM signal conditioning system	77-79
7	140	Electronic interface card	72-78
8	7	Traducteur + power supply	66
9	14	Accellerometer Donner	64-76
10	6	Accellerometer SFIM	54-60
11	7	Airspeed transducer Kolsmann	64-70
12	2	Air data computer IDC	66-68
13	1	Altitude/airspeed transducer (minidrum)	77-80
14	2	Altitude transducer Kolsmann	64
15	1	Radaraltimeter Bendix	68
16	2	Slip/attack angle transducer	67
17	41	Position transducer	65-67
18	2	Pressure transducer Gulton	74
19	65	Pressure transducer ACB	57-71
20	28	Pressure transducer Penny & Giles	64-67
21	8	Pressure transducer AMA	60
22	14	Pressure transducer SFIM	64-72
23	3	Arinc clock	80
24	4	Contact clock	70
25	1	Transducer for steering force	53
26	1	Temperature transducer	69
28	7	Rate gyro SFIM	64-66

TABLE I - specification of equipment shipped to Portugal

External Dimensions	wing span	19.00 m	(62 ft 4 in)
	lenght overall	15.20 m	(49 ft 10 1/2 in)
	height overall	20 ft 8 in	(6.30 m)
Internal Dimensions	cabin (between flight deck and rear-loading door)		
	lenght	5.00 m	(16 ft 4 3/4 in)
	width	2.00 m	(6 ft 6 1/2 in)
	height	1.70 m	(5 ft 7 in)
	volume	17,5 m3	(618 cu ft)
Weights	max payload	2000 kg	(4,410 lb)
	max T.O. weight	6300 kg	(13,889 lb)
	weight empty	3905 kg	(8,609 lb)
Performance	max never-exceed speed (EAS)	240 knots	(276 mph or 445 km/h)
	max cusing speed at 12,000 ft (3,660m)	194 knots	(223 mph or 359 km/h)
	stalling speed, flaps down	62 knots	(112 mph or 116 km/h)
	service ceiling	26.700 ft	(8,140 m)
	TO run	350 m	(1148 ft)
	landing run	207 m	(679 ft)
	range at 12,000 ft (3,660 m) with max fuel and 2,303 lb (1045 kg) payload	949 mm	(298 miles or 480 km)

TABLE II - CASA C212 Aviocar characteristics

TABLE III - bar chart of major tasks

	control	monitoring
pilot	-main switch	-rec. on -test number (RC)
operator	-rec. on/off sw. -run counter (RC)	-power supply -rec. on -begin-of-tape (BOT) -end-of-tape (EOT) -monitoring of the recorded value of any of 2 channels

TABLE IV - Control and monitoring capabilities of each crew member

Category	Ref	Parameter	Code	Range
General	01	test signal	TS	
	02	time base 1	TB1	
	03	time base 2	TB2	
	04	run counter	RC	1/99
Air Data	05	differential pressure	PD	0/10 kPa
	06	static pressure	PS	30/105 kPa
	07	calibration pressure	PC	0/15 mb
	08	total air temperature	TAT	-50/+50 °C
Configuration	09	ground/flight switch	GFS	on/off
	10	wing flap position	DF	0/45 deg
Control	11	elevator deflection	DE	-30/+20 deg
	12	LT aileron deflection	DA1	-20/+20 deg
	13	RH aileron deflection	DA2	-20/+20 deg
	14	rudder deflection	DR	-25/+25 deg
	15	elevator force	FE	-450/+450 N
	16	aileron force	FA	-300/+300 N
	17	rudder strain A	FR1	
	18	rudder strain B	FR2	
	19	rate of pitch	RP	-20/+20 deg/s
	20	rate of roll	RR	-60/+60 deg/s
	21	rate of yaw	RY	-20/+20 deg/s
	22	acceleration (X-dir)	AX	-1/+1 g
	23	acceleration (Y-dir)	AY	-1/+1 g
	24	acceleration (Z-dir)	AZ	-2.5/+2.5 g
	25	angle of attack	AA	-35/+35 deg
	26	angle of side-slip	AS	-35/+35 deg
	27	angle of pitch	AP	-90/+90 deg
	28	angle of roll	AR	-90/+90 deg
	29	INS valid signal	INSf	on/off
Propulsion	30	engine speed L	N1	0/41730 rpm
	31	engine speed R	N2	0/41730 rpm
	32	fuel flow L	FF1	0/1050 lb/h
	33	fuel flow R	FF2	0/1050 lb/h
	34	turbine gas temperature L	TGT1	0/930 °C
	35	turbine gas temperature R	TGT2	0/930 °C
	36	torque pressure L	TP1	0/65 psi
	37	torque pressure R	TP2	0/65 psi
Auto-Flight	38	autopilot engaged	AE	on/off
	39	flight director mode	FDM	1/5
Navigation	40	true heading	HDG	0/360 deg
	41	HDG valid signal	HDGf	on/off
	42	radio altitude	RA	0/2500 ft
	43	RA valid signal	RAf	on/off
	44	localizer deviation	LLD	-90/+90 deg
	45	LLD valid signal	LLDf	on/off
	46	glide slope deviation	GSD	-80/+80 deg
	47	GSD valid signal	GSDf	on/off
	48	drift angle	DFT	-18C/+180 deg
	49	DFT valid signal	DFTf	on/off
Thrust	50	A mount strain (L engine)	FA1	
	51	B mount strain (L engine)	FB1	
	52	C mount strain (L engine)	FC1	
	53	D mount strain (L engine)	FD1	
	54	E mount strain (L engine)	FE1	
	55	F mount strain (L engine)	FF1	
	56	G mount strain (L engine)	FG1	
	57	H mount strain (L engine)	FH1	
	58	A mount strain (R engine)	FA2	
	59	B mount strain (R engine)	FB2	
	60	C mount strain (R engine)	FC2	
	61	D mount strain (R engine)	FD2	
	62	E mount strain (R engine)	FE2	
	63	F mount strain (R engine)	FF2	
	64	G mount strain (R engine)	FG2	
	65	H mount strain (R engine)	FH2	

TABLE V - Parameter list

nº of channels	input type
42	dc voltage
3	dc current
11	synchro signal
3	frequency
3	variable resistors
20	discrete signals
3	digital inputs (15 bit)

TABLE VI - actual Data Acquisition System capabilities

analog inputs (10 bit resolution)	digital inputs (20 bit resolution)
72	0
70	1
68	2
66	3
64	4
62	5
60	6
58	7
56	8

TABLE VII - maximum NDAS input channels using the parallel output

nº of tracks	16
input data type	digital
magnetic tape	1.1 MIL tapes 1" wide and 4600ft long
tape speed	15 ips
max recording time	3600 sec (1h)

TABLE VIII - Kinelogic Model YE tape recorder characteristics

Unit	breakers gauge		power requirements	
	28Vdc	115Vac		
PCM DCU	2A		28Vdc 28Vdc	600mA 230mA
rec	5A 7.5A	2.5A 0.5A	electronics (28Vdc 45W) motor write-amplifiers (55Vdc) heating	
SCU1		1A	5Vdc ±15Vdc	2A 1A
SCU2		1A	5Vdc ±15Vdc	3A 1.2A
SCU3		1A	5Vdc ±15Vdc	4A 0.3A
SDCU		3A	115Vac	
transducers		1A	±15Vdc +7.5Vdc 26Vac400Hz	
external devices	5A 5A	6A 6A	28Vdc 115Vac	300W 1kVA

TABLE IX - Power requirements of each unit and breaker's gauges

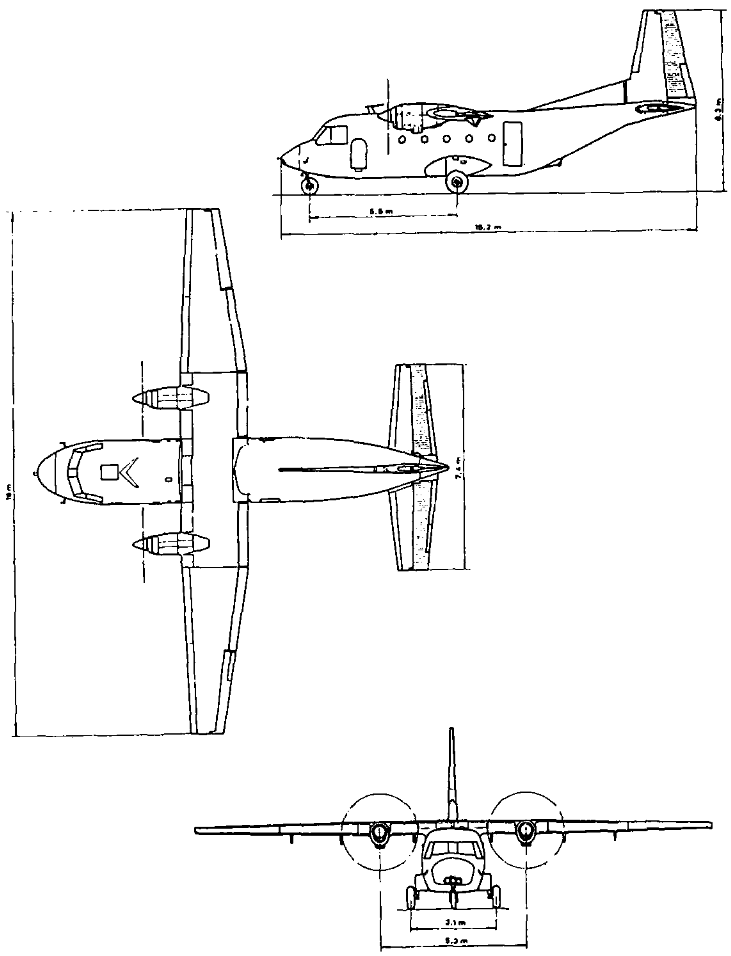


FIGURE 1 - three view drawings of the CASA C212 Aviocar

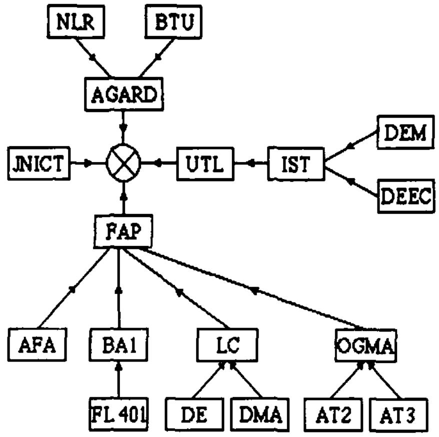


FIGURE 2 - Institutions participating in the programme

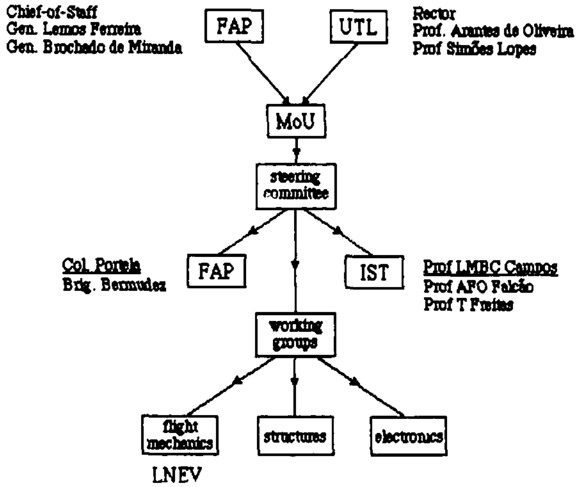


FIGURE 3 - cooperation between Air Force and University

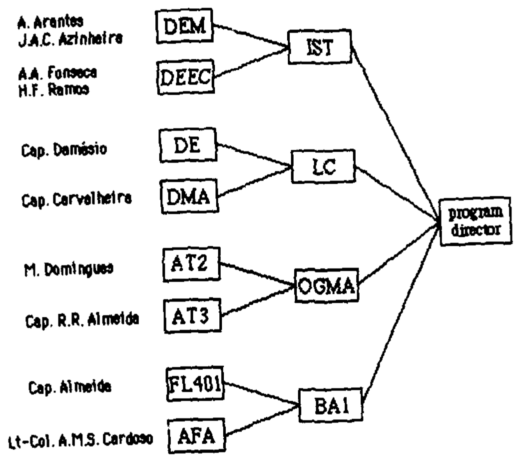


FIGURE 4 - working group on LNEV

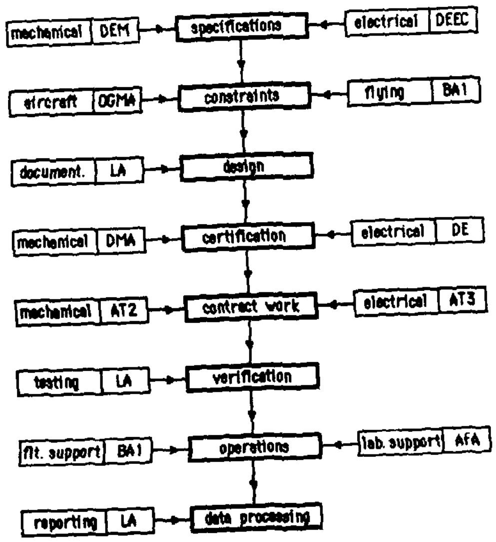


FIGURE 5 - implementation of the FTI system

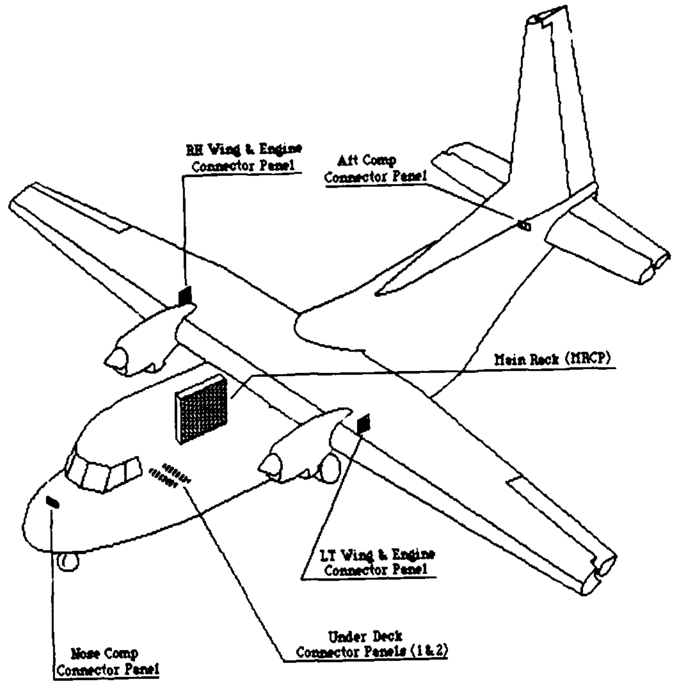
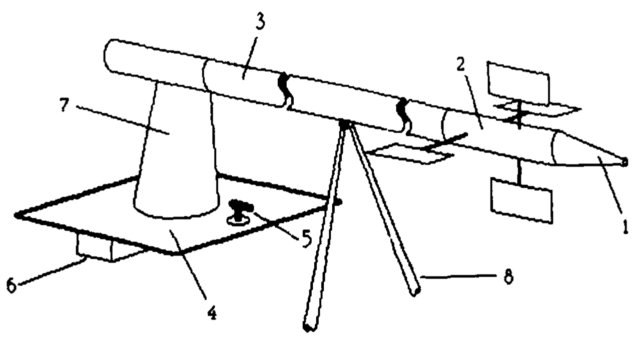


FIGURE 6 - connector panels location



notes:
1 - pitot tube and static port
2 - two section cilindrical vane
3 - cilindrical body
4 - upper emergency exit
5 - temperature sensor
6 - connector panel
7 - main support
8 - supports

FIGURE 7 - nose boom

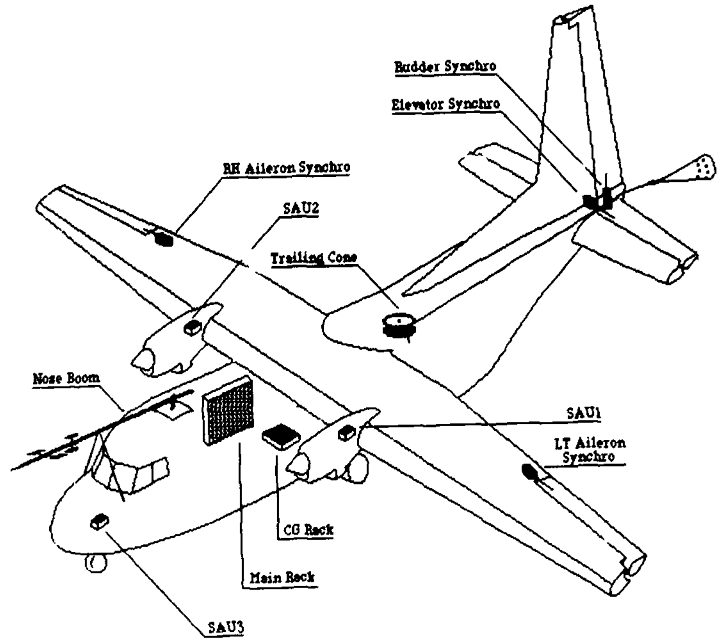


FIGURE 8 - sensor location

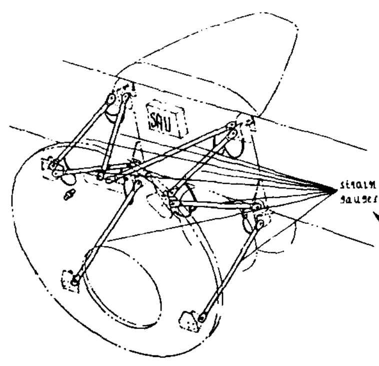


FIGURE 9 - detail of strain gauges in the 8-bar engine mountings

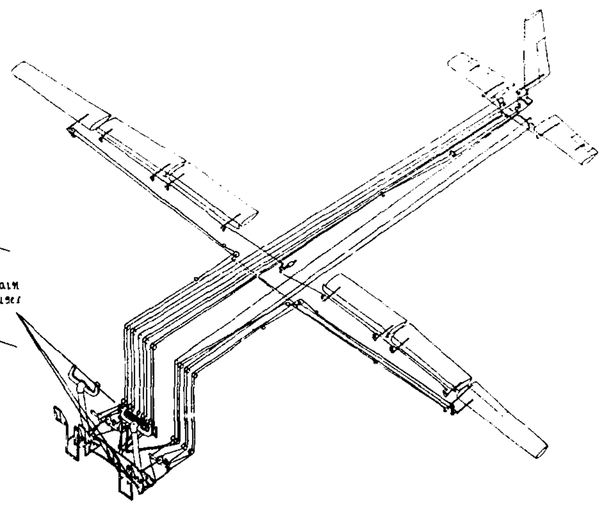


FIGURE 10 - detail of strain gauges for stick forces measurements

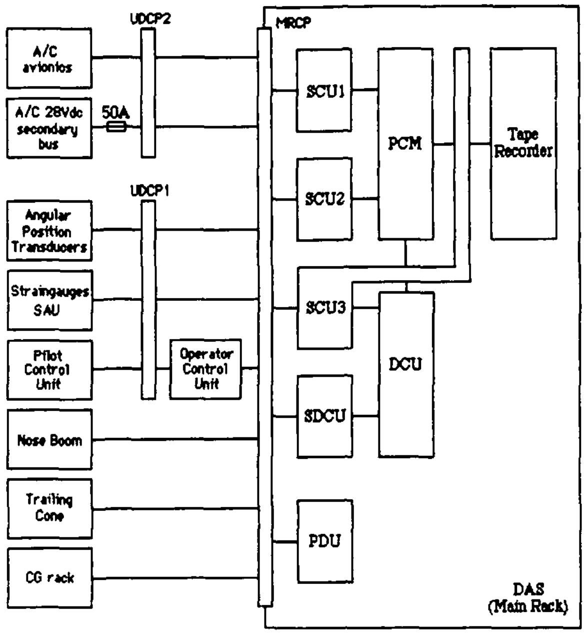


FIGURE 11 - DAS internal block diagram

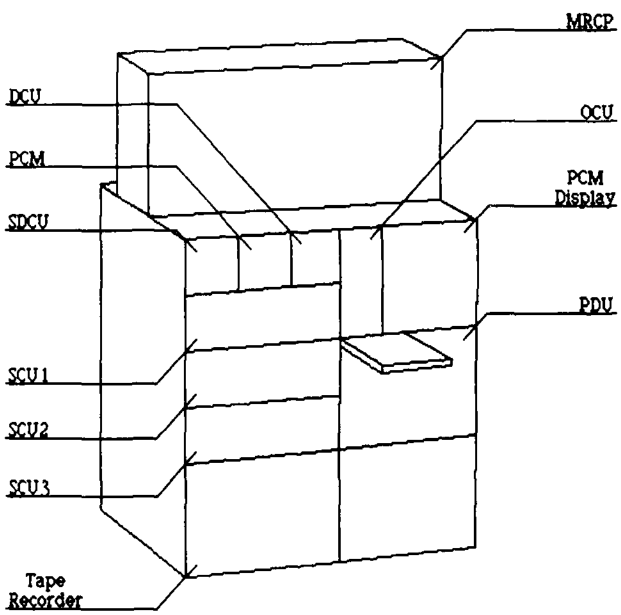


FIGURE 12 - Main Rack layout

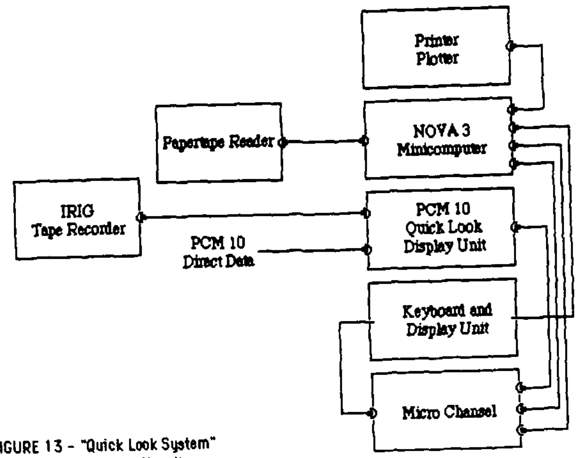
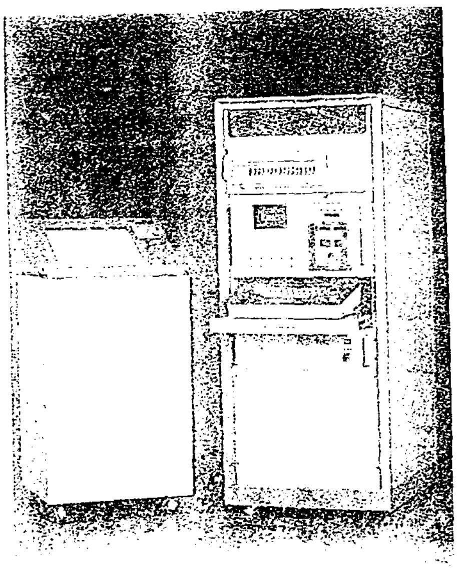


FIGURE 13 - "Quick Look System" interconnection diagram



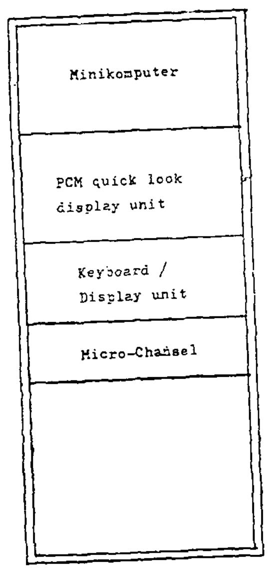


FIGURE 14 - the "Quick Look System"

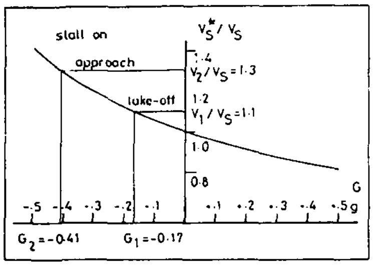


FIGURE 15 - Ratio of stalling speeds in the presence V^*_S and absense V_S of atmospheric disturbances, versus distuebance intensity G, including the critical intensities causing stall at unstick for take-off (V_1G_1) and on approach to land (V_2G_2)

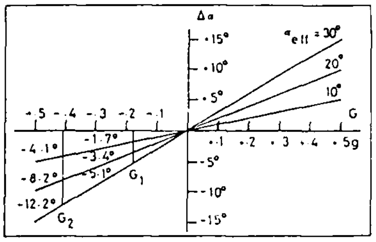


FIGURE 16 - Change in angle-of-attack $\Delta\alpha$ versus disturbance intensity G, for fixed initial values of the effective incidence α_{eff}

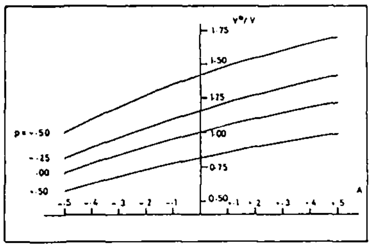


FIGURE 17 - Staight, horizontal flight in atmospheric disturbances: ratio of velocities (V^* in the presence and V in the absence) versus vertical acceleration A, for fixed values of the disturbance incidence parameter p.

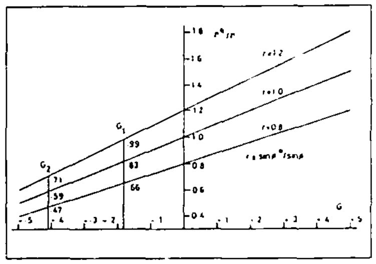


FIGURE 18 - Ratio of centripetal accelerations (in the presence l^* and in the absence l of disturbances), versus disturbance intensity G, for fixed values of the bank angle parameter r (defined from the angles of bank ϕ^*, ϕ).

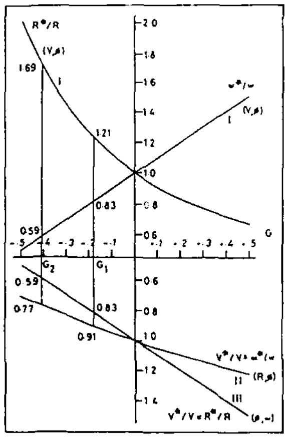


FIGURE 19 - Effect of the disturbance intensity G on a horizontal turn, in the cases where two parameters are kept constant and two parameters are allowed to vary: (I) ratio of tangential velocities V^*_S/V_S and instantaneous turn rates ω^*/ω at constant radius of turn R and bank angle ϕ; (II) ratio of tangential velocities V^*_S/V_S and radii of turn R^*/R at constant turn rate ω and bank angle ϕ; (III) ratio of radii of turn R^*/R and instantaneous turn rates ω^*/ω at constant tangential velocities V and bank angle ϕ.

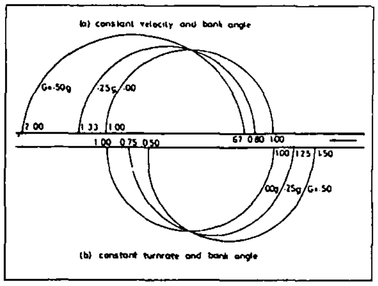


FIGURE 20 - Deformation of the circular trajectory, corresponding to horizontal turn in still air, due to atmospheric disturbances of intensity G=0.25, 0.50, in the cases of: (a) a downwind oval at constant tangential velocity and bank angle; (b) an upwind at constant turn rate and bank angle.

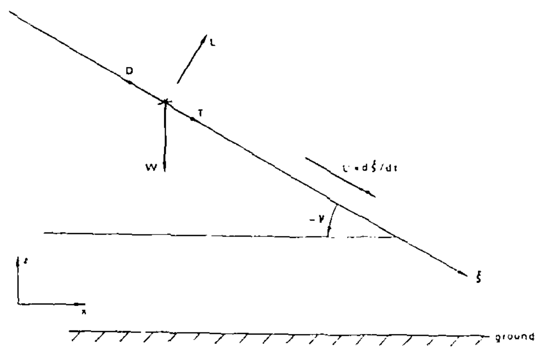


FIGURE 21

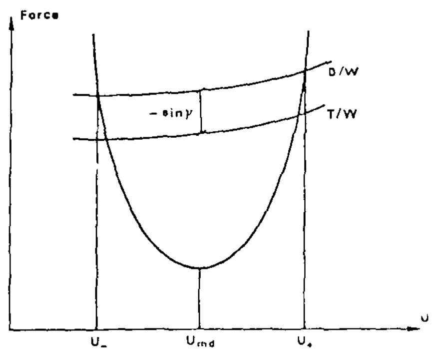


FIGURE 22

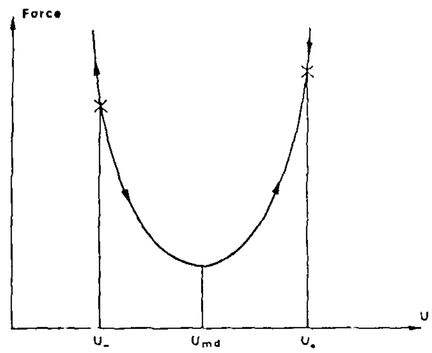


FIGURE 23

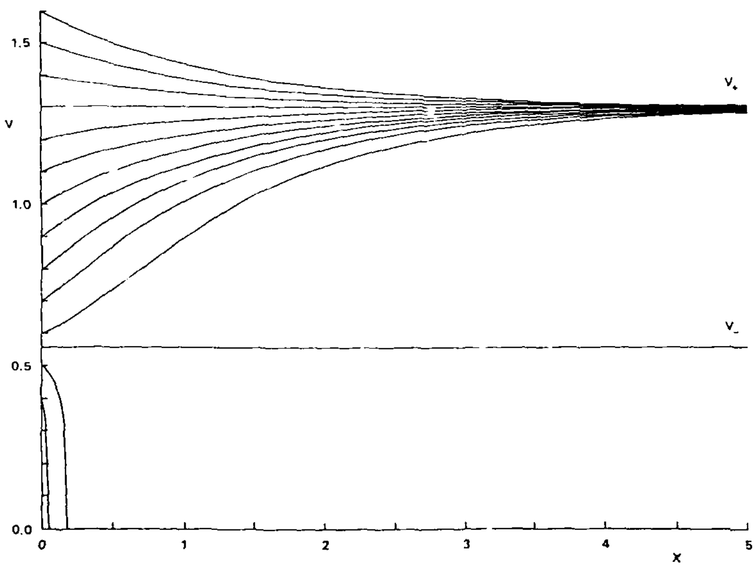


FIGURE 24

REALTIME PROCESSING AND DISPLAY

by
A. Redeker
Aerodata Flugmeßtechnik GmbH
Rebenring 33
D-3300 Braunschweig
Federal Republic of Germany

SUMMARY

Concept and realization of a data system for airborne research and development applications are presented. Aspects of hardware architecture and software organisation will be discussed. The system is used for prototype testing and certification and is also layed out for the operation of research aircraft, where sensor configurations and measuring instruments are varying from one experiment to another.

All system users have access to a pool of common sensor- and software ressources. Special hard- and software interfaces have been defined for a simple integration of individual sensor signals and computation algorithms.

The system performs realtime processing, recording, and on-line monitoring of sensor data. It is the aim of the system design to enable the operator to perform quick error detection as well as to optimize the flight conditions for an experiment.

Examples for flight-mechanical and meteorologigal system applications are given.

1. DEMANDS ON AIRBORNE DATA SYSTEMS FOR RESEARCH AND DEVELOPMENT

Airborne data acquisition systems for research aircraft require special attributes for the execution of effective flight tests. The operation of complex sensor systems under difficult measuring conditions on board an aircraft is rather expensive. It becomes more expensive, if flight tests are unsuccessful because the personnel on board the aircraft is not informed about the quality of the measurement. Thus, the operators should have a good insight into all sensor data, measuring parameters, and flight conditions. This gives the chance to influence the further operation and to optimize the conditions of the measurement, but on the other hand it imposes high demands on the capabilities of the on-board data system.

A scientific user or an engineer, who wants to take the aircraft as a sensor carrier for his special purpose, is not necessarily a computer specialist. The data acquisition system is a tool, which has to be transparent for him. From the user's point of view it is important to be able

- to have access to the standard aircraft states, e.g. position, flight level, speeds, attitude angles, angular rates, accelerations, flow angles, etc.
- to integrate individual sensor equipment into the data system
- to add private on-line computation algorithms
- to record user data together with standard data on the same data carrier
- to have a look into sensor data and on-line computation results
- to initiate and to stop private sensor alignment and calibration procedures

For the realization of an equipment which fulfills these requests, one needs a powerful computer system, which is flexible and modular in it's hardware and software components. The system architecture has to be outlined in a way, that its components can develop their performance. Interfaces for additional hardware and software have to be provided.

3. HARDWARE CONFIGURATION

An important attribute for an on-board data system is its realtime capability. It is a well known experience that it is not suitable to handle data acquisition and data processing with the same processor. Even powerful CPUs can't develop their throughput, if they are disturbed by high interrupt rates, which are unavoidable for a continuous data sampling from different signal sources.

Therefore data acquisition and data processing should be separated and executed from different system parts (FIG.1). The main processor is freed from simple but time-intensive I/O-operations and reserved for complex numerical and monitoring operations.

Both system parts are working in parallel. Once per sampling period they are exchanging I/O-data arrays.

The data acquisition system is responsible for the signal conditioning, sampling, conversion and synchronisation of different signals. Data from multichannel serial datalinks like ARINC-429-busses have to be assorted.

After this data collection process, the data preprocessor puts the data into an array of fixed length and structure, which is transfered once per sampling interval into the main computer (e.g. every 20 milliseconds, = 50 Hz). Depending on the physical distance between data acquisition system and main computer, the data transfer is either performed as a serial single-line PCM-signal (PCM=Puls Code Modulation) or by a parallel data link.

FIG. 2 shows the hardware of a data preprocessor, which is used in a so called METEOPOD. The meteopod is a container mounted under the wing of an aircraft. It contains all sensors for the determination of wind, turbulence, humidity and other aerological parameters.

Realtime computations on the main computer have to be carried out within one single samplimg period. This includes the decoding and converting of received data into engineering units as well as the computation of realtime algorithms, the production of other secondary data, and data storage on a computer compatible streamer tape.

The main computer, a ruggedized airworthy Q-Bus system, (FIG.3) can be alternatively equipped with a LSI-11/73 CPU (1 megabyte of main memory, 64 kilobytes of fast cache-memory), or with a DEC microVax II (8 or 16 megabytes of main memory).

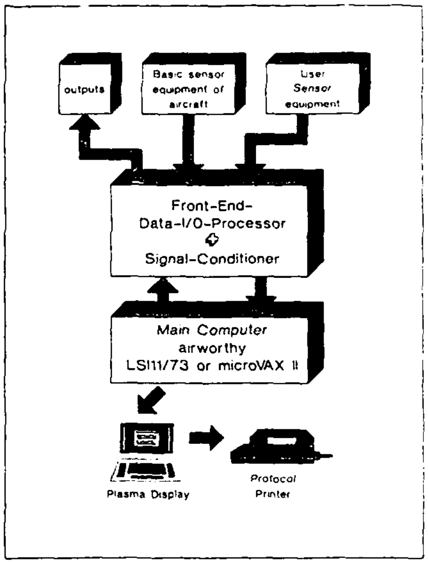


FIG.1 : Structure of the data *acquisition system*

Both versions are working with two 3 1/2" winchester drives (20 megabytes each), two streamer tapes (60 megabytes), a realtime clock, and different serial and direct memory access-interfaces (DMA). Power is supplied directly from the aircraft's 28 Volt DC Bus or from 115 Volts AC, 400Hz.

FIG. 2 : Hardware of a Data Preprocessor

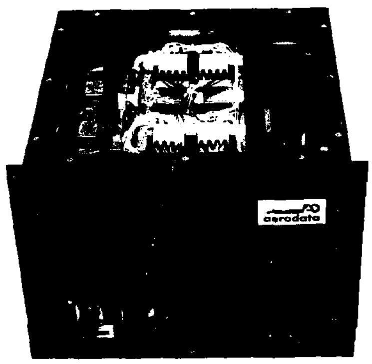


FIG. 3: Hardware of a Main Computer

FIG.4 shows the architecture of the main computer. *The configuration can be adapted* according to the expected realtime load, which is approximately proportional to the product of sampling rate and signal channel number. If this product is small, no preprocessor is required and the signal converters are installed directly in the main computer.

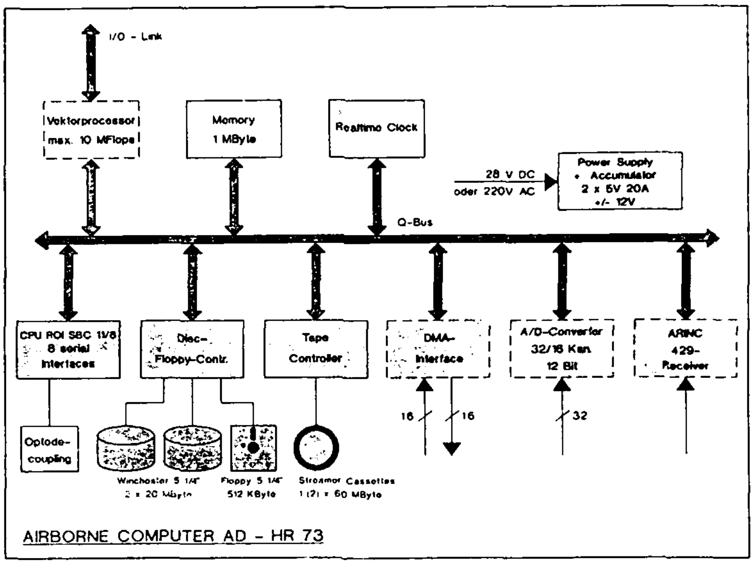


FIG. 4: Configurations of the Main Computer

For most applications, the main computer does not contain signal converters and the preprocessor *is* interconnected *by a Direct Memory Access Interface (DMA).*

If more power for numerical operations is needed, a vectorprocessor will be installed in the main computer. It relieves the Central processor from large parts of the realtime task. The vector processor itself is equipped with an external I/O-port and replaces the DMA for the preprocessor. It is now directly communicating with the preprocessor, reads raw data, decodes and converts them into engineering units, and feeds them by it's own DMA logic into the main computer memory.

Console operation of the computer is done with high-resolution plasma-terminals, because they have no cathode-ray-tube, which can implode, and there is no high voltage, which may lead to problems at low air density (e.g. at high altitudes in an unpressurized aircraft).

A few words about hardware specifications: A rather safe way concerning reliability is the use of mil-specified components. They are well suited for all applications, where a fixed system configuration has to be operated with high reliability for a long time. However, for the present discussed application they have several disadvantages: because of the time and cost intensive certification procedures they are very expensive and generally one or two generations behind the state of the art. Another argument against the use of mil-spec-components for this purpose is that most measurements for R&D applications are not really safety critical.

On the other hand, problems of commercial computers with environmental conditions on board an aircraft are well known, especially those with power supplies, mechanical loads, and electromagnetic interference. Thus it may be expensive to use cheap commercial equipment.

The computer hardware of the presented system is not mil-specified. However, the integrated system hardware differs significantly from commercial computers. Several measures have been undertaken to make the system airworthy. To ensure the electromagnetic compatibility, power supplies are equipped with effective RF-Filters. Electrical connections between main- and preprocessors or serial I/O-lines are isolated by optocouplers. The mechanical construction is ruggedized and the main computer is protected against power fails by a battery backup. Thus, the system has an excellent electrical and mechanical stability.

3. REQUIREMENTS TO THE SOFTWARE

Besides data storage and processing of on-line calculations the on-board system has to perform another task, which is of great importance for effective measurements: monitoring. A complex measuring system can be operated with far more success, if there is a way to verify

- that sensors and algorithms work correctly
- that the operating conditions of the whole system are as required

The software for the presented data system is named MODAMS (Modular Data Acquisition and Monitoring System). For a comfortable monitoring the operator needs a basic set of instructions to tell the system, what he wants to observe, what kind of calibrations are to be done, what parts of the realtime process are activated or switched off, etc.

Software is partitioned into separate processes. For example, there are realtime, dialogue, and monitoring processes.

Each process itself is divided into a standard part, which is common to all users, and a user module, where private functions can be implemented. The user has access only to those data structures, which are defined as public in the respective user interface of the process.

Standard data processing and monitoring is performed by the use of parameter tables. The contents of the tables adapts the system to its special application. Parameter tables describe the characteristics of sensor data channels as well as the contents of monitoring menus and the instructions for the operator dialogue.

4. PROGRAMMING LANGUAGE

It was already stated that the software system has to be modular and requires the installation of parallel processes.

As it is a very large and complex system, the programming language should give some support to avoid the typical problems of large software systems. On the other hand the programming language should be acceptable for a scientist who is not a computer specialist.

All concurrent software processes including the process-scheduler are part of the MODAMS-Software. The interfaces to operating system dependent functions are reduced to a minimum. Therefore MODAMs can be transfered to different computers and operating systems, which normally don't support concurrent processes. MODAMS is written in the modern high level language MODULA 2, which combines the advantages of PASCAL and C. The important difference to PASCAL, C and FORTRAN is a strict modular concept, which for example allows argument checks already at compilation time even with procedures from external libraries.

The operating system for MODAMS on the LSI-11 is the RT11 single-Job monitor, just now MODAMS is transferred to the microVax under VMS.

5. ORGANISATION OF SOFTWARE PROCESSES

The software system is divided into standard modules and user modules. User modules have to be written according to the system conventions, and are linked to the standard system.

FIG.5 shows the data flow between the different processes. The realtime process and the data recording process are executed with the highest priority. Sampling rates are normally between 10 Hz and 100 Hz. The remaining processor capacity is used for the purpose of monitoring, operator dialogue, and for generating the flight test protocol.

The dialogue process receives commands from the system operator, the command line interpreter decodes them and modifies control variables. These variables are controlling the execution of process-parts (see FIG.5) and the data flow between different processes, like sampling of realtime data probes for monitoring, and starting of calibration routines, etc.

Several system parameters are organized in parameter tables. These tables describe the attributes of sensor signals, the arrangement of monitoring menus, the arrangement of the flight test protocol, etc. The parameters of these tables may be edited by the user with a special service program before flight. All elements of standard sensors are predefined and supplied to the user.

The Acronym table contains the following information about each parameter:

- signal name , unit
- sensor offset, gain
- second order coefficient, if necessary
- data type of raw-integer

Furthermore there are :

- graphic menue parameter tables
- alphanumeric menu parameter tables
- printer protocol table
- data recording table

The tables are loaded during the initialization part of the on-board software and are printed once into the flight protocol for the purpose of flight test documentation.

After receiption of a raw data frame at the beginning of a time share, raw data have first to be decoded. Because the sensor data are collected from different sensor types, the raw data frame represents an inhomogeneous sum of different raw data structures.

The task of raw data decoding is to transfer all sensor signals to a consistent data representation. The result of frame decoding is a so called raw-integer vector, which contains exactly one 32-bit two's complement number for each data channel. The position index of the data channel is equal to the acronym-number of the channel.

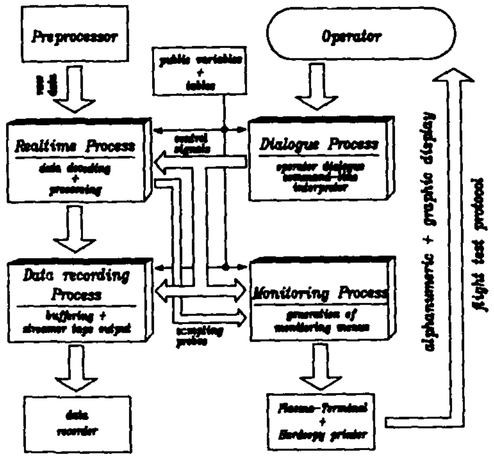


FIG.5 : Signal flow of MODAMS

Next step of the realtime processing is the generation of primary physical data by multiplication of a gain vector and addition of an offset vector. A linear converting algorithm is adequate for most of the sensor signals. Nonlinear conversion is not a part of the regular scaling algorithm; it has to be as simple and quick as possible. If nonlinear equations are to be calculated for the standard sensor equipment (e.g. for semicoductor temperature probes), there is a module reserved, where these equations can be explicitly programmed (FIG.6). After the conversion to engineering units has taken place, there is one real*32bit element in the output-vector for each sensor channel including all the spare channels of the preprocessor. This output (index from 1..n, FIG.6) is called primary output data.

The elements with higher index numbers (n+1 until n+m) are reserved for secondary data, which are calculated from primary data in the user module. The user's algorithms may represent simple combinations of primary data, or they ar more complicated like observers and Kalman filters.

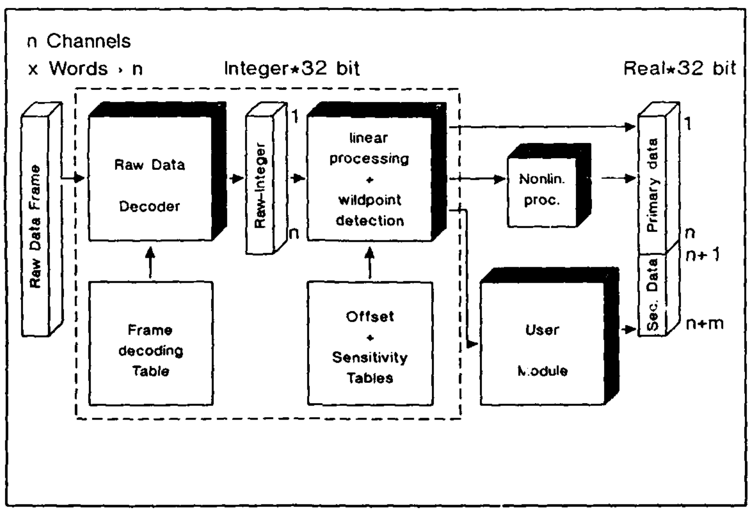


FIG. 6: Data Flow of the Realtime Process

The dashed line of FIG. 6 encloses those parts, which are processed by the vector processor, if it is available in the system configuration.

The user gets information about data structures and system conventions from a hand-book and is able to integrate his algorithms in a rather simple way. This means: he is freed from system-specific work and can take all his efforts for the solution of his own problems.

Due to the fact that primary and secondary data are situated in the same data structure, they can be treated equally for all further processing like monitoring and data registration. The attributes of secondary data are decribed by the same acronym-table in the same way with only one exception: there is no sense for an allocation of raw data type, the sensitivity and offset-attributes, because no raw data equivalent exists for a secondary datum.

The remaining CPU-time is shared by several processes of lower priority:

- an operator dialogue process asks for operator requests

- a monitoring process is continuously displaying a system status menu, alphanumeric and graphic menus, which were selected i. the operator dialogue.

6. OPERATING CONCEPT

The system may be operated from a single terminal. For this purpose the operator console is divided into three windows (FIG.7), which are assigned to different processes. The upper left window (1) is used like a normal small terminal for the dialogue with the operator. The upper right field (2) is a status window, continuously keeping all important system states, e.g. status of the data transfer from the pre-processor, realtime load, time information, event counter, and status of data recording.

The large window, the so called monitoring window (3), is used for alphanumeric and graphic monitoring.

A command line interpreter (CLI) decodes operator inputs. User interfaces are defined for the CLI as well. Users can add a private subset of commands and dialogue-procedures. This is a very helpful feature, it enables the control of the user realtime parts, whatever they may do.

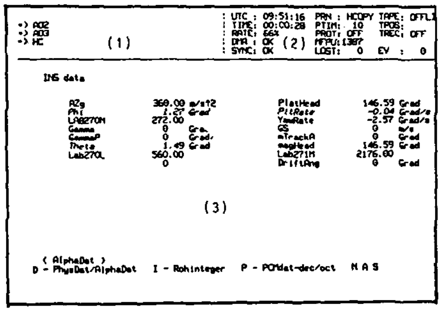


FIG. 7 : Hardcopy of an alphanumeric monitoring menu

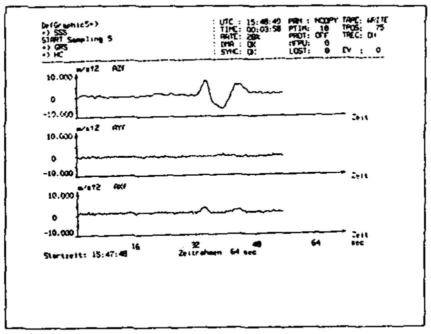


FIG. 8 : Hardcopy of a graphic monitoring menu

The operator has the choice between several monitoring menus: 10 alphanumeric and 6 graphic menus are accessible. FIG. 7 shows the example of an alphanumeric menu.

An Alphanumeric menu shows up to 32 signal channels with their names, values, and units. A graphic menu up to three channels versus time or two channels plotted versus another channel. FIG.8 demonstrates a graphic menu with 3 channels displayed versus time.

Principally each graphic menu is supplied with data buffers for each of its channels. Sampling of menu data is independent from menu display. Several menus may be sampled at the same time, but only one menu with up to 3 channels may be displayed at the time. Sampling of a menu is started with the "start sampling" instruction of the operator dialogue.

Graphic menus can be operated in two different modes:

Graphic menus can be operated in two different modes. The modes lead to a different display handling, if the sampling process arrives at the time maximum, which was selected for the menu. This is the moment, when the sampling buffer becomes full, or in other words, the displayed curves arrive at the right limit of the display area. Depending on the selected mode for the menu, the following happens:

1. Single sample mode: After the sampling buffer has been filled, sampling is stopped. The Menu may be displayed as long and as often as required until a new sampling is started by the operator.

2. Infinite mode: The whole curve is shifted half a time-axis to the left, the first half is lost, absolute starting time is updated, and sampling continues. This mode is only selectable, if channels are displayed versus time.

In the normal case, the arrangement of the different menus has been selected before flight and remains constant. However, in the case of sensor faults it may be necessary to change the display mode or the scaling-limits of a menu. For this purpose, a definition dialogue is designated to change actual menu parameters even during flight.
A flight test protocol has to give a brief overview about the relations of the most important flight parameters like time, position, and height. Therefore it is a guide for the later data evaluation.

Before takeoff, some important static information are documented in the flight test protocol: Date, program version, contents of parameter tables, altimeter setting, initialization data, etc.

During flight, a data record is periodically printed and contains the following information: Time, position coordinates, altitude, speed, course, true airspeed and a list of user selectable data from the data output vector. The time interval is selected by the user. If the operator hits the so called 'Event-key', an event-counter in the status display is incremented and an additional record, which is marked with an event-identifier. Additionally, a one-by-one pixel-hardcopy of the actual display contents can be sent into the flight protocol at any time, including all monitoring information.

In case of measuring heavy reproduceable phenomena like weather, it is important to have a discussion base immediately after landing, which enables the scientists to decide, if something has to be repeated or not. The flight test protocol can be an illustrative material to do this decision.

7. APPLICATION EXAMPLES

The presented airborne data aqcquisition and monitoring system and its derivatives have been installed on different research aircraft. They are presently used for the following applications:

- Research aircraft Dornier 128-6 of the Technical University of Braunschweig, FRG, twin engine turboprop, use for flight guidance, flight-mechanical, meteorological and air-chemical research.

- Research aircraft Dornier 228 of the Alfred Wegener Institute for Polar Research, Bremerhaven, FRG; two twin engine turboprops used for research programs in arctic and antarctic regions.

- Hawker Siddeley HS-125, twin engine jet of Conti Flug GmbH, used by Aerodata GmbH and Fraunhofer Institute of Atmospheric Research, FRG for air-chemical investigations. This jet will participate in projects of the US Environmental Protection Agency in autumn 1988.

- Research aircraft Dassault FALCON-2 of the German Aerospace Research Organisation (DFVLR, Oberpfaffenhofen, FRG), twin engine jet. This aircraft is mainly used for meteorological research.

6. CONCLUSIONS

The presented hardware structure of an airborne measuring system enables powerful on-line computation and monitoring with state-of-the-art microcomputers. A special Software has been developed for this purpose.

This software system satisfies the basic requirements of on-line computation, monitoring, and storage of primary data. The system user has the possibility to add his individual procedures. It is not necessary for him to have detailed knowledge about the internal system structure.

On-line monitoring during flight allows immediate verification of flight conditions and sensor signal quality. The operator can directly influence the actual flight test and optimize measuring conditions by the selection of better suited flight levels and operating areas.

FLIGHT TEST INSTRUMENTATION AND DATA PROCESSING AT BRITISH AEROSPACE, WARTON, U.K.

AUTHOR ROGER E. PORTER
HEAD OF FLIGHT TEST INSTRUMENTATION DATA PROCESSING
BRITISH AEROSPACE PLC
WARTON

CO-AUTHOR MICHAEL F. ARMOUR
SENIOR FLIGHT TEST INSTRUMENTATION ENGINEER
BRITISH AEROSPACE PLC
WARTON

SUMMARY

British Aerospace Warton has been engaged in the development and testing of a wide range of Military Fighter Aircraft including Jaguar, Tornado and the E.A.P. During the next decade Warton will be a major flight test centre for the European Fighter Aircraft E.F.A.

This paper gives an overview of the Instrumentation and Data Processing facilities in current use, planned development for the next 10 years covering the proposed instrumentation system for E.F.A. It describes the ground station, computing hardware, specialised analysis capability, onboard equipments, the extensive use of telemetry for real time processing and intermanoeuvre analysis in real time. Also included are the facilities for integration of airborne data acquisition systems, together with procurement, automated calibration and design and development.

CONTENTS

FLIGHT TEST INSTRUMENTATION DEPARTMENT RESPONSIBILITIES

- Airborne Data Acquisition, Recording and Telemetry Transmission systems on JAGUAR, TORNADO, E.A.P. and E.F.A. aircraft.
- Telemetry Aerials and Receiving equipment.
- Telemetry Monitor Room.
- Ground Replay and Data Analysis Computers.

ADVANTAGES OF TELEMETRY

- Certainty that expected results are captured.
- Quality of manoeuvre fit for purposes.
- Option of repeat.
- Certainty of Instrumentation parameters are serviceable.
- Ability to go to optional tests.
- Flutter clearance in bigger steps (in quasi real-time using Hewlett Packard).
- Load monitoring in critical conditions (limited number of parameters).
- Pilot confidence of having Engineer in the cockpit, especially early in the programme.
- Quick appreciation of results during/immediately after flight allows better decisions on next flight activity etc.

TELEMETRY AT WARTON
HISTORY

LIGHTNING	SPINNING	1950's	8 Analogue parameters
JAGUAR	SPINNING	1969/1972	14 Analogue parameters
TORNADO		1970's	4K Frame 512 Analogue 240 Bi-level events 16 Frequencies 64 mixed words from Main Computer 64 D.T.L. mixed words
E.A.P. Experimental Aircraft Programme		1980's	2 * TORNADO DATA RATES
E.F.A. European Fighter Aircraft		1990's	4 * E.A.P. DATA RATES

SYSTEM DEVELOPMENT FOR TORNADO

1980 o Decision to replace Sigma computer by VAX system.

1982 o Commenced software development of BAe system based on VAX 11/780 (first VAX installed second half of 1982).

1984 o Sigma's removed, second VAX 11/780 installed - live VAX Telemetry.

(Dec) o Telemetry 11/780 upgraded to 11/785 giving 60% improvement in throughput.

New Monitor included:-
- New console
- New V.D.U's
- New pen recorders

late 1985
- Aircraft position display.

TORNADO

The Inter Dictor Strike Development Programme.

1973 - choice made by three Partner Companies to purchase identical systems based on the U.S. Navy system at Pax River. Xerox Sigma computer system complete with software.

1st Sigma in 1974
2nd in 1976

Antenna System from Scientific Atlanta (1974) - Autotracking in Azimuth and Elevation.

DEVELOPMENT OF TRACKING SYSTEM

1985 o Electronics Department developed software on BBc Micro to provide:-

Aircraft Bearing
Heading
Range
Altitude

- all derived from the St. Annes secondary radar via Warton Control Tower land lines.

Alternatively the aircraft Inertial Navigator position data can now be extracted from the Telemetry data stream.

1986 o Replacement Main Dish for Scientific Atlanta fixed in position for a total cost of £18,000 giving full 360 degrees coverage.

FLIGHT TEST INSTRUMENTATION AIRBORNE DATA GATHERING SYSTEMS
INTRODUCTION

Since the Flight Test Instrumentation department of British Aerospace Warton was formed in 1958, a large number of diverse recording techniques and transducer types have been used in six generations of military aircraft produced by the Division.

The earliest system, developed in the late 50's, employed 'Auto-observers'. These were, as their name suggests, a method of automatically recording the readings of a bank of aircraft gauges, the recording medium being cine film. (Prior to this, and indeed concurrently, aircrew/observers were obliged to note down gauge readings during periods of interest on note-pads). Auto-observers were very bulky and required fitting in spacious accommodation ... in this case the bomb-bay of Canberra jet bombers. A typical limit on the number of parameters was around 20 to 30.

The advent of the Lightning interceptor in the early 60's introduced the concept of space constraints and trace recorders were used for the first time, offering much needed size reduction and vastly improved duration and frequency response.

The TSR2 bomber of the middle 60's employed a combination of trace and analogue recording techniques. For the first time broadband FM was used, mainly for vibration measurements, complemented by a 400 parameter mechanically-switched time division multiplex system also recorded on tape. Predictably, the main weakness of the system was the mechanical switching.

By the late 60's, a combined digital/analogue and trace recording system was used in the early Jaguar Flight Test program. The analogue (FM) and trace systems were similar to those employed on the TSR2 aircraft, however, the mechanically-switched TDM system was replaced by an 8 bit parallel PCM technique (i.e. each of the 8 bits were recorded on a separate track of the tape recorder). The system had a capacity of 144 analogues, along with bi-level and frequency inputs.

CURRENT SYSTEMS

1. Air Defence Variant Tornado (ADV)

The instrumentation system currently used on the ADV Tornado prototypes was developed for the Inter Dictor/Strike (IDS) Tornado prototypes in 1974. (Figure 1).

MAJOR SYSTEM COMPONENTS

Tape Recorder

A 14 track wideband IRIG-standard tape recorder using 1" tape on 14" reels fitted with two Direct Record (DR) tracks and 12 FM wideband 1 tracks. At the usual speed of 7.5 ips, this gives a duration of 4 hours and bandwidths of:

Direct Record : 500Hz to 125KHz
FM wideband 1 : dc to 5KHz

Data Collector (CCU)

The Central Control Unit of the data gathering system has the following capacities:
8 PAM outputs from 8 Signal Conditioner racks (8 x 64 analogues)
16 Frequency inputs
4 Totaliser inputs
24 parallel words inputs (or 24 x 10 events)

the unit provides:
a serial Bi-phase PCM data stream (for recording)
a serial NRZ PCM data stream (for telemetry)

format:	word length:	10 bits + odd parity
	max data field:	128 words/minor frame
		32 minor frames/major frame
		1 major frame/second
	bit rate:	45Kbits/second
	packing density:	6Kbits/inch at 7.5 ips

Signal Conditioner Racks

Each signal conditioner rack houses 64 signal conditioners along with voltage stabiliser and multiplexer cards providing a 64 parameter PAM stream. 29 analogue outputs are available for recording as FM or FM/FM (Frequency Multiplexed/FM) parameters.

AIIU (Avionic Information IFU)

The AIIU takes information from the aircraft main computer and inserts the parameters into the existing PCM data stream. Every 1/16th second 63 parameters are stored in the AIIU and the CCU selects the required parameters from this store.

(ie) max. sample rate: 16 samples/second
max. no. parameters: 63 x 16

Each avionic word is 32 bits long and each word is reduced to 2 x 10 bit words for recording on the FTI system.

FM/FM System

This is a Frequency Multiplex/FM system employing:

21 constant bandwidth subcarrier oscillators BW: 200Hz (mod. index = 5)
1 wideband subcarrier oscillator BW: 100KHz (mod. index = 2)
5 mixer amplifiers
Calibration facilities

The constant BW subcarriers are modulated by the analogue parameters, whilst the wideband subcarrier is modulated by the NRZ PCM serial output of the data collector.

Outputs are therefore available for recording on the FTI tape recorder and for modulation of the telemetry transmitters.

The number of analogue parameters handled by the system is increased by a factor of 6 by the use of group select electronics.

Time Code Generator

The Time Code Generator is a self-contained unit which produces IRIGB serial timecode for use by the data collector and for recording on the tape recorder. Outputs are also available for cockpit time displays.

Telemetry

Two telemetry transmitters, each on a different frequency, drive two separate antennae (one on the aircraft top surface, one on the lower).

Camera System

A cine camera system is fitted for weapon release analysis. Eight cine cameras are used overall, of which six are phase-locked to a central control clock and housed in two external wing pods. All cameras are individually idented with externally generated events ('weapon gone' etc.) and a common timebase. Camera frame pulses and the timebase are recorded as FM parameters on the tape recorder.

System Variants

Two of the ADV prototypes carry out radar evaluation work, and as a result have need to record high frequency radar data. This is accomplished by replacing the 14 track tape recorder with a 28 track model, and by using 10 of the extra 14 tracks (DR) to parallel radar data. The recorder is run at 4 times normal speed (30ips), the packing density being 16Kbits/in.

2. Production IDS Tornado

The production Tornado instrumentation system was defined in 1980. At this point in the flight testing program, the number of parameters required for analysis had reduced, as had the available space for mounting instrumentation boxes - due in main to the production avionics fit.

A great reduction in the size of the data collector and the signal conditioner racks has been achieved.

Equipment Changes from ADV Fit
Data Collector

The data collector has the following capabilities:

128 analogue inputs
9 frequency inputs
2 totaliser inputs
50 event inputs

The unit provides the following outputs:

IRIGB timecode
a serial Bi-phase PCM data stream (for recording)
a serial NRZ PCM data stream (for telemetry)

format:	word length:	10 bits + odd parity
	max data field:	128 words/minor frame
		32 minor frames/major frame
		1 major frame/second
	bit rate:	45Kbits/second
	packing density:	64Kbits/inch at 7.5 ips

Signal Conditioner Racks

The signal conditioner racks accept 8 signal conditioner cards (same cards as used in the ADV system), along with a power supply unit. The output of each conditioner is available in both filtered and unfiltered forms, rather than a serial PAM stream.

Advantages of this approach include small size, relative simplicity and loss of only 8 parameters in the event of a power supply failure.

Avionic Tape Recorder Interface (ATRIU)

The ATRIU is often used in systems as an alternative to the AIIU where space and cost considerations outweigh the flexibility given by the AIIU. The unit takes data from the aircraft main computer and reformats it for recording on the FTI tape recorder.

The data rate from the main computer is 1953 words/second, each word being 32 bits long (i.e) 62.5Kbits/sec. The ATRIU extracts the 16 data bits from each of these words, converts them into two words and outputs them in serial Bi-phase form at a nominal 32Kbits/sec.

Camera System

The camera system is similar to that fitted on the ADV prototypes except that eight of the ten cameras are carried in four external wing pods (inboard/outboard). The cameras are all free-running (non phase-locked).

3. Experimental Aircraft Program (EAP)

(Figure 3)

The instrumentation system for EAP was defined in 1983. On this particular aircraft the trend towards more data bus monitoring has continued, and needless to say, the space available for mounting FTI has reduced still further.

The size of the data collector has been greatly reduced from that used in the IDS Tornado, while the analogue signal conditioning units have returned to a 64 channel capacity, the size reduction being mainly accomplished by mounting 4 signal conditioning channels on one card (as opposed to using single channel modules) and by using hybrid/surface mount circuitry.

Major System Components

Data Collector

The data collector selects data from analogue and digital slave units along with bus information from a data bus interface unit.

The data collector has the following capacities:

3 serial outputs from 3 Analogue Acquisition units (192 parameters)
8 frequency inputs (from the Digital interface unit)
30 event inputs (from the Digital interface unit)

the unit provides:

a serial Bi-phase PCM data stream (for recording)
a serial NRZ PCM data stream (for telemetry)

format:	word length:	10 bits + odd parity
	max data field:	256 words/minor frame
		32 minor frames/major frame
		1 major frame/second
	bit rate:	90.1Kbits/second
	packing density:	12Kbits/inch at 7.5 ips

Analogue Acquisition Units (AU)

Each AU accepts 64 analogue parameters, conditions and then multiplexes them into a serial data stream for transmission to the data collector. Each word is 10 bits long.

Digital Interface Unit

This unit accepts data from a Flight Control System (FCS) computer. Protocol is RS422, not 1553 data bus. Up to 124 data words at rates up to 50/second can be acquired and stored in the unit. The data collector extracts the required parameters at rates from 1 to 64 samples/sec.

FCS data words are 16 bits long. These are split into two 8 bit bytes and occupy two PCM words in the data collector output stream.

The unit also houses 8 frequency counters and a 30 input event card.

Data Bus Monitor

The Data bus monitor contains two dual-redundant 1553 databus monitors. It has no ability to transmit on the bus nd plays no active role in data bus activities.

The data is processed in a similar way to the Digital Interface unit. (Up to 124 data bus words at rates up to 50/second can be acquired from each data bus).

Other major system components are similar to those described for Tornado.

4. European Fighter Aircraft (EFA)

(Figure 4)

The EFA data acquisition system is currently at the specification stage. However, the main requirements of the system have been defined and are described below.

Architecture

The PCM Central Control (PCC) will accept data from Remote Data Acquisition Units (RDAU) via a maximum of 8 FTI data buses. Each bus will accept up to 8 mixed function RDAUs, carrying data at rates up to 50K words/second and will have the capability of addressing 2000 parameters. This implies a maximum sampling rate of 8192 samples/second.

Major System Components

PCM Central Control (PCC)

The unit will be able to communicate with RDAUs over a distance of at least 30 metres.

7 outputs will be available:

4 Bi-phase IRIG standard PCM streams (to tape recorder)
1 NRZ IRIG standard PCM stream (to telemetry transmitter)
1 (probably RS422) PCM stream to video IFU (for data insertion)
1 (probably RS422) data stream to FTI cockpit display

format:	word length:	max 12 bits + parity
	max data field:	256 words/minor frame
		64 minor frames/major frame
		1 to 8 major frames/second
	bit rate:	500Kbits/second*
	packing density:	16.5Kbits/in at 15 ips (2 tracks)

* anticipated maximum required

It will store 6 programs to configure the RDAUs and to format the data on the 7 output streams.

Data from 16 parameters will be able to be converted to engineering units for output to the multifunction cockpit display and video processor.

Self test and program verification facilities will also be available.

Remote Data Acquisition Units (RDAU)

These units will be of modular construction.

A maximum of 64 analogue parameters will be acquired and conditioned by the unit, with a maximum sampling rate of 8192 samples/second.

Signal conditioner gain, offset and filter cut-off will be programmable (via the PCC).

Transducer excitation will be provided by one of the modules.

The RDAUs will also house Data Bus Processing Modules (DBPM).

Data Bus Processing Modules (DBPM)

Three types of module will be available to capture data from the following data bus types:

MIL-STD 1553B (STANAG 3838)	6 buses
STANAG 3910	2 buses
FCS CCDL	4 buses

The DBPMs will operate as 'eavesdroppers' only and will be able to extract 256 parameters from the bus.

MIL-STD 1553B modules will connect to dual-redundant buses via two channels of opto-isolation in different modules. They will also be programmable to output the total bus for recording via the Bus-PCM-Video interface.

Time tagging will be available.

Tape Recorder

A conventional Tornado type 14 track tape recorder will be used.

Current anticipated bit rates from the PCC (500Kbits/sec) will be accommodated by using only two of the available four PCC PCM streams. With the recorder running at 15ips, a duration of 2 hours is achieved with a half-theoretical maximum bit packing density.

Typical track allocation:	2 PCM (DR)
	1 Timecode (WB 1)
	1 Speech (WB 1)
	1 Tape speed reference (WB 1)
	9 Analogue parameters (WB 1)

Telemetry

The telemetry system will be dual channel/space diversity (as Tornado). Unlike Tornado, the PCM will be on base-band, with sub-carriers for speech and events. The transmission will be encrypted.

Video Transmission

It is proposed to use a single 10 watt transmitter modulated by a CCIR standard video signal (encrypted). The video source will be one of:

FTI video camera
Aircraft video source
Aircraft databus (via Bus-PCM-video IFU)

Time Code System

The Time Code generator will output IRIG A or B (selectable) and will give elapsed time from 'START' or, by using an appropriate interface, will be synchronised to the National Time Standard to give absolute time.

The time code will be decoded by the PCC and be available in any of the PCM streams for recording/transmission/display.

The system will be integrated with a Global Positioning System (GPS) if available.

Bus-PCM-Video Interface

This is a contingency requirement for recording the entire contents of a MIL-STD 1553B data bus. The bus information will be reformatted in a form enabling it to be recorded on a standard video recorder or on up to 4 tracks of the conventional tape recorder.

Multifunction Cockpit Display

This display will be a software-configured control and display panel upon which FTI status information will be available, along with the display of up to 5 FTI parameters in engineering units.

Camera System

Up to ten phase-locked cameras are expected to be used, with improved identing (on-film numeric time display) and increased flexibility of control (multiple groups/programs).

FUTURE TRENDS

Although the first EFA prototypes will use longitudinal type conventional tape recorders, the second generation test aircraft will be likely to use helical-scan recorders. These will be particularly suited to handling high speed data such as radar instrumentation. Eventually, optical recording techniques will probably become the dominant recording method.

Increased use of real-time analysis is an inevitable development, allowing as it does the sequencing of various inter-dependent test-points during one test flight. Work of this type (vibration analysis) is already carried out as part of the EAP flight testing program.

High speed/high definition video will eventually replace wet-film cine cameras for weapon release work. Many advantages will accrue from this, such as the reduction in required electrical power (high speed cine cameras are notoriously greedy for power), increased recording duration (hours instead of seconds) and ease of installation.

DATA RATES
TORNADO

10 Bits/Word + Parity (ODD)
128 Words/Main Frame
32 Main Frames/Major Frame
1 Major Frame/Second

e.g. 128 * 32 = 4096 Words/Second

DATA RATE = 11 * 4096 = 45,056 Bits/Second

N.B. Each Main Computer or Data Transmission Line (T.T.L.) word is 16 bits hence require 2 Instrumentation words (10 & 6 bits).

E.A.P

Data rate twice that of Tornado.

Majority of parameters are from the three MIL 1553 data buses and are present as 16 bit aircraft words.

Because the Instrumentation word has only 10 bits all data bus words are split into a Coarse and a Fine half. Each half contains eight data and two status bits.

CURRENT
E.A.P

- Data Rate twice that of Tornado
- New configuration display
- Warning panel display
  - MIMIC of cockpit display
- Link to Flight Mechanics Computer (F.M.C.)
- Workstations
  - Microvax based

Unlike Tornado, attempting to access the loads model in quasi real time.

- Link to Flight Mechanics Computer for loads Monitoring.

Slow data sampled at twice/second throughout flight (6 parameters) gives Aerodynamicist assessment of flight conditions.

From Pilots countdown, 15 seconds worth of 30 to 40 parameters captured by VAX and then burst sent to F.M.C.

Based on the data recovered, calculated loads at various stations in aircraft from control inputs and aircraft response. From that it is determined if the loads are as predicted.

MICROVAX STATION IN TELEMETRY ROOM

- Performs Z Transform analysis

From Pilots countdown 15 seconds worth of data (6 parameters) is captured by the Microvax. This is displayed as short time history.

Sample of data required for analysis selected by screen cursor.

Results arrive within two minutes of manoeuvre, giving frequency and damping from Dutch Roll (typical) using small pedal input compared with prediction.

Engineer can then assess whether to proceed to next step using the output plots and tables.

AIRBORNE INSTRUMENTATION SYSTEMS
FM WIDE BAND

- 9 Tracks
- Frequency Range
  - DC to 5 KHz

FM/FM SYSTEM

- 21 Channel V.C.O. & Frequency Mixer
  - Converts voltage (±2.5V) to frequency (Centre Frequency ±1KHz)
  - 4 KHz separation (e.g. 8, 12, 16, ... 88 KHz)
  - 2 KHz Bandwidth
  - Frequency Range is DC to 200 Hz

PULSE CODE MODULATION (P.C.M.)

- Sensors
  - MIL 1553 Data Bus Interface(s)
    U.S.M.S.
    Avionics
    F.C.S.
  - Events e.g. Undercarriage Up/Down ...
  - Frequencies e.g. Engine Turbine Speed ...
  - Transducers e.g. Fuel, Pressures, Angles ...
  - System Tappings e.g. Hydraulic Pressure
- Data Acquisition (Central Control Unit)
  - Signal Conditioning
  - A/D Conversion
  - Multiplexer
  - Serial P.C.M. outputs Bi-O & NRZ-L

TAPE RECORDER

- o 14 (MARS 1400) or 28 (MARS 2000) Tracks
  - P.C.M. Bi-0
  - FM (9 Tracks)
  - FM/FM
  - Speech
  - Time Code IRIG B Standard (1 KHz Carrier)

DATA TRANSMISSION

- o High Frequency Voltage Control Oscillator (V.C.O.)
  - Adds high frequency Carrier to (600 KHz) P.C.M.
- o Combiner
  - Mixes Speech, P.C.M. and FM/FM
- o FM Transmitter
  - Adds MHz Carrier
  - Frequency Range 1430 MHz to 1540 MHz
  - Modulation Type is FM
  - 2 per aircraft
    One for each Antenna (mounted below and above spine of aircraft)

GROUND SYSTEM
ANTENNA

- o 6 foot Dish
- o Bandwidth 1438 MHz to 1540 MHz
  D Band (previously known as L Band)
- o Coverage from West Freugh down to South of Anglesey
- o Manual tracking
- o Standby Antenna exactly same as main Antenna
- o Research by BAe Electronics Department resulted in:-

Dish	from	Andrews (Worthing)
Gearbox)		Dennords (Fleet)
Drive)		
Pedestal		Sadlers (Lytham)
Electronics		BAe (Warton)

Total cost (1983) £30,000

RECEIVERS

- o Removes MHz Carrier
- o 4 Receivers in all (require only 2 per aircraft)
- o All 4 configured for D Band 1435 KHz to 1540 KHz
- o Crystal control for only allocated frequencies, but may tune to any frequency within D Band.

FILTERS

- o HI-PASS
  - Removes all Frequencies < 200 KHz
- o LO-PASS
  - Removes all Frequencies > 200 KHz

21 CHANNEL DISCRIMINATOR

- o Separation of 4 KHz (8, 12, 16 .... 88 KHz)
- o Also contains LO-PASS filter (< 2 KHz) for Speech

HI-FREQUENCY DISCRIMINATOR

- o Removes 600 KHz carrier from P.C.M.

EMR 720 BIT SYNCHRONISER

Eliminates "noise" from serial P.C.M. data stream.

- Data Stream
  - NRZ-L P.C.M. from Telemetry
  - Bi-O P.C.M. from Primary aircraft tape
- Bit Rate
  - 45 Kbits/sec for TORNADO
  - 90 Kbits/sec for E.A.P.

EMR 710 FRAME FORMATTER

Separates data words from frame synchronisation words and then converts the serial stream into a parallel data stream.

BAe VAX INTERFACE

"Holes" left by the frame synchronisation words are filled with time and quality words.

The derived parallel stream is input to the VAX computer.

VAX 8600 COMPUTER

Reads data from the DR11B buffers and drives the various telem.try monitor room displays.

Generates Secondary Tape for Post Flight Analysis.

ICL COMPUTER

Primary function to perform analysis in Batch mode.

MONITOR ROOM

- Front Console
  - Meters
  - Event lights
  - Numerical tubes
  - Configuration panel
- Engineer Stations
  - Alphanumeric V.D.U.'s
  - Heat Sensitive pens (8 track)
- Spinning Station
  - Pen bank
  - Traffic lights
  - Stick Position display

DESCRIPTION OF TELEMETRY SOFTWARE PROCESS

1. SET-UP

   Loading of calibrations and decommutation of data into memory for subsequent programming of Front End units.

2. EXTRACTION

   Selecting required parameters for display from total transmitted by aircraft.

3. CALIBRATION

   Converting raw data into engineering units by applying calibrations to selected parameters.

4. ELABORATION

   Combination of several parameters to produce new ones such as airspeed, mach number etc.

5. DISPLAY

   The direction of calibrated parameters to analogue outputs, screens, cross plotters etc.

10 man-years of software writing to produce the Real-Time program comprising the above modules. Total system contains approx. 400 modules:

Real time - 250
Z-Transform - 150

GROUND REPLAY AND DATA ANALYSIS COMPUTERS
PRIMARY ANALYSIS

- o VAX 8600
  - Telemetry capability plus quick-look and secondary tape making.
  - KDX-11 link to Flight Mechanic's Computer.
- o VAX 11/785
  - Can be used for telemetry in place of 8600, but mainly used for quick-look and secondary tape making.
- o The two VAX computers are connected together and to the Telemetry Monitor Room Microvax via an Ethernet.

KDX-11 LINK

- o Continuous stream
  - 20 parameters sampled twice per second are passed continuously throughout flight. Transfer rate of 160 bytes per second.
- o Higher density burst
  - Post manoeuvre transfer.
    32 parameters sampled at 32 times per second, each for a 15 second period. data held in memory during manoeuvre.
- o Immediately after manoeuvre, data transmitted to Flight Mechanics Computer via the link at an average rate of 5000 bytes per second.

SECONDARY ANALYSIS

- o ICL 2966 is connected to a terminal network in Flight Test and Instrumentation departments to provide the following facilities:
  - Secondary Analysis system (560 modules) which allows F/T engineers to initiate jobs to run in Batch mode.
  - Provides Tertiary tapes to the central NAS XS-80 computer and to external companies.
  - Flight Diary database
  - Results database
  - Pre-flight software and database
  - Instrumentation Stores database
  - General Word Processing Facilities

FUTURE

Telemetry on two aircraft simultaneously

- Two Telemetry Rooms, VAX 8600 running one and the VAX 785 running the other.

Real Time Analysis

- Replacement of ICL 2966 with large VAX computer (current contender, VAX 8820) connected into existing VAX network via the Ethernet.
- Installation of VAX workstations and terminals throughout the Flight Test department to provide rapid access to flight data in real and quasi real time for interactive, inter-manoeuvre analysis. Workstations will have power/capacity for indepth data manipulation, analysis and presentation of results.

Displays

	NEW TEKTRONIX 4325	OLD TDS 4200
Resolution	1280 * 1024	256 * 256
Colours	256	8
Segmentation	YES	NO
3-D	FIRMWARE	SOFTWARE
Speed	Much faster	Slow in comparison

E.F.A. Data Encryption

Primarily due to information being extracted from the aircraft's six data buses and these being sampled at up to 256 times per second per parameter. There will be between 600 to 700 parameters transmitted with the data being encrypted.

- Telemetry on two aircraft simultaneously
- Real Time Analysis
- Displays
  - Faster, better graphics
  - Rolling time histories on screens
  - Cross plots
- P.C.M. Flutter
- E.F.A. data encryption

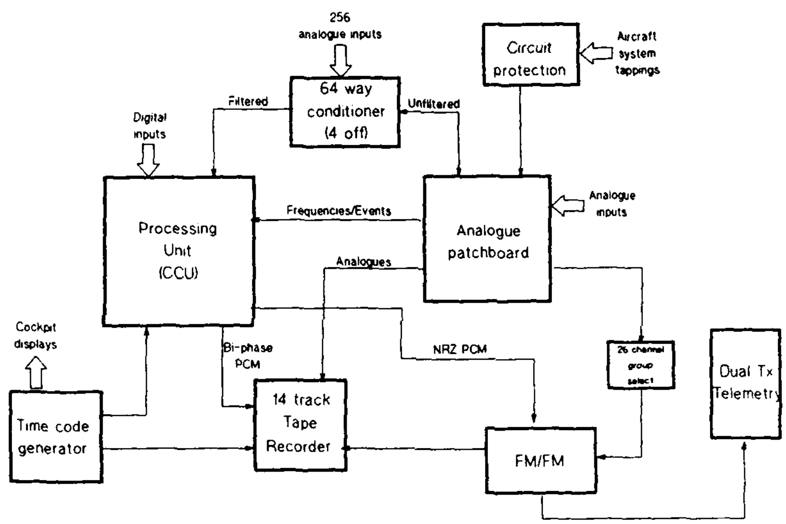


Fig.1 Prototype ADV Tornado Instrumentation system

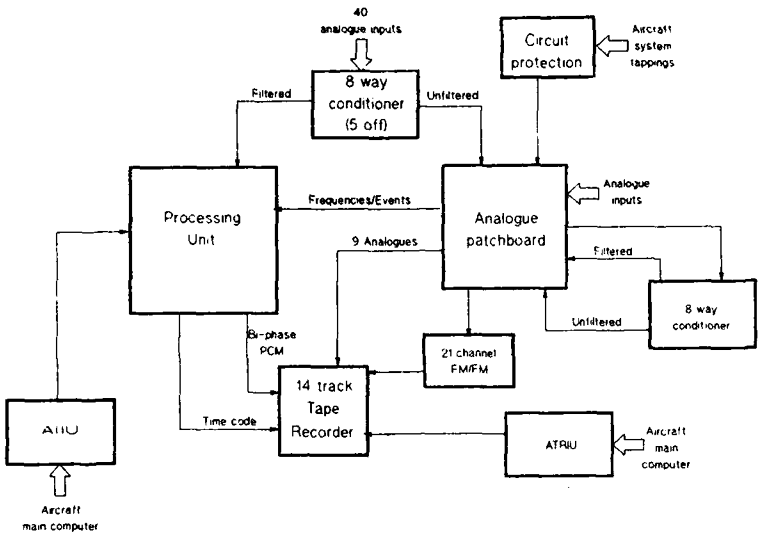


Fig.2 Production IDS Tornado
Instrumentation system

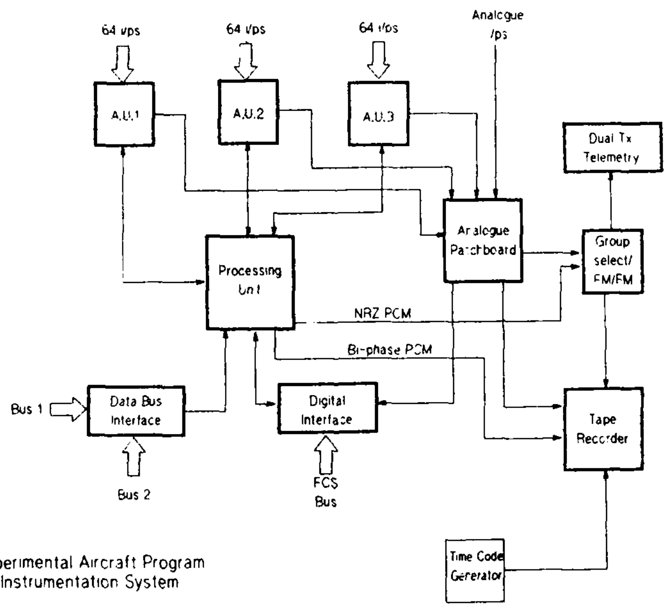


Fig.3 Experimental Aircraft Program
Instrumentation System

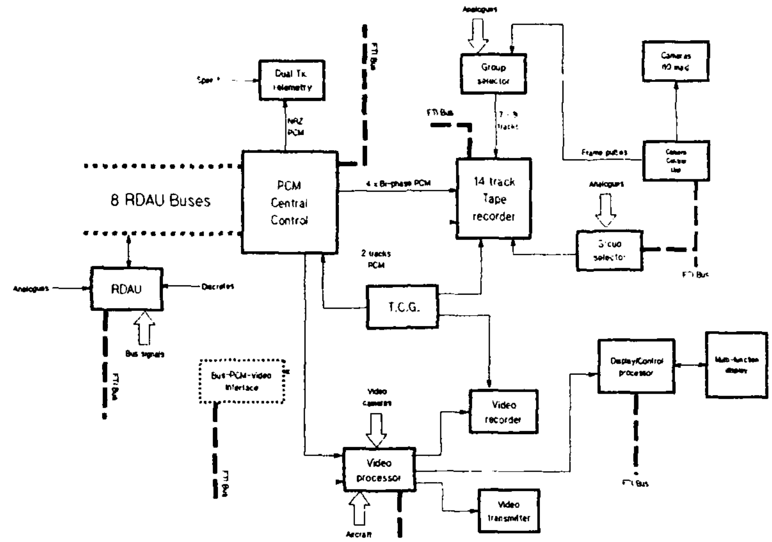
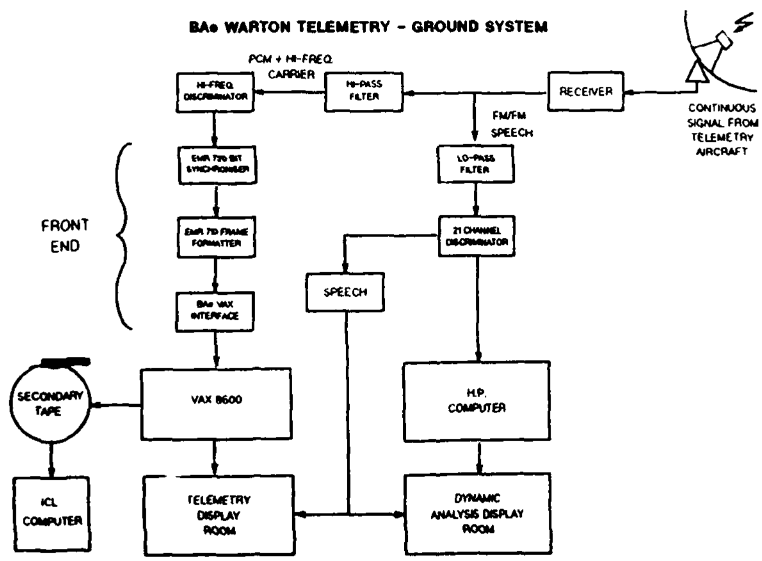
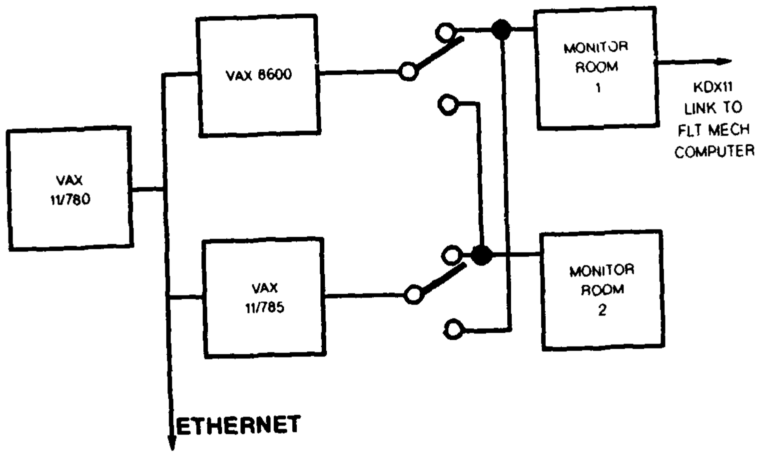


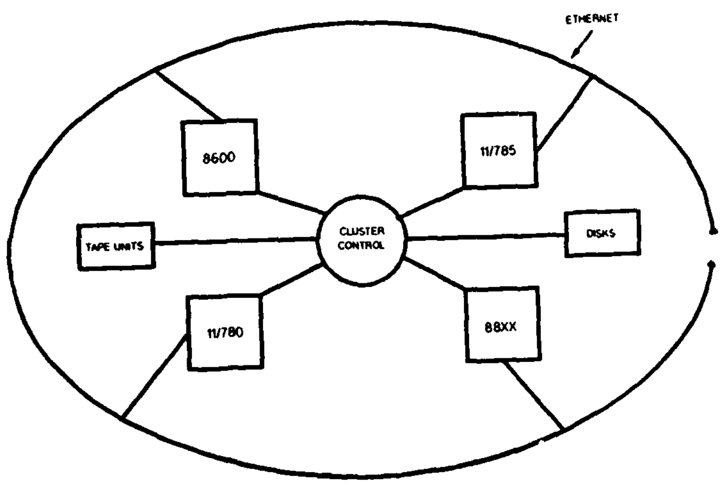
Fig.4 European Fighter Aircraft (EFA) Instrumentation system

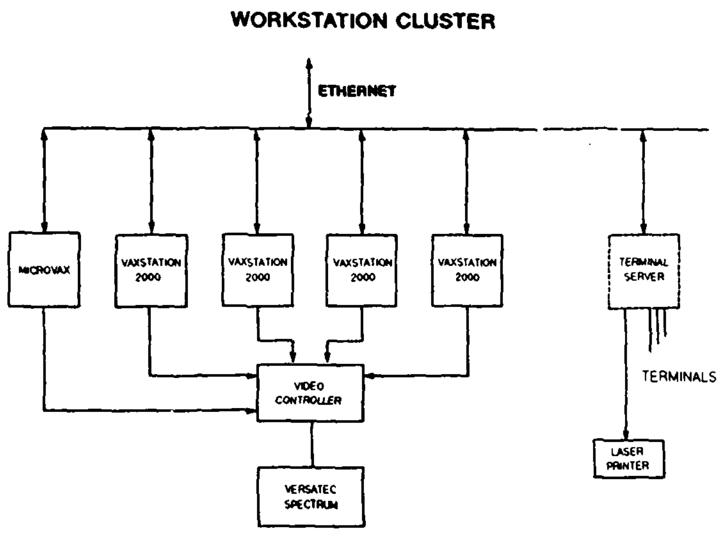


PLANNED PRIMARY ANALYSIS SYSTEM



PLANNED VAX CLUSTER




WORKSTATION CLUSTER
ETHERNET
MICROVAX
VAXSTATION 2000
VAXSTATION 2000
VAXSTATION 2000
VAXSTATION 2000
TERMINAL SERVER
TERMINALS
VIDEO CONTROLLER
VERSATEC SPECTRUM
LASER PRINTER

REPORT DOCUMENTATION PAGE			
1. Recipient's Reference	**2. Originator's Reference** AGARD-CP-452	**3. Further Reference** ISBN 92-835-0509-3	**4. Security Classification of Document** UNCLASSIFIED
5. Originator	Advisory Gr up for Aerospace Research and Development North Atlantic Treaty Organization 7 rue Ancelle, 92200 Neuilly sur Seine, France		
6. Title	FLIGHT TEST TECHNIQUES		
7. Presented at	Flight Mechanics Panel Symposium held at Edwards Air Force Base, California, USA, 17—20 October 1988		
8. Author(s)/Editor(s) Various			**9. Date** July 1989
10. Author's/Editor's Address Various			**11. Pages** 432
12. Distribution Statement	This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.		
13. Keywords/Descriptors Flight tests Flight simulation Avionics	Computer systems programs Data processing Cost engineering		

14. Abstract

The Flight Mechanics Panel believes that it is vitally important that the flight test community meet regularly so that new techniques for flight test, instrumentation and data analysis can be disseminated to ensure that safe, efficient and timely testing is accomplished.

The number of new systems being submitted for airborne trials and testing is continuously increasing. The major development and test challenge common to most current and future aircraft is that of avionic/software subsystems development and integration. New systems and applications include programmable signal process radars, integrated flight, fire and propulsion systems, multifunction pilot displays, and other on-board software-intensive systems. The acquisition and processing of large quantities of avionics multiple data is a challenge that must be met.

This Symposium focussed on the means of reducing the overall cost of test and development within defence programmes. Ground based simulation offers the potential for saving many test flying hours in the area of system integration. It is an indispensable tool for system failure analysis for sophisticated avionics of all types.

AGARD Conference Proceedings No.452
Advisory Group for Aerospace Research and Development, NATO
FLIGHT TEST TECHNIQUES
Published July 1989
432 pages

The Flight Mechanics Panel believes that it is vitally important that the flight test community meet regularly so that new techniques for flight test, instrumentation and data analysis can be disseminated to ensure that safe, efficient and timely testing is accomplished.

The number of new systems being submitted for airborne trials and testing is continuously increasing. The major development and test challenge common to most current

P.T.O

AGARD-CP-452

Flight tests
Flight simulation
Avionics
Computer systems programs
Data processing
Cost engineering

AGARD Conference Proceedings No.452
Advisory Group for Aerospace Research and Development, NATO
FLIGHT TEST TECHNIQUES
Published July 1989
432 pages

The Flight Mechanics Panel believes that it is vitally important that the flight test community meet regularly so that new techniques for flight test, instrumentation and data analysis can be disseminated to ensure that safe, efficient and timely testing is accomplished.

The number of new systems being submitted for airborne trials and testing is continuously increasing. The major development and test challenge common to most current

P.T.O

AGARD-CP-452

Flight tests
Flight simulation
Avionics
Computer systems programs
Data processing
Cost engineering

AGARD Conference Proceedings No.452
Advisory Group for Aerospace Research and Development, NATO
FLIGHT TEST TECHNIQUES
Published July 1989
432 pages

The Flight Mechanics Panel believes that it is vitally important that the flight test community meet regularly so that new techniques for flight test, instrumentation and data analysis can be disseminated to ensure that safe, efficient and timely testing is accomplished.

The number of new systems being submitted for airborne trials and testing is continuously increasing. The major development and test challenge common to most current

P.T.O

AGARD-CP-452

Flight tests
Flight simulation
Avionics
Computer systems programs
Data processing
Cost engineering

AGARD Conference Proceedings No.452
Advisory Group for Aerospace Research and Development, NATO
FLIGHT TEST TECHNIQUES
Published July 1989
432 pages

The Flight Mechanics Panel believes that it is vitally important that the flight test community meet regularly so that new techniques for flight test, instrumentation and data analysis can be disseminated to ensure that safe, efficient and timely testing is accomplished.

The number of new systems being submitted for airborne trials and testing is continuously increasing. The major development and test challenge common to most current

P.T.O

AGARD-CP-452

Flight tests
Flight simulation
Avionics
Computer systems programs
Data processing
Cost engineering

and future aircraft is that of avionic/software subsystems development and integration. New systems and applications include programmable signal process radars, integrated flight, fire and propulsion systems, multifunction pilot displays, and other on-board software-intensive systems. The acquisition and processing of large quantities of avionics multiple data is a challenge that must be met.

This Symposium focussed on the means of reducing the overall cost of test and development within defence programmes. Ground based simulation offers the potential for saving many test flying hours in the area of system integration. It is an indispensable tool for system failure analysis for sophisticated avionics of all types.

ISBN 92-835-0509-3

and future aircraft is that of avionic/software subsystems development and integration. New systems and applications include programmable signal process radars, integrated flight, fire and propulsion systems, multifunction pilot displays, and other on-board software-intensive systems. The acquisition and processing of large quantities of avionics multiple data is a challenge that must be met.

This Symposium focussed on the means of reducing the overall cost of test and development within defence programmes. Ground based simulation offers the potential for saving many test flying hours in the area of system integration. It is an indispensable tool for system failure analysis for sophisticated avionics of all types.

ISBN 92-835-0509-3

and future aircraft is that of avionic/software subsystems development and integration. New systems and applications include programmable signal process radars, integrated flight, fire and propulsion systems, multifunction pilot displays, and other on-board software-intensive systems. The acquisition and processing of large quantities of avionics multiple data is a challenge that must be met.

This Symposium focussed on the means of reducing the overall cost of test and development within defence programmes. Ground based simulation offers the potential for saving many test flying hours in the area of system integration. It is an indispensable tool for system failure analysis for sophisticated avionics of all types.

ISBN 92-835-0509-3

and future aircraft is that of avionic/software subsystems development and integration. New systems and applications include programmable signal process radars, integrated flight, fire and propulsion systems, multifunction pilot displays, and other on-board software-intensive systems. The acquisition and processing of large quantities of avionics multiple data is a challenge that must be met.

This Symposium focussed on the means of reducing the overall cost of test and development within defence programmes. Ground based simulation offers the potential for saving many test flying hours in the area of system integration. It is an indispensable tool for system failure analysis for sophisticated avionics of all types.

ISBN 92-835-0509-3