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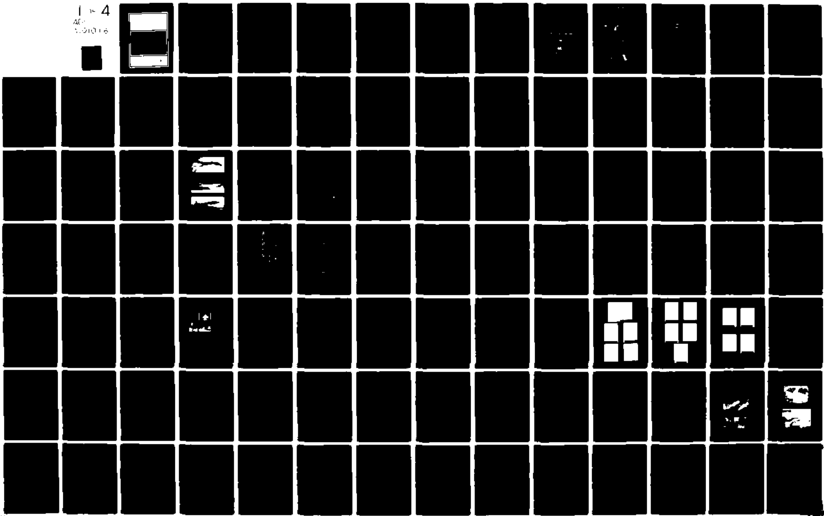

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Papers presented at the Flight Mechanics Panel Symposium on Subsystem Testing and Flight Test Instrumentation held in Geilo, Norway, 27–30 October 1980.

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- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
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PREFACE

The Flight Mechanics Panel of AGARD held a previous symposium on Flight Test Techniques in October 1976. It closed with the feeling that certain areas would benefit from further coverage; in particular that a survey of the development and proving of aircraft sub-systems, together with associated environmental testing and instrumentation facilities, was highly desirable. Thus the present symposium was conceived: it was to provide detailed information on these topics for both the flight test engineer and the flight test instrumentation engineer, and to encourage an exchange between the two.

With the range and complexity of modern aircraft systems the field was too broad to cover in its entirety. The programme was therefore directed at presenting a representative selection of the diverse problems facing today's test engineer, and illustrating more generally through these the variety of techniques which are used to deal with them. The symposium was organised around four broad subject areas –

- Navigation/Attack Systems Testing
- Aircraft Systems Testing
- Environmental Testing
- Instrumentation Techniques.

J.F.O'GARA
F.N.STOLIKER
Members, Flight Mechanics Panel

CONTENTS

* Paper not available for publication

** Paper not presented at Symposium due to illness of author, but included in this volume

TACTICAL NAVIGATION SYSTEM TESTING

by
DOUGLAS M. CARLSON
US Air Force Armament Development Test Center/TEOF
Eglin AFB, Florida 32542
USA

SUMMARY

The development of today's sophisticated and accurate navigation and weapon delivery systems has made the job of the tester increasingly difficult. These systems are being designed to accomplish the entire spectrum of tasks from pure navigation to target acquisition/identification to delivery of both guided and unguided munitions.

Most of these systems are software intensive, a fact which is altering the entire approach to development and testing. The need for thorough simulation and integration efforts is apparent in almost every program through both successes and failures.

Three systems, the AN/ARN-101 Digital Modular Avionics System, the AN/AVQ-26 Pave Tack, and the Stores Management System have all undergone development on the F-4 and/or F-111 at Eglin Air Force Base and are excellent examples of these software intensive systems. Following a brief description of these systems and the instrumentation used, a discussion of the simulation and integration efforts in these programs is presented.

The differences in testing these systems are described in order to convince the program manager, engineer, or tester that in today's world of compressed schedules, decreased budgets, and rising costs, he must alter the more traditional approach to testing.

INTRODUCTION

Two of the most complex tactical navigation and weapon delivery systems to be developed by the U. S. Air Force are in the final stages of their testing at Eglin AFB. These systems are the AN/ARN-101 Digital Modular Avionics System for the F-4E and RF-4C, and the AN/AVQ-26 Pave Tack for the F-111F, F-4E, and RF-4C. Both of these systems have undergone several years of development requiring the rather extensive facilities at Eglin. The history of these programs along with that of the Stores Management Systems (also done at Eglin) will be used as examples in a discussion of tactical navigation and weapon delivery system testing. Emphasis will be placed upon instrumentation and the need for simulation and integration in the development process. In today's world of compressed schedules, decreased budgets, and rising costs, the tester, be he program manager or engineer, must understand the nature of these software intensive systems and the requirement for a testing approach which is unlike anything previously used.

SYSTEM DESCRIPTION

The most complex avionics system test recently conducted at Eglin AFB is the AN/ARN-101. It is comprised of the Inertial Measurement Unit (IMU), and an improved LORAN-C system that interface with a digital control computer. It replaces the present aircraft navigation computer, the inertial navigation set, and the weapon release computer set. In some ways, the ARN-101 is a follow-on to the AN/ARN-92 LORAN Set which was installed in the F-4 during the latter portion of the Southeast Asian conflict. It is, however, considerably more sophisticated than its predecessor. Through a series of Kalman filters, the system integrates aircraft sensors and increases the accuracy of its basic navigation and makes it less than susceptible to degraded modes of operation.

Although the integrated navigation feature represents a significant improvement over the present F-4 navigation system, it is the tie-in with other aircraft systems which provides the greatest advance in capabilities and presents the greatest challenge to the tester.

The digital ARN-101 must interface with the analog F-4 and existing systems such as the Lead Computing Optical Sight, Fire Control Radar, Target Identification System Electro-Optical (TISEO), and Low Altitude Bombing Systems (LABS). The ARN-101 also integrates a complement of weapons to include the LASER/Infrared Maverick, and GBU-15 Modular Guided Glide Bomb. The development task has been even further complicated by the simultaneous development and integration of the AN/AVQ-26 Pave Tack which will be discussed later.

One of the basic concepts behind the development of the ARN-101 was to use existing components whenever possible. The IMU, for example, is the same one that is used in the F-16.

Figures 1-4 show the front and rear cockpits of the F-4E and the extent to which they have been modified by installation of the AN/ARN-101. In the front cockpit, the key changes are: (a) the Flight Director Mode Selector Control which allows the pilot to select the various modes of navigation (i.e., TACAN, VOR/TACAN, ARN-101), (b) the Auxiliary Digital Display Indicator which provides the pilot with a display of ARN-101 parameters selected by the rotary dial on the panel, (c) the Course Select panel containing two toggle switches which select course data and provide cueing during missions, and a rotary switch which selects the weapon parameters from computer memory and provides the information to the aircraft bombing system.

The rear cockpit of the F-4 has undergone the greater number of changes due to the ARN-101 installation. The key modifications include: (a) the Keyer Control which has three rotary switches for selection of steering modes and navigational coordinate systems, as well as a means for data entry and display, (b) the Integrated Hand Control which replaces the standard F-4E antenna control handle and also provides sensor slewing and slaving signals, and discrete signals for the ARN-101/sensor interface, (c) the Navigation Computer Set Control which provides digital modular avionics system (DMAS) on-off control and selection of navigational operating modes, (d) the Target Insertion Controller which is used with the integrated hand control to exchange target and Initial Point (IP) coordinate information between the DMAS and the aircraft sensor, (e) the Sensor Select Panel which provides centralized switching of all the sensors associated with ARN-101 cueing, (f) the Digital Display Indicator which is the primary alphanumeric display unit that presents navigation and dynamic parameters such as altitude and airspeed. In addition to the alphanumeric display, there are five status annunciator lights which give indications of LORAN lock-on and overall DMAS status, and (g) the Data Transfer Module receptacle which is used for insertion of a small cassette used for loading ground programmed missions and retrieving post mission data.

In addition to controlling internal computer computations, the Weapon Systems Officer (WSO) can program up to a combined total of 52 destinations, targets, and avoidance areas. The software program also allows the aircrew to designate two weapon delivery programs from the ballistics stored in the computer.

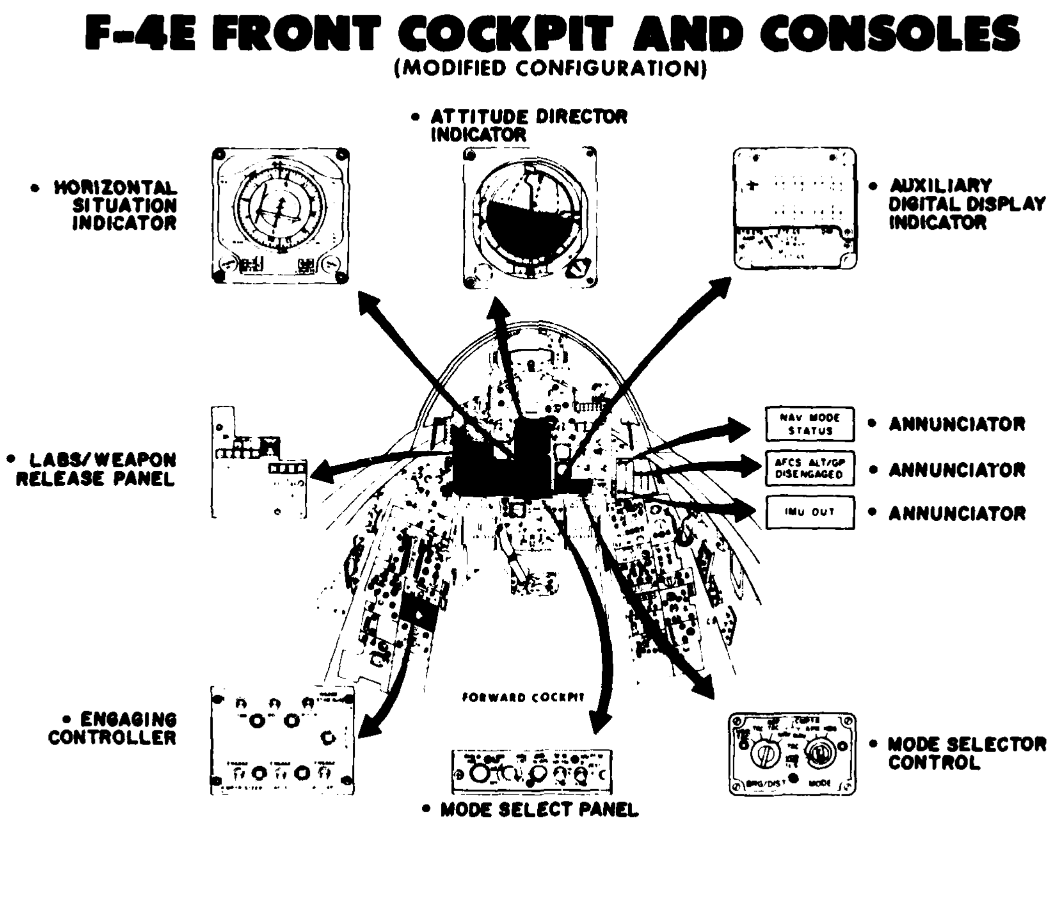


FIGURE 1.

F-4E FRONT COCKPIT WITH AN/ARN-101

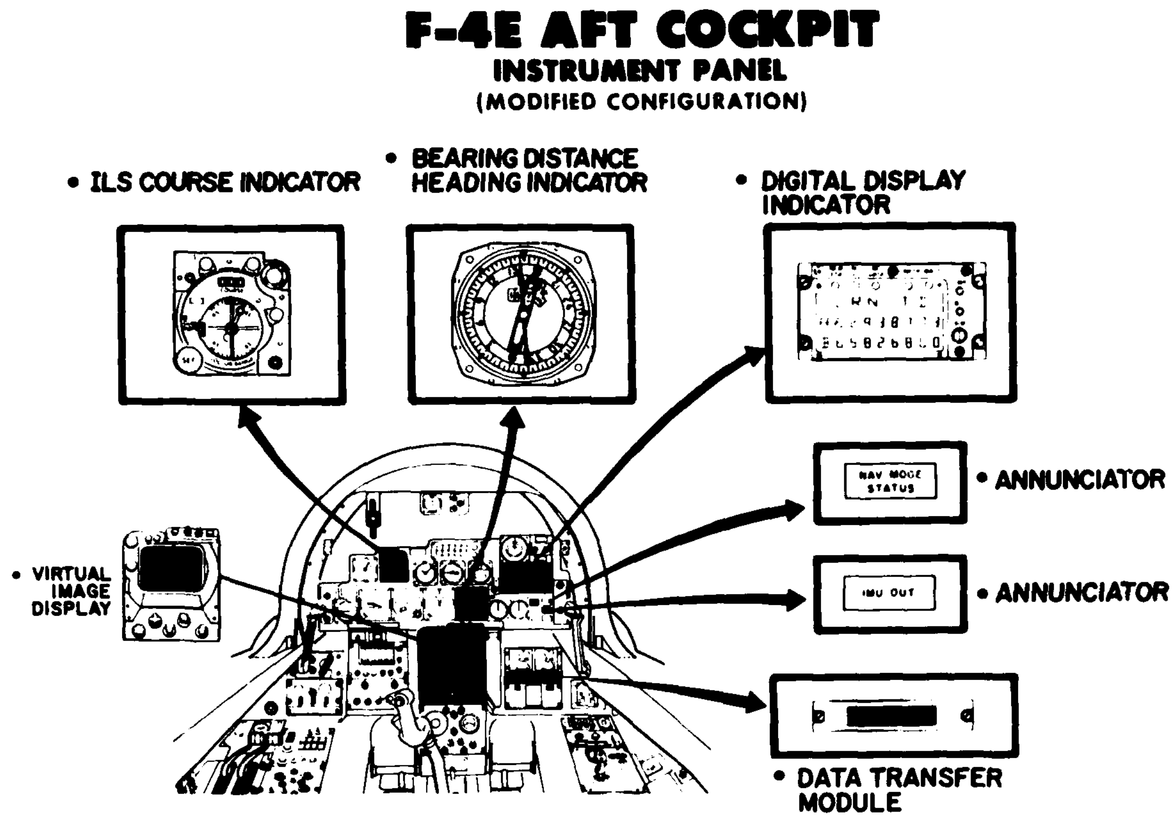


FIGURE 2.

F-4E AFT COCKPIT CONSOLE WITH AN/ARN-101

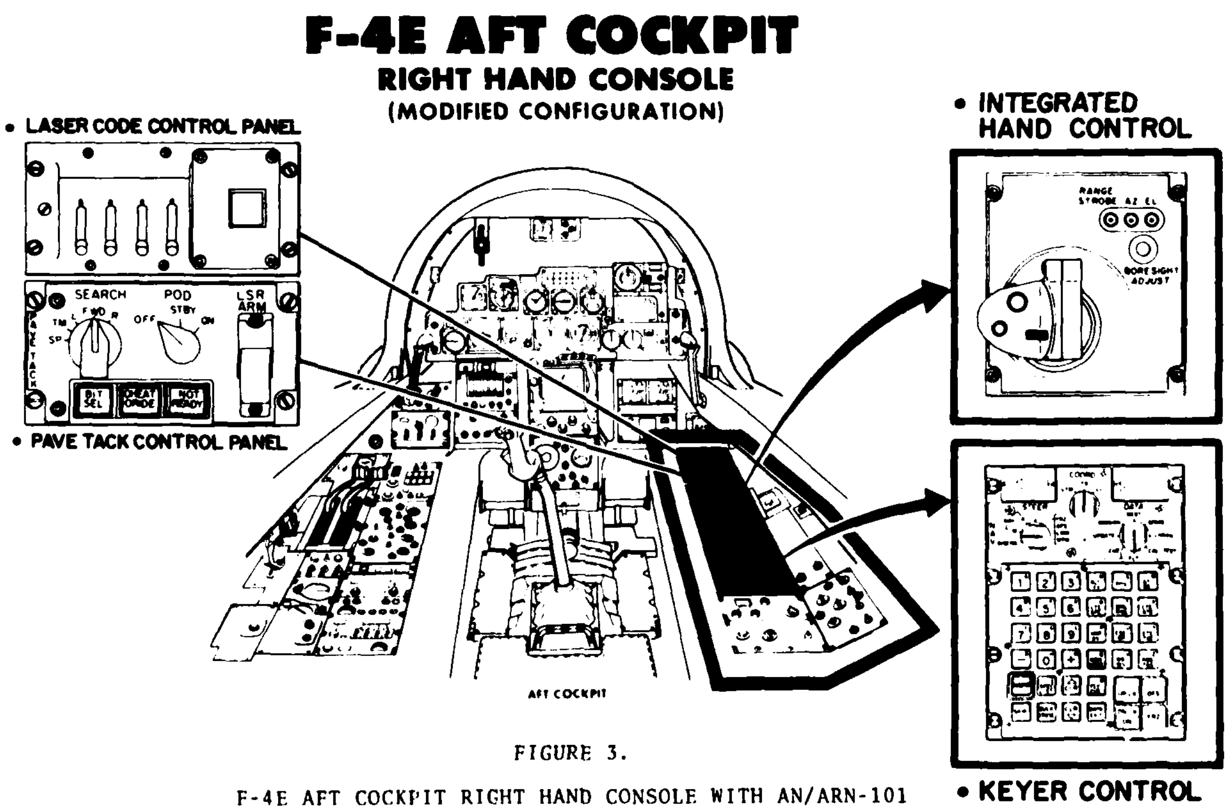


FIGURE 3.

F-4E AFT COCKPIT RIGHT HAND CONSOLE WITH AN/ARN-101

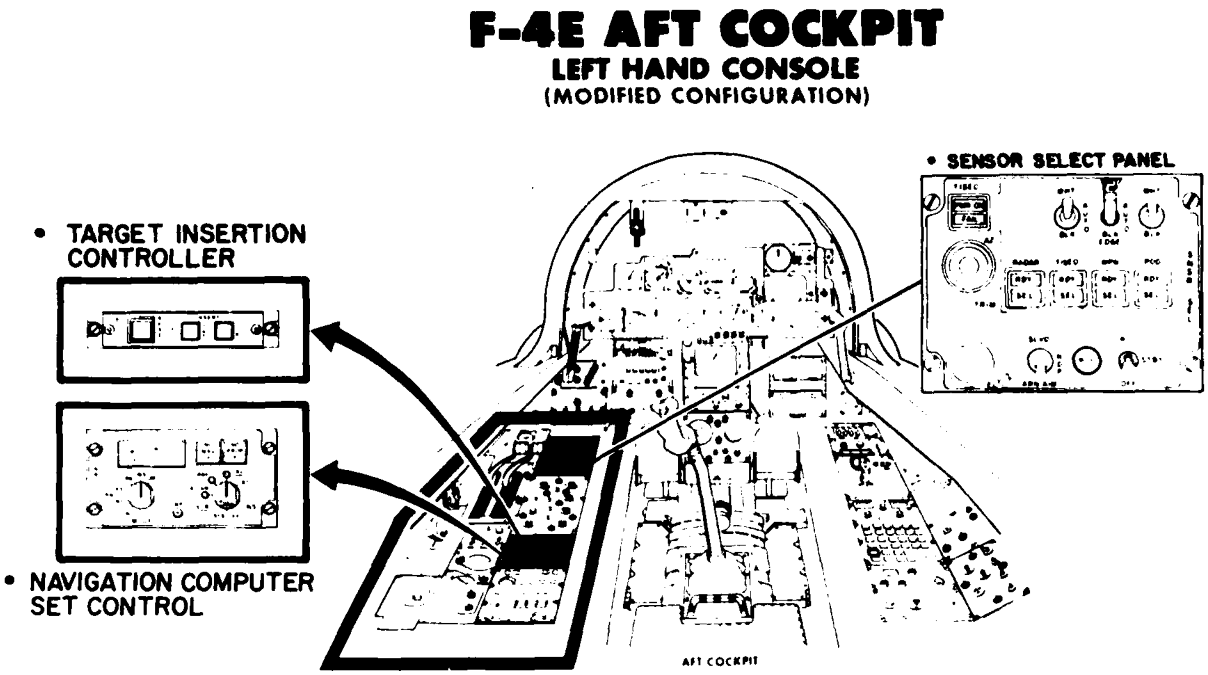


FIGURE 4.

F-4E AFT COCKPIT LEFT HAND CONSOLE WITH AN/ARN-101

The F-4E ARN-101 Weapon Delivery modes are as follows:

a. Blind

b. Dive-Toss

c. Continuously Computed Impact Point

d. Guns

e. Rockets

f. Air-to-Ground Missile

g. Integrated AN/AJB-7 attitude reference and bombing computer

In the RF-4C, the ARN-101 provides the standard navigation function plus the reconnaissance functions of:

a. Mosaic Mapping

b. Route Reconnaissance

c. Point Reconnaissance

The reconnaissance system is also capable of weapon release computations for the Pathfinder mission where the RF-4C leads fighter aircraft to the target and weapons release.

In both the navigation and weapon delivery modes, the system can cue (slave) sensors to the destination or offset points for navigation updates or as secondary targets. The cueing decision tree is very complex which makes the system very powerful, but also makes operation very confusing.

The AN/AVQ-26 Pave Tack is an electro-optical target acquisition, and laser designator system which is carried on the F-4E, RF-4C, and F-111F aircraft. The basic design objective of the system is to provide an increased capability for weapons delivery against preplanned targets when operating at high speeds and low altitudes, both during day and at night.

The Pave Tack System consists of an externally mounted pod, cockpit-mounted controls and displays, and necessary support equipment. The general arrangement of the Pave Tack pod is shown in Figure 5. It is divided into two basic sections: (1) a Base Section Assembly, and (2) a Head Section Assembly.

The Head Section contains an optical bench on which are mounted the Forward Looking Infrared (FLIR), laser designator, laser range receiver, and the stabilized sight. The optical bench is mounted in the turret structure which rotates in pitch and is driven by the turret servo-drive motors.

The turret is mounted in the head structure which rotates in roll with respect to the Base Section. This design permits complete lower hemispheric coverage of the line of sight with respect to the aircraft. The electronic assemblies required to operate the Head Section and to interface with the aircraft are located in the Base Section.

The Pave Tack pod is 4.14 meters in length, 50.8 centimeters in diameter, and weighs 579 kilograms. The pod is carried by an adapter which is hand-mounted to the pod. On the F-4, the adapter is mounted to the standard fuselage-mounted bomb rack. On the F-111, the pod fits into a rotatable cradle in the weapons bay.

The imaging infrared presentation has a 10 shades-of-gray brightness variation with either white hot or black hot polarity selectable from the cockpit. The operator also has available to him cockpit selectable wide and narrow field-of-view plus a double magnification.

The interface with aircraft avionics is accomplished by an internal digital computer and signal generator. The computer is a 16-bit stored program, fixed-point digital computer, with a memory capacity of 16K words with growth potential to 40K words. The signal generator processes the Infrared Detecting Set video output to add alphanumeric annotation. The alphanumerics include three rows of character displays at the top of the display, a pod head pointing reference, and a time-to-go index for weapon release to impact. The three rows at the top of the display show system and/or target status.

The Base Section contains provision for a video tape recorder for the FLIR output. This recorder was used during the initial portion of the flight test but was replaced by a video transmitter for telemetering video to a ground station during the latter portions of the testing.

The rear cockpit of the F-4 and the right side of the F-111F cockpit are modified for the Pave Tack system displays and controls. The cathode ray tube display is enhanced by the use of a lens system which provides an approximately double magnification to the tube image. Line-of-sight control is accomplished by use of a modified hand control with a two-axis pressure sensitive thumb tracker. In the F-4, this is the same hand controller as described earlier.

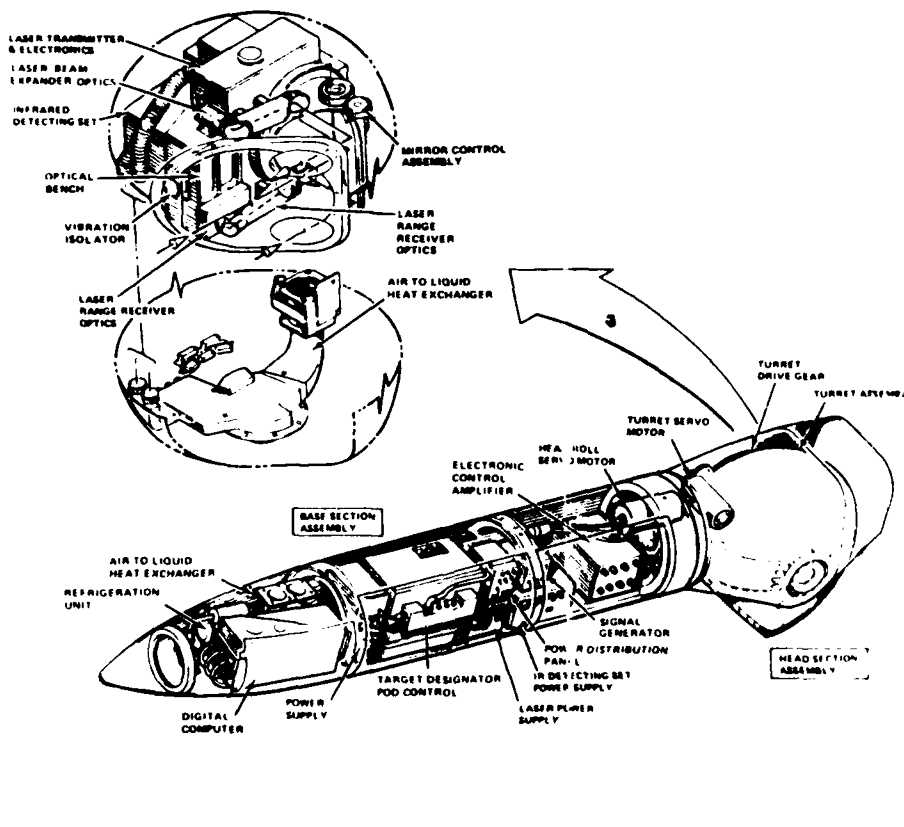


Figure 5. Pave Tack Assembly

Pave Tack has two basic operating modes: search and track. The search modes are intended for use in target location and are controlled by the tracking handle movement. Forward, Right, and Left Acquire are intended for use in acquiring targets of opportunity that are visually located by the aircrew. Snowplow will cause the sightline to be stabilized in space for use in searching roads or other topographical features along which the aircraft may be flown. Terrain Monitor will provide a view of the terrain directly ahead of the aircraft flight vector to aid in terrain avoidance during low level flight. Cue points the sightline at a ground aimpoint designated by the aircraft avionics system.

The track modes are used to accurately track the groundpoint after acquisition in one of the search modes. In Manual Track, the operator manually controls the sightline with the thumb tracker to keep the target centered in the video display. He is assisted in the tracking task by rate aiding signals generated by the Pave Tack computer which improve in accuracy as tracking continues. Memory track can be commanded or is automatically engaged when aircraft maneuvers result in sightline masking by the aircraft fuselage or stores. When the sightline masking no longer exists, the sightline will be pointed at the target location and manual tracking can be reinitiated. Once the target is being tracked, the operator can use the laser range finder/target designator to designate for laser guided munitions and as a highly accurate source of ranging for both guided and unguided munitions. The Pave Tack can also be used for position updating and for damage assessment although this latter capability will probably not be used t[illegible]ch.

Although not yet a part of the production version of Pave Tack, testing is being conducted on a Visually Augmented Tracking System (VATS) which gives Pave Tack the capability to actually lock onto an infrared source, allowing the operator to divert some of his attention to other tasks during the target run.

The Stores Management System (SMS) was a prototype of a digital/multi-plexed, airborne monitor and control system for managing the carriage and delivery of aircraft stores. It was specifically developed to meet the following requirements: Enable changes in the stores list for an aircraft without major modifications of the existing aircraft systems; reduce the need for hardwires from the cockpit switches to the stores stations; provide serial digital data to certain modern weapons; reduce the size, weight, volume, and power requirements of the system; provide self-testing and fault isolation on the ground and in the air; consolidate the pilot controls into a small number of switches and a display panel; provide the pilot with the ability to preplan and program weapon delivery or jettison sequences; and provide the pilot with visibility into his current weapon inventory. The F-4 was used as the testbed, although a similar system could be used on almost any weapon carrying aircraft. This system was designed, built and tested almost entirely within the Armament Division at Eglin. It interfaced with the following principal F-4E aircraft subsystems: Central air data computer, inertial navigation system, weapons release computer set, and the attitude reference bombing computer system.

The SMS was a ROLM 1606 Nova, general purpose, digital mini-computer with 64K, 16-bit words of memory. The SMS was composed of four basic parts: Main logic unit (MLU), multiplex bus, station logic unit (SLU), and a pilot's control panel (PCP).

The MLU processed inputs from the pilot, store stations, and other aircraft avionics systems via the multiplex bus; and outputted the required information to the PCP, the store station, and the aircraft weapon system via the bus. Figure 6 is a pictorial representation of the F-4E/SMS interface.

The multiplex bus was a shielded pair of wires which provided the common communication media for the various components of the SMS. It included the multiplex data bus controller (MBC) in the MLU, and the dual transmission lines. Each remote terminal (RT) on the bus had two sets of transmitters, receivers, sync-coding circuitry, and RT address decoding logic. The RTs listened on both busses. When a valid command synch and RT address was detected, the RT locked onto that bus until the message was complete or a maximum time had elapsed. This allowed the bus controller to transmit on either bus in any sequence.

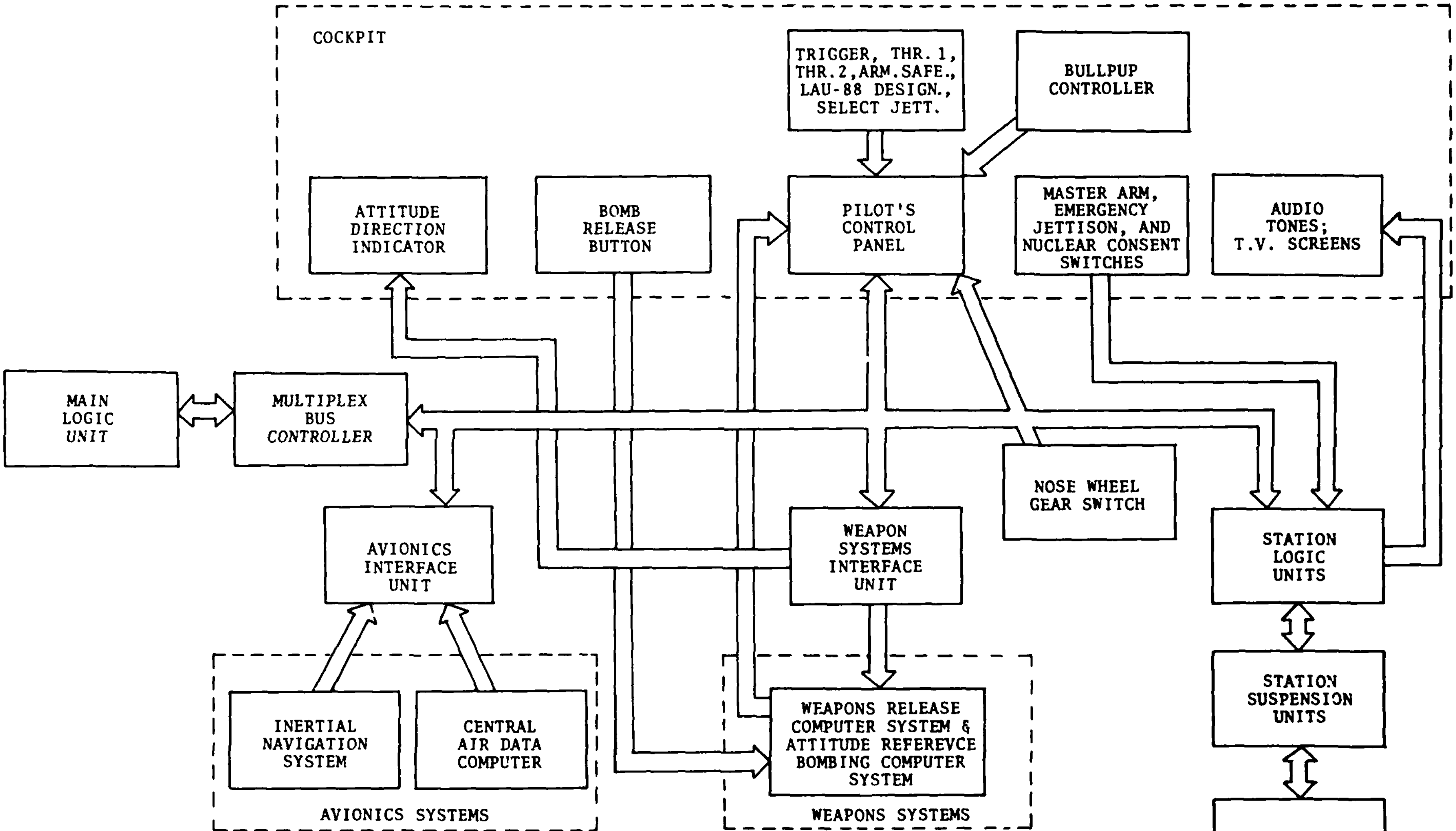
An SLU was located in each of the four weapons pylons. The function of the SLU was two-fold. At each station or pylon, the SLU monitored the status of all suspended weapons. Each SLU also decoded all messages coming to it and generated the appropriate signal or electrical power to the store or back to the MLU. The SLU could control all stores in the F-4E flight manual. In addition, each was designed to control future digital weapons and fuzes.

The PCP was the interface between the system and the pilot. This scope had a programmable alphanumeric display which allowed the pilot to program various release sequence options and call them up at a later time for store release or launch including jettison. It also informed the pilot of possible hazardous conditions if he tried to release stores in an unsafe manner.

INSTRUMENTATION

The testing of systems as complex as those just described required a large number of data sources. The AN/ARN-101 is an example which will be used to examine the problem.

Internal to the AN/ARN-101 itself is the capability to store a number of aircraft parameters automatically at weapons release or selectively at any other time during the



NOTE: ARROWS SHOW ELECTRICAL INTERFACES (LESS POWER CIRCUITS)

FIGURE 6.

SMS Hardware Items

mission. The memory in the system computers also has extra space which is used for test related instructions and data while having little impact on the operational flight program. Additionally, the aircraft alway carried an Eglin designed instrumentation pod. Figure 7 is a block diagram of the pod.

Digital data from the ARN-101 is buffered and formatted in the pod by the Digital Data Interface Unit (DDIU). In addition to the data from the ARN-101, the DDIU also receives inputs from the specially instrumented weapon carriage/ejection racks, and from a discrete bomb release signal which identifies when the pilot actually commanded release. These instrumented racks were necessary because the delay encountered from ARN-101 weapon release signal to actual weapon release turned out to be significant in final impact point determination.

The time code generator is a standard piece of instrumentation in virtually all of the testing done at Eglin. Without this capacity, the merging of data would be impossible.

Outputs from the DDIU are sent to an onboard 14 channel magnetic tape recorder and are also telemetered to a ground station for real time monitoring and control.

The data are transmitted from the ARN-101 to the instrumentation pod in frames of sixteen 16-bit words at a 1 megabit rate. These frames of data are transmitted at an average rate of 20 frames per second, although there is not always 50 msec between frames.

Output from the DDIU to the recorder is in frames of 1000 bits. The data within each frame is arranged in 22 16-bit words numbered 0 to 21 for a total of 352 bits of data.

The instrumentation pod for the ARN-101 project is fairly typical of those used for airborne data gathering. Obviously, the pod gives the instrumentation engineer more space to work with than a system within the aircraft itself.

The data recorded in the instrumentation pod or telemetered to the ground station is processed by the Math Lab located at Eglin. The Math Lab also receives data provided from the Time Space Positioning Indicator (TSPI) Radar, cinetheodolite tracking cameras, laser scoring in the case of Pave Tack, and actual weapons scores. The Math Lab performs the data merging. Figure 8 shows the basic flow of information into the Math Lab. The PDP 15 is used as an input compiler and does not actually process any of the telemetered data. One of the CDC 6600 or Cyber 176 computers is used for the actual data merging and reduction. Using all of this input data, software programs have been written which output a time correlated picture of the AN/ARN-101 sensor pointing angles, the resultant weapon trajectories, and, of course, impact points.

The Pave Tack and SMS programs made use of very similar instrumentation systems. As mentioned earlier, the Pave Tack pod has the capability of on-board recording or data telemetry. The SMS aircraft had a time code generator, an Eglin designed pod which recorded a cockpit video signal and telemetered it to the ground, and the same 14 channel magnetic tape recorder as in the ARN-101 instrumentation pod. The cockpit video was obtained by an over the shoulder in-cockpit camera located on the left canopy rail in the front cockpit. The 14 channel recorder was used to record the information traffic on the multiplex data bus, the time code generator information, and the intercommunications between the pilot and WSO.

Again, as in the ARN-101 system, the SMS testers made extensive use of the Eglin Math Lab. The SMS test pilots also provided the test engineer with a detailed written flight test report after the completion of each mission. This type of report is necessary in any test where aircrew interface is great as in any of the tests discussed in this paper.

It may be fairly obvious that the ARN-101, Pave Tack, and SMS tests had the benefit of modern equipment, but a brief summary of the Eglin facilities actually used is interesting.

The heart of Eglin's mission control is its Centralized Control Facility (CCF) which is a real-time mission control and analysis computer center. The multi-purpose computers which support the CCF have access of most range resources (radars, telemetry, and airborne units). Through specialized computer hardware/software configurations, mission data is processed to provide real-time test analysis and control information. The CCF computers, which include a CDC 6600, Cyber 176, IBM 360, and PDP 15, may be used as stand-alone batch processors or as multi-interfaced, real-time computers.

The two mission analysis and control areas contain two real-time analysis and either two or four real-time control stations equipped with state-of-the-art graphic display system and communications panels.

The use of telemetry in conjunction with the CCF has proven to be extremely useful. Data and functional readouts are available down to the level of switch positions and computer modes. With a test engineer present in the CCF, it is possible to monitor a flight and make changes as circumstances dictate. Many a mission has been "saved" because proper test design allowed the ground-based expert who has spotted a siwtchology error or suggested a successful fix when equipment malfunctioned.

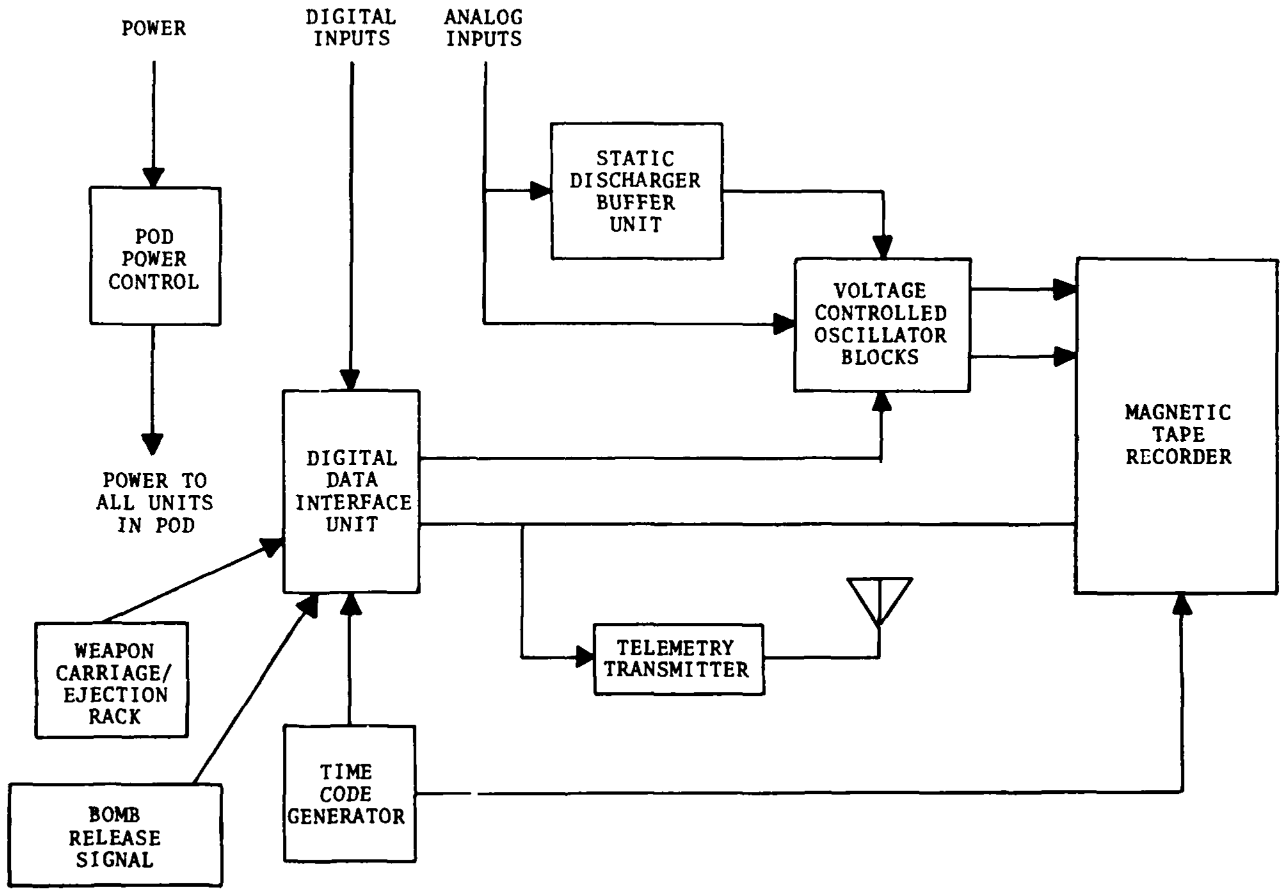


FIGURE 7

ARN-101 Instrumentation Pod Block Diagram

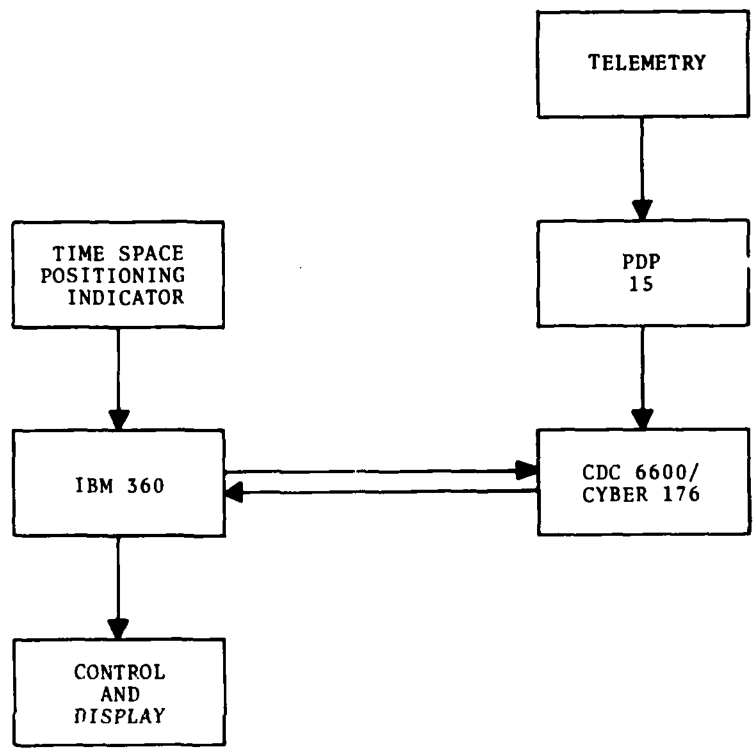


FIGURE 8

Math Lab Information Flow for AN/ARN-101

To support missions, Eglin also has available eight FPS-16 precision tracking radars capable of covering the entire 724 square miles of land range and numerous cinetheodolite cameras located on six land ranges.

INTEGRATION AND SIMULATION

The complexity of the ARN-101, Pave Tack, and SMS is obvious. These are all software intensive systems which are using modern digital computers and techniques to accomplish their design goals. The basic techniques used within the systems themselves do not differ appreciably; however, the manner in which they were developed and integrated onto the F-4 and/or F-111 is quite different. As digital systems become commonplace, the problems of system integration and simulation increase in importance. This is not always an easy lesson for a tester or program manager to learn, because initial costs are often large and the methods are not a natural growth of what was done in the past.

The AN/ARN-101 program was one of the first to attempt this difficult integration task. Many of the problems encountered were not anticipated and the general practices which software designers use today had not yet evolved. As a result, the program suffered many development problems and delays. The F-111 Pave Tack and SMS programs occurred later than the AN/ARN-101 program. The managers and testers had the benefit of learning from their predecessors and were able to structure their programs in anticipation of the problems of aircraft software integration.

The need for complete ground simulation facilities is one of the most important things that today's tester must be aware of. In the past, cost reductions in developmental programs have been attempted. Such facilities, designed to emulate the interfaces between pieces of equipment that will be experienced in actual environments, but under controlled conditions, are very expensive. However, once the decision not to employ this type of facility is made, the test program is committed to a "trial and error" approach to discovering problems, which can be even more expensive. This is particularly true with airborne equipment, where the many environmental variables are very expensive to achieve, not even considering sufficient repetition to give statistical validity to test results.

If a "hardware-in-the-loop" ground simulation facility is developed, it can be used very economically in the verification of software, hardware, aircrew training, and perhaps most important, the integration of new, digital avionics systems with each other and with older analog aircraft systems. The integration of the Stores Management System and of the ARN-101 into the F-4 represent such situations.

The SMS program made excellent use of computer and simulation facilities from its very beginning. One aspect of the SMS program which worked well was that of initially testing most of the software modules in the CDC 6600 prior to being installed in the minicomputer. This was made possible primarily through the use of a higher order language. Programming of the SMS was an area of primary concern to the testers, so they built a cockpit mock-up with the new SMS control panel which was connected to computers that functioned as the aircraft digital interface although this facility was not sophisticated, it was ideal for this test. Numerous software errors were identified, new capabilities programmed and useless ones deleted, and equally important, aircrews trained on a complicated system. This facility no doubt saved a great many valuable flight hours and increased the safety of the flight test program too, as most all missions involved weapon delivery.

Early in the ARN-101 program, it was determined that a "hardware-in-the-loop" simulation facility would not be constructed. As hypothesized previously, this lack of a sophisticated ground test facility committed the test program to an expensive and time consuming method for uncovering problems. The almost infinite number of combinations of the system's input variables and system operating modes made it impossible to evaluate each combination. Each flight could only concentrate on one small portion of the operating matrix in a restricted area of the aircraft's performance spectrum.

The typical result was that problem definition and solutions were slow in coming. When the proposed fix was incorporated into the ARN-101 system software, there was no way (except more flight tests) to verify the effect of that patch on the original problem or its effects on other parts of the system. The ARN-101 system software is very extensive and complex, occupying 48k bytes of memory. Since coding additions, deletions, or changes were made in only that area of the operating system with the known errors, it was possible to determine what effect the local changes had on the overall program. These effects on other parts of the system often turned out to be the most significant. A computer controlled "hardware-in-the-loop" simulation facility could have been used to great advantage here. Verification of software patches could be accomplished both in a local and a global sense. Any desired flight profiles and/or sequences of weapon delivery events could have been programmed into the simulation facility's control computer. The computer could then run through this verification series with engineers and aircrews available to make inputs as necessary. Additionally, a well designed and properly programmed control computer could help isolate and analyze new faults.

This verification series could run in real time, as fast as each of the events being simulated. The results would then be available for analysis by the entire test team. If airborne test missions were required to accomplish the same number of events, the time consumed would be much greater and the cost could be enormous. The cost of saving money is high!

Make no mistake, there is no intent to suggest in any way that actual flight testing can or should be eliminated from a DT&E test program. Exactly the opposite. It must be used to validate the results obtained from any type of ground simulation. But it must be used effectively and efficiently. The most efficient use is to employ flight testing to validate and update the fidelity of computer models and verify hardware/software interfaces and assumptions. The models will gradually then improve in accuracy. With sufficiently good models, flight testing could really be minimized and a program manager could still reasonably attach good confidence to performance predictions for his weapon system in real world environments, even though the number of test flights is limited. Flight testing is becoming more and more expensive every day. It will soon be a limited test resource and must not be squandered.

An additional advantage of constructing an adequate ground simulation facility is that once paid for, it will always be available. Procurement contracts should be structured such that the military service keeps final custody of unique equipment. If the weapon system is procured, the logistics item manager should take responsibility for proper maintenance and use of the simulation equipment. When tactical operator proposed hardware changes and software updates come along (and they always do), the support agency can use the "hardware-in-the-loop" simulation and high fidelity models constructed during full scale engineering development to minimize the time and cost to field changes.

The need for complete simulation facilities goes hand-in-hand with the other concern of today's tester; that of thorough integration of new and existing systems. In the past, as a new or refined weapon delivery or target acquisition system was built, it was simply introduced into an empty space in the aircraft, and testing could begin. This was all the integration that was required. Today, with computer controlled systems that interface with virtually every existing aircraft system, the integration task becomes immense. This fact has not always been recognized or acknowledged. Those initially involved in method of test development are not always aware of the problem, either because of a lack of knowledge or understanding of the new system's capabilities, or due to no experience background in this type of test. Unwillingness to acknowledge the potential magnitude of the integration task can sometimes be traced to cost considerations on the part of the program manager.

The integration of the Pave Tack System on the F-4 and F-111, although not trouble free, represents both the strengths and pitfalls of testing as examined in this paper. The F-4/AN/ARN-101 Pave Tack program is an example of what can happen if early integration is not thorough. During the establishment of the program, a separate contract for integration of the various systems was not let. Therefore, no one agency had responsibility for making the systems work together. By default, the USAF assumed the effort, but was severely hampered by a lack of proper facilities and detailed systems's knowledge. The original integration of the two systems and the aircraft was planned to consist of four flights, which were also to be used for aircrew training. Some six months and 80 flights later, the integration testing was terminated. This termination was due primarily to time and fiscal constraints rather than test objective completion. The only ground facility used in this program was the GRM-99, a LORAN simulator which did not, however, have the capability to tie in with the Pave Tack or other aircraft systems. Unforeseen integration problems included inertial reference and boresight discrepancies, spikes in ranging information which affected weapons delivery solutions lost, and motion in pointing angle computations. Because many of these problems were unresolved at the termination of the first test, it has been necessary to add two more test programs. The AN/ARN-101 completed a six-month tape verification and validation test in May 1979 and an F-4/ARN-101/Pave Tack Interoperability Test which ended in September 1980.

On the other hand, the F-111 Pave Tack program made provisions for an integration effort from its very onset. Central to this effort were integration facilities at the contractor's plant and at the test site. Both of these facilities consisted of cockpit mock-ups which were tied into computers which simulated the aircraft. They were originally built as F-111F simulators designed as flight software program validation and verification facilities. When the Pave Tack development started, these simulators were modified with the new cockpit configurations and Pave Tack inputs to the aircraft. Aircrews who were to be flying the system went to the contractor's plant early in the program when changes in such things as switch positions, displays, and operator/computer interfaces were inexpensive.

At the test site, both the engineers and aircrews made regular use of the Integrated Support Facility (ISF). This facility was also being used by the USAF Logistics Command for operational flight software program work, so it was (and still is) constantly growing both in size and capabilities to the extent that the on-site computers were not large enough to handle the demands upon them. As a result, a tie-in with a remote computer was established.

Pave Tack integration with the ISF was accomplished in two ways. First, the pod flight program was simulated in the ISF software. Second, the actual pod was connected to the ISF. The first method is more difficult to achieve because so many factors must be accounted for in developing a realistic simulation. However, once developed more realistic missions can be "flown" on the simulator than with the second method where the pod has a limited target area and lasing is restricted.

The question as to how useful were the integration facilities is best answered by the test results. The original F-111 Pave Tack test plan called for 20 integration flights, many more than the F-4 effort. However, the integration was completed in 25 sorties, a small growth in the world of testing.

CONCLUSION

As complex, software intensive systems such as the AN/ARN-101, Pave Tack and SMS continue to be developed, both the program manager and engineer must adjust their thinking and approach to testing if they are to be truly successful in development efforts. This is especially true in today's world of inflation, rising costs and smaller budgets.

The AN/ARN-101 program did not develop a sophisticated ground simulation facility. This fact led to an inability to efficiently discover and verify corrections to software faults. This in turn has led to an extended series of test programs, all designed to figure out what was really going on in the system's computer and then modify it as necessary. Alternatively, a good hardware-in-the-loop simulation facility such as that used in the F-111 Pave Tack and SMS programs aided their development processes immensely. The facility should be tailored to the project requirements. As in the SMS, the complexity and cost need not be high.

Of course, all programs are bound to suffer from many mistakes. Within the USAF Systems Command, there is currently a good deal of emphasis on "lessons learned." Much to the credit of those involved, the ARN-101 program seems to have learned some lessons. The contractor now has an overlay program which fits right into the software and provides position, altitude, and velocity in place of the IMU and allows ground testing of various profiles before flight. There is also a capability of adding an aircraft stick for aircrew training.

There is evidence, however, that other programs are not getting the message about the importance of simulation and integration. Hopefully this evidence is wrong. Updating our thinking, no matter where we fit in the systems development cycle is essential to the future success of our efforts.

FLIGHT TESTING AND INSTRUMENTATION

OF AIRCRAFT NAVIGATION SYSTEMS

by
M.J. Taylor, C.Eng., M.R.Ae.S.,
Chief Flight Development Engineer
BRITISH AEROSPACE - AIRCRAFT GROUP
Manchester Division
Chester Road-Woodford-Bramhall-Stockport-Cheshire SK7 1QR
England

ABSTRACT

Reconnaissance aircraft which operate in areas remote from independent ground-based positioning references have special navigation accuracy requirements. For navigation systems designed to meet these requirements, the flight development task also presents special problems.

Experience gained in developing navigation systems for the 'Nimrod' series of aircraft over the past thirteen years has led to the derivation and establishment of test methods which meet the needs of the job.

This paper presents an account of the way in which flight trials on Nimrod Mk.1, Mk.2 and Mk.3 aircraft were formulated to meet the different operational patterns expected in service, the various choices of navigation system modes and the constraints imposed by test facility and air traffic limitations. The instrumentation of the aircraft, its navigation system and the navigation reference system are covered, together with the data analysis methods used.

The philosophy of flight trials is discussed as influenced by all the above factors.

LIST OF ABBREVIATIONS

A&AEE	Aeroplane and Armament Experimental Establishment
ADF	Automatic Direction Finding
AEW	Airborne Early Warning
CTS	Central Tactical System
DF	Direction Finding
DME	Distance Measuring Equipment
ESM	Electronic Support Measures
FIR	Flight Information Region
GMC	Gyro Magnetic Compass
IFF	Identification, Friend or Foe
IN(S)	Inertial Navigation (System)
MR	Maritime Reconnaissance
MSA	Mission System Avionics
NIDAR	Nike Digital Instrumentation Radar
PCM	Pulse Code Modulation
RAE	Royal Aircraft Establishment
SAC	Synchronous Astro Compass

1. INTRODUCTION

The flight testing and development of an aircraft navigation system has elements common to the flight testing of any aircraft or other airborne system.

There is a specification to be met, defining the performance required of the system. There are design features of the system to be demonstrated, which are essential to its functioning but not necessarily conducive to its best performance. The design may contain areas of technical risk, which must be explored. There are the requirements to make the system capable of operation by a competent person in the airborne environment and to attain a satisfactory state of airworthiness. A logical sequence of tests must be devised and carried out to demonstrate satisfactory system performance. The test programme must allow for the possibility of problems needing solution and be carried out in a minimum of time.

In the flight testing of the Nimrod MR Mk.1, which began thirteen years ago, the development of the Navigation System introduced a new field of trials activity which was subsequently evolved through the Nimrod MR Mk.2 and is being applied currently to yet another version of the aircraft; the AEW Mk.3.

This paper describes the trials methods developed and used for 'Nimrod' navigation systems against a background of the constraints applied to test flying in the UK between 1967 and 1980.

2. THE NIMROD AIRCRAFT

Designed originally to meet a Royal Air Force requirement for a Maritime Reconnaissance and anti-submarine aircraft, the Nimrod Mk.1 was based on the well proven Comet airframe. A weapons bay was added beneath the fuselage, accommodation was provided for tactical sensors; avionic equipment was installed for navigation and communications. The aircraft has since been updated with more modern equipment to the Nimrod MR Mk.2, against the same basic requirement and a number are being converted to the Nimrod Mk.3 which is to be the Airborne Early Warning aircraft for the Royal Air Force.

All variants of the Nimrod aircraft perform a reconnaissance function, operating largely over the sea, far from land-based positioning aids. The aircraft are expected to work in conjunction with other defence forces and, in the maritime role, to deliver weapons effectively.

These features lead to the need for Navigation Systems of sufficient accuracy to deploy the aircraft to the required patrol area; when on station, the aircraft sensors must collect data whose precision in geographical position must be fully compatible with the defence network. When directing weapons systems or delivering weapons, the Nimrod navigation system makes an important contribution to the overall accuracy required for a successful strike.

2.1 Nimrod MR Mk.1

This aircraft (Figure 1) entered service with the Royal Air Force in September 1969. The particular navigation system needs for this aircraft amounted to the following:-

(i) Airways Navigation Aids to meet normal transit and terminal requirements.

(ii) An absolute geographical position indication contained within a specified error growth rate from start of flight.

(iii) A tactical position relative to sonobuoys contained, whilst manoeuvring, within a second specified error growth rate.

(iv) Attitude, heading and velocity data for use with tactical sensors.

(v) Ability to be coupled to the Automatic Flight Control System.

To meet these needs, the overall navigation system equipment contained the following items, which are detailed in Table 1.

(a) Airways and terminal aids:- VOR/ILS, TACAN, Gyro Magnetic Compasses, ADF, Periscope Sextant and Air Data Computer. (Omega has been installed recently as an in service fit).

(b) Elliott Inertial Platform with Doppler Velocity slaving, supplemented by LORAN fixing. Back up was provided by Doppler velocity and GMC heading.

Outputs from the above were fed selectively to a Navigation Computer for Routine and Tactical Displays and also for use with sensors, sonobuoys and weapons.

2.2 Nimrod MR Mk.2

Being an update of the Nimrod Mk.1, the aircraft has a similar appearance, (Figure 2), except that wing tip pods have been designed to accommodate new ESM equipment. The basic needs for the Navigation System are similar to those for the Mk.1 aircraft; but the specification is significantly tighter in some ways, namely:-

(i) The airways navigation requirement is being updated to meet the current Transatlantic Standard.

(ii) Improvements to accuracy in both absolute geographical position and tactical positioning.

(iii) Improved reliability.

(iv) Improved data for Tactical Sensors which now cover, for example, classification of Radar targets and ESM contacts.

NOTE: Being maritime targets, these may have low speeds comparable with the potential rate of error growth of the Navigation System.

The equipment to meet these needs, (also shown in Table 1), includes a new IN Platform by Ferranti with standby provided, as before, by Doppler/GMC. The use of the Radar near land is also available as an aid and improved Sonobuoy Homer/DF equipment is fitted. Omega is being introduced subsequent to entry into service.

2.3 Nimrod AEW Mk.3

The Navigation System for this aircraft, Figure 3, relies heavily on the technology and equipment in the Nimrod Mk.2 (See again Table 1).

The requirements are strongly directed at mission reliability and to meet the needs of a comprehensive Mission System Avionics fit, comprising long range radar and secondary radar (IFF), ESM (similar to Mk.2 equipment) and communications capable of transmitting data to describe the defence scene observed from the Nimrod AEW. The flight profile for this aircraft, however, does not contain the tactical manoeuvring necessary to weapon delivery.

3. DEVELOPING THE TRIALS PROGRAMME

The essentials of the Nimrod Mk.1 Navigation System are listed in Table 1. Beginning with this equipment in 1966, it was easy to see that the airways navigation equipments could be tested by traditional means, checking aerial coverage for blanking, using visual 'on top' positioning to check range and bearing from known beacons. Ground swings on GMC and ADF systems would also suffice provided the Compass Base was to Class 1 standard. When it came to the Inertial System however, new ground was being broken. In fact the new system relied on a Doppler/Inertial Mix (Doppler Velocities, used to keep the platform erect, which itself provided IN heading) whose initiation, after static ground erection, depended on starting position information, followed by a runway alignment procedure. When the aircraft was manoeuvred tactically, Doppler attitude limits were likely to be exceeded often and during those times the system would then be purely inertial. This would also be the case if Doppler failed. Figure 4 shows a block diagram of the Doppler/Inertial Navigation System.

It would be necessary to assess overall errors which might arise from initial alignment of equipment, sensitivity of sensors, bias within sensors and noise giving rise to oscillatory and unbounded drift rates.

3.1 Instrumentation

In the integration of the number of equipments from a variety of manufacturers and for which British Aerospace (then Hawker Siddeley Aviation) had the responsibility, a good deal of care had to be taken to identify the contributions of the components to overall system performance. The schedule of parameters for recording, given in Table 2, is ample confirmation of the detailed examination which was undertaken. It is perhaps surprising to look back and see that three separate recording media were used, namely 35mm ciné film (photopanel), 6 inch - 150mm photographic paper (galvonometer traces) and 1 inch - 25mm magnetic tape, recording digital data in PCM format.

The time constants within the system which were thought to be critical in the initiation and detection of system errors, led to the choice of data sampling rates for the ciné and mag. tape records. On the basis of the recording of these parameters, subsequent analysis was expected to reveal behaviour characteristics of the system giving rise to errors of any significance.

The assessment of the installed system performance confronted the trials planners with a special problem. Here was a navigation system providing a near continuous output of position data to a resolution not previously experienced. In order to gain test data of any value to the assessment of navigation accuracy, it was necessary to use a position reference system from which continuous information could also be derived. The accuracy and consistency of this data must be such as not to add significantly to the analysis task. The availability of the reference system all the year round, 24 hours a day and covering the geographic area available for testing, made for an ambitious target.

3.2 Trials Flight Planning

Figure 5 shows a map of the UK air space boundaries and airways network. Flying within foreign FIR boundaries is not permitted for trials purposes and flights through controlled airspace require careful pre-planning and co-ordination with air traffic authorities.

The type of flying envisaged was to be based at Woodford and would be composed of straight transit legs of 42 mins (half Schuler period) duration and of search patterns and tactical manoeuvres representative of those expected to be used by the aircraft in service. Representative altitudes for each flight phase were looked for and some flight patterns were planned to be repeated on a number of different basic headings. In particular, an attempt was to be made to separate North-South and East-West errors in the transit legs.

3.3 The Navigation Reference System

Looking at the area which would be needed for such flights it was evident that quite large blocks of UK airspace would be covered and that whatever reference system was used must extend over the required area. The following possibilities were considered:-

(1) TACAN

(2) DME

(3) Decca

(4) Precision Radar

(5) Photographic Survey (confirmation only)

The availability of the various equipments up to about 1970 is indicated in Figure 6.

The use of TACAN and DME stations as triangulation bases appeared possible, although coverage was not ideal. However, for such a solution a navigation computer would have had to be developed for the reference system. This proposition was not favoured as an accompaniment to the aircraft development programme.

The use of existing Decca Navigator chains appeared attractive, offering a 68% probability of fixing errors within 25 metres on selected paths during full daylight. Diurnal and Seasonal variations detracted somewhat from this performance, but the coverage and the reliability of Decca were very favourable factors. The A&AEE at Boscombe Down in the UK had gone to considerable trouble to make a photographic survey of one particular Decca-based route and had thereby produced evidence of the claimed accuracy. Another route was surveyed using an F49 camera mounted in the prototype Nimrod. This was the Purple Bisector (line of best accuracy) of the English Decca Chain. Corrections to the Decca data were thus available to produce an even finer accuracy if the need arose.

Since these two routes were nearly East-West and North-South respectively, the decision was made to use them as the basis of all transit leg work. The aircraft pilot was to be provided with a Deccometer which simply had to be maintained at a constant reading for 42 minutes. A dedicated operator was to be employed to monitor and log Decca position throughout the relevant parts of the flight. Decca output was interfaced to the magnetic tape recording system appearing as a series of four 18-bit words recurring every second.

In the meantime, discussion with the RAE missile tracking range organisation at Aberporth in West Wales, established that use of one of their FPS 16 radars in specified areas might be fitted in with missile work, to cover the tactical or short term manoeuvre patterns (Figure 7). Using an area behind the main range (to allow other work to continue), the FPS 16 position accuracy was given as about 1 metre per 5 Km range. This accuracy was achievable in areas not subjected to multipath errors introduced by high ground, in a manner indicated by the contour shown in Figure 6. Further, it was necessary to avoid the random errors of skin tracking on a large aircraft and to fit a suitable transponder. It was then necessary in analysis to allow for the distance between this transponder and the IN platform and it was of course vital to attain a high degree of accuracy in the synchronisation of aircraft and radar records. This was achieved by the identification of the trailing edge of a precisely timed tone released on a dedicated V/UHF radio.

The option of recording over-land 'fixes' photographically was retained, using an F49 vertical camera in lieu of aircraft operational equipment.

The flight plans which emerged were based on the surveyed Decca routes and on the Aberporth range, in this case using the normal Decca area coverage as a lead in. Flying was normally to take place in daylight hours and was thus more constrained in the winter period. For a programme expected to last some months, this was an undesirable yet unavoidable situation.

3.4 Aircraft and Runway Alignment

The final detail of the navigation reference arrangements was the determination of aircraft true heading before and after a trials flight, and the recording of aircraft mean heading during the runway alignment phase. The first essential was the harmonisation of navigation sensors with an aircraft centre line reference. This was achieved in final build stages by transferring an IN platform between the various mountings. Determination of true heading was made by a simple theodolite base at Woodford referenced to accurately surveyed landmarks. Mean heading on the runway was measured firstly by marking the static position of the aircraft lined up for take-off. Its position at the end of the runway alignment phase (about 600m distance gone) was then recorded by a mark left on the runway when a small paint capsule was fired from a gun mounted vertically in the aircraft. Simple measurements then produced the required information to better than 0.01° accuracy.

3.5 Trials Support

The overall avionics development programme for the Nimrod Mk.1 had provisioned for a system rig similar in physical layout, power supplies, inter rack wiring and equipment cooling to the real aircraft. As well as working up the equipments, sub-systems and software before flying began, this rig was invaluable in the support of the flying programme. It enabled hardware and software faults to be simulated and allowed development to take place away from the aircraft. In particular, special nav. computer software overlays, for trials purposes, were validated on the rig.

After a normal acceptance procedure, spare components were also operated on the system rig before being fitted to the aircraft.

3.6 Analysis

A block diagram of the data flows for analysis of flight records is shown in Figure 8. After a flight involving the Aberporth radar, the radar record would be recovered by road to the main analysis site. For the Nimrod Mk.1 trials this radar record had to be digested by a mainframe computer before merging with the flight data. For Decca based runs, the same basic analysis arrangement was used without the radar input of course. In each case the first line analysis would be the breakdown of Navigation position errors into Northerly and Easterly components, also along track and across track errors. After this process had been completed, the more laborious analysis of trace records would be undertaken selectively in order to relate component behaviours to the particular flight path, system mode, and environmental conditions experienced. This same procedure was used for all the available Navigation System modes (ie. Doppler/IN, Pure IN, Doppler/GMC, Air Data/GMC) and it was the intention to build a mathematical model, particularly from the IN based results.

3.7 Programme

Two comprehensively instrumented aircraft were allocated for navigation trials, contained within a programme including other trials, especially on the closely allied tactical systems. One of the aircraft was a systems prototype but the other was an instrumented production aircraft with development equipment on board. Overall flying rate was planned at 20 hours per aircraft per month into which all development changes, routine servicing and maintenance were fitted. On this basis the contractor's navigation development trials were completed in a period of two and a half years, using about 330 flying hours shared between the two aircraft.

3.8 Trials Experience and Results

The early period of prototype flying was given over to shakedown of systems. In common with experience elsewhere, it was found that electronic systems using low voltage signals passing through a multitude of wires and miniature connectors were not without their problems. The normal rigours of the flight environment were enough to reveal snags not uncovered by exhaustive ground checks. Painstaking efforts were needed to achieve reliability and produce a fully serviceable navigation system.

The flying of the various navigation patterns was successfully negotiated. Good co-operation by civil and military air traffic controllers enabled the specialised demands of the Nimrod flights through controlled airspace to be met with minimal disturbance. Careful pre-planning of the use of the Aberporth range likewise yielded successful results, although missile firing sometimes delayed Nimrod sorties and winds over 60 km/hr were sufficient to deflect the tracking antenna mountings and render radar data unreliable. Sorties were never abandoned due to radar or Decca transmitter failures.

The gradual accumulation of reference data and experience of operating the navigation sorties enabled reliable figures to be built up on detectable errors in the navigation system. Position errors could <u>always</u> be detected to an accuracy better than 30 metres, a measurement comparable with the size of the aircraft. Rate of error growth could <u>always</u> be measured accurate to about 0.6 km/hr. These figures were, of course, improved on close to the radar range head. These data were gathered during performance and fragmented navigation patterns described in Paragraph 3.3.

From such information it was intended to build a mathematical model of the navigation system but this was found difficult to achieve at the time of the trials, partly because instrumentation coverage was not ideal. Modelling of the Nimrod Mk.1 navigation system was subsequently successful as reported in Reference 1. The behaviour of the inertial system was ultimately found to be dependent upon the entire history of a flight from 'brakes-off' to the current position. However, work had been done to establish statistical evidence of the performance of a number of IN platforms on which the primary navigation mode of the Nimrod Mk.1 relied. This included ground rig work on drift rates. Subsequent production series equipment could then be shown to produce part of the same population of results.

The passing off of production aircraft systems was reduced to a straightforward routine. A 'bolt-on' Decca set was mounted in the aircraft coupled to an ADF/Loran aerial.

The IN was subjected to drift runs both on the systems rig and in the aircraft to trim the platform. Four consecutive straight transit runs were made on the 'Purple Bisector' of the English Decca chain. Simple data recovery from the aircraft's Mission Analysis Recorder, (programmed to accept Decca on test flights) produced error plots on the same day as the flight. Evidence of the consistency of some production results is shown in Figure 9.

4. NIMROD MK.2

By the time deliveries of Nimrod Mk.1 to the Royal Air Force were complete, updated tactical sensors were already under serious consideration and it was realised that a more accurate and reliable Navigation System would be needed to complement the new avionics fit. The Ferranti FIN 1012 Inertial Platform was chosen for the primary navigation mode, reversionary modes being as for Nimrod Mk.1. Loran was retained as a fixing and updating aid when in good groundwave cover. Information from all the Navigation sensors was to be input to a new Central Tactical System providing revised Routine Navigation and Tactical displays, based on the best navigation information available.

The Ferranti platform is self-aligning, requiring no runway alignment phase. Confidence in the long term accuracy of this platform had been gained during development by Ferranti, (Reference 2) and subsequently in evaluation trials by the Royal Aircraft Establishment at Farnborough.

It was arranged that, prior to British Aerospace trials, each platform used for assessment of navigation performance would have a declaration of performance. This was to be based on data gathered from flights by the RAE with small batches of development platforms. (This process of platform qualification was continued on production platforms by the A&AEE at Boscombe Down).

4.1 Trials Objectives

After experience with the Mk.1 aircraft, the purpose of the Mk.2 trials could be somewhat refined.

In demonstrating Navigation performance, statistical evidence could also be shown that this performance was not degraded from that declared for the platforms in the RAE tests.

The navigation performance in reversionary modes with special reference to the modified GM7 Compass would also be demonstrated.

Integration of the Navigation Sensors with the Central Tactical System and hence with the Tactical Sensors would be proved. Equipment conditioning and compatibility trials would be made to ensure that the overall installation was satisfactory.

One of the significant purposes was to gather sufficient data to validate a mathematical model of the Navigation System on which comprehensive performance characteristics could be determined. Assessments of ergonomic design, operating and servicing procedures and built in test functions were also to be made.

4.2 Navigation References and Trials Planning

Based on the previous experience and results, the aim for the Mk.2 was to plan trials sorties which were far more nearly representative of operational flight patterns.

This meant longer sorties with less constraints on tactical manoeuvres than had been accepted for the Nimrod Mk.1. With Decca coverage of the UK now improved, (Figure 6) its use on an area basis was again planned, with Bisector runs for special purposes. The Aberporth FPS 16 radar accuracy was good enough for tactical work but availability and range boundaries were restrictive so the Army radar range at Benbecula off the West coast of Scotland was investigated, (See Figure 6). This range had been re-equipped with NIDAR, a radar of comparable performance with the FPS 16. Despite considerable advance planning efforts, a way could not be found to use this range effectively, partly through inadequate calibration and partly through the difficulty in managing priorities. Finally therefore a series of long sortie plans, mixing transits, search patterns and tactical manoeuvres was decided on, with the later manoeuvres contained within the Aberporth range boundaries.

4.3 Aircraft Alignment

Harmonisation of the new navigation sensor equipment with an aircraft centreline reference was effected for the Nimrod Mk.2 by the use of a Wild ARK-1 gyro theodolite. Mounting adaptors at Doppler and IN positions were specially made for this purpose and reference was made to the aircraft compass sighting rods for datum purposes. External aircraft markings were used for pre and post-flight heading measurements as with Nimrod Mk.1 but using the Wild gyro theodolite.

4.4 Trials Support

Probably the greatest advance in support between the two versions of the aircraft, was made in the Systems Integration rig. Constructed to represent the Tactical compartment of the aircraft, it contained a Navigation System and Central Tactical System. Air Data Computer, Radio Altimeter and Loran were not fitted but interfaces between these equipments and CTS were representative, allowing realistic signal inputs to be used.

An INS simulator was incorporated to create 'live' navigation data. Simulation of GM7 and Doppler inputs was also developed to match the simulated IN movements. Sub-system and system integration tests were performed on the rig, exercising components over their full input-output ranges, unlikely to be achieved in flight. Validation of flight software, including trials overlays was also completed.

As before, flight instrumentation was incorporated in the rig, but here the more comprehensive avionic instrumentation modules could be fully represented. As before analysis procedures were run and developed before flight trials began.

4.5 Instrumentation and Analysis

With regard to recorded parameters, the basic philosophy was unchanged from Nimrod Mk.1. However, even greater coverage of sub-system parameters at greater recording frequency was adopted as a result of Nimrod Mk.1 experience and the variety of recording media was not worthy of repetition. Digital magnetic tape recording of all flight data was adopted with a more efficient data processing system based on the PDP 11/45 computer. Merging of radar range and flight data was carried out at the one facility and second line analysis was handled on a separate PDP 11/40 machine. First line analysis could be completed within 6 hours of the reception of the radar record.

4.6 Programme

Again, two fully instrumented aircraft were allocated to navigation trials. One was an early prototype aircraft representative only in the fit of navigation equipment. The second aircraft was built to the full Mk.2 conversion standard and was capable of longer flight durations. This feature was of importance because the improved navigation system did not reveal easily detectable errors until the flight was well under way. The decision to use the Aberporth radar range to a greater extent on each sortie was clearly correct in that the most important phase of navigation error detection was flown against the more accurate reference. Greater conflict with missile work on the range was inevitable. However, the navigation development flying was completed in conjunction with other tasks over a period of 14 months, using 240 hours of flying time.

Production conversion of Nimrod Mk.2 is in the early stages, but it is already evident that the reliability of the navigation system is aiding more rapid clearance. A long sortie is a feature of this procedure when navigation and tactical sensor checks are interleaved to mutual advantage. Only airways type navigation aids are needed for navigation reference. The long sorties are terminated with ground examination of residual platform drift.

5. NIMROD MK.3

This aircraft uses the same primary navigation sensor as the Nimrod Mk.2. The Ferranti FIN 1012 is duplicated for overall Mission reliability and the live platform is used as attitude and heading reference for the Mission System Avionics. Having a less vigorous flight profile to endure the overall Navigation System errors are likely to be less than those measured on the Maritime reconnaissance aircraft.

The Mk.3 trials have just begun. It is confidently predicted that a considerably reduced flight time for navigation system development will be required, even though its interface with a new Mission System Avionics fit is involved.

The principles evolved through Nimrod Mk.1 and Mk.2 trials are being refined and adopted once more:-

(i) A Systems Integration rig, including MSA is being commissioned. IN and compatible navigation system simulators are in use. Representative instrumentation is fitted.

(ii) Trials software for the on-board navigation computer has been developed on the Systems Integration rig and is in use on the first development aircraft.

(iii) Flight sorties will follow representative operational profiles. Precision radar will be used to cover as much as possible of these sorties. Decca is being used for simple drift assessment and lead in to patrol pattern flying.

Navigation trials on the Mk.3 are scheduled on two fully instrumented aircraft and, interleaved with other development, trials are expected to occupy about 100 flying hours of flying time.

6. CONCLUDING REMARKS

The flight development hours flown by British Aerospace on a series of Nimrod navigation systems shows a diminishing total with each successive system (Figure 10). At the same time, it must be recognised that considerable research and development effort was made by the manufacturers of individual sensors and by government establishments. Also the support work on the ground has progressivley increased in sophistication. It is now pertinent to consider what the future requirements will be for flight testing navigation systems.

Already, brief experience has been gained on a Nimrod flight with an IN platform of significantly lower drift rate than any of the equipment dealt with so far. Also some evaluation has been made of Omega accuracy. Using the current standard of navigation reference, it now becomes necessary to fly longer sorties before reliable measurements of drift error can be made. For non time dependent systems, flying closer to the radar range head becomes necessary, thus restricting the available trials area.

It seems likely that military aircraft will continue to exploit the most accurate navigation position and heading information that can be made available. This information will continue to be needed both for long term geographical position and short term weapon-aiming purposes. We may therefore expect further development of the independent IN based systems, possible integration with superior position fixing aids and the power of the on-board computer to be utilised to enhance the value of whatever sensors are selected.

Assuming that future systems live up to their promise, it becomes increasingly difficult to find a reference against which to declare a navigation performance. In the UK the provision of an enhanced precision radar facility may fulfil the needs of the immediate future. The linking of the Aberporth and Benbecula ranges could be a useful step in this direction.

It must be admitted however that in face of sustained improvement in navigation systems, it will be a struggle to keep continuously recording references anything like an order more accurate than the system under test.

It is quite possible that we may find that successful development in this field lies in a far greater effort on the ground, accompanied by a relative reduction of flight trials effort. The system development would be made through more sophisticated software designs and mathematical modelling. The flight trials would reduce to a few precision measurements of position and heading allied to carefully controlled flight profiles. It is the author's belief that this will be the way ahead.

7. REFERENCES

1. M. Attwood — Hawker Siddeley Aviation
   Validation of an error model of an aircraft navigation system
   1976
   Society for Computer Simulation
   Proceedings Volume 6 No. 2

2. W.H. McKinlay — Ferranti Limited
   Inertial Navigation and its impact on future avionic systems
   1974
   Avionics To-day and Tomorrow SERT/SLAET Swansea 1974
   Pages 147 - 163

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NIMROD MR MK.1	AIRWAYS AIDS: Twin Sperry GM 7 Compasses; Marconi AD 360 ADF; Hoffmann AN-ARN 72 TACAN; Twin Marconi AD 260 VOR/ILS Elliott 81-01-26 Air Data Computer (Litton LTN 211 Omega Retrofit) TACTICAL NAVIGATION: Elliott E3 Inertial Navigation System Decca 67M Doppler Radar Decca ADL 21 LORAN CDC/Kollsman Synchronous Astro Compass NAV/TAC COMPUTING: CDC Automatic Wind Computer CDC Sea Motion Corrector CDC Spherical Data Computer CDC Ground Speed Coupler CDC Ground Speed Resolver Elliott 920B Computer
NIMROD MR MK.2	AIRWAYS AIDS: Twin Sperry GM 7 Compasses; Marconi AD 360 ADF; Hoffmann AN-ARN 72 TACAN; Twin Marconi AD 260 VOR/ILS Elliott 81-01-26 Air Data Computer (Litton LTN 211 Omega Retrofit) TACTICAL NAVIGATION: Ferranti FIN 1012 Inertial Navigation System Decca 67M Doppler Radar Decca ADL 21 LORAN NAV/TAC COMPUTING: CDC Spherical Data Computer Marconi Avionics 920 ATC Computer
NIMROD AEW MK.3	AIRWAYS AIDS: Single Sperry GM 7 Compass; Marconi AD 360 ADF; Marconi Avionics AD 2770 TACAN; Twin Marconi AD 260 VOR/ILS Elliott 81-01-26 Air Data Computer TACTICAL NAVIGATION: Twin Ferranti FIN 1012 Inertial Navigation Systems Decca ADL 21 LORAN NAV COMPUTING: CDC Spherical Data Computer Marconi Avionics 920 ATC Computer

TABLE 1

NIMROD NAVIGATION SYSTEMS - MAJOR COMPONENTS

	PHOTOPANEL RECORDER	PAPER TRACE (GALVO) RECORDER	MAGNETIC TAPE RECORDER
HEADING SYSTEM	Primary heading changeover to Ground Speed Resolver. Primary true heading to computer from IN. Secondary magnetic heading Nos. 1 and 2 GMC. True heading output from No.2 true synchro amplifier. True heading output from heading repeater. SAC corrections to IN and GMC systems. Input to variation unit. Secondary system comparator operations No.1/No.2 and Primary/No.2.	Primary/secondary heading selected. Nav mode selected. Destination or track selected. Analogue computation in memory. Ground speed correction for sea motion. Drift correction for sea motion.	Primary true heading. Secondary true heading. True track for 920B computer. Primary/secondary heading selection. Destination or 'track set' mode. Selected destination (lat/long) when 'steer' is selected. Distance to destination. Across track error. Accumulated meridian convergence change since last computation of required track.
IN ALIGNMENT	Selection of runway alignment. Runway orientation setting coarse and fine. Selection of Read heading/Read track, Track angle IN Platform Heading.	Inertial N/S and E/W velocities. Inertial N/S and E/W velocities to Ground Speed Resolver. Steer button. Selection of runway alignment.	Inertial N/S and E/W velocities.
DOPPLER/IN MIXING	IN mode selection and failure warning. IN roll and pitch attitude. Doppler attitude limits. Doppler signal/noise trip. Doppler error trip and interlock.	Doppler/IN mix signals. N/S and E/W. Doppler ground speed slaving signal from Auto Wind Computer. IN platform and Doppler failures. Auto Wind Computer failure land/sea selection. Sea Motion Correction On/Off.	
AIR DATA COMPUTER	Outside air temperature. Altitude output.	Windspeed computed or set in air data mode. TAS input to Automatic Wind Computer from Air Data Computer. TAS input to 920B Computer from Air Data Computer Interface.	True Airspeed (TAS).
DEAD RECKONING COMPUTATION	Sea Motion Corrector On/Off. Automatic Wind Computer fail. Automatic Wind Computer mode. IN good. Doppler good. Corrected drift angle from Automatic Wind Computer. Computed wind direction. DR lat and long to 920B Computer. Actual true track to nav display. N/S and E/W ground mileage increment outputs from Ground Speed Resolver.	Unresolved ground speed from Ground Speed Resolver to Automatic Wind Computer. Error in Automatic Wind Computer drift relative to Ground Speed Resolver drift. Ground speed output from Automatic Wind Computer to 920B Computer. Unresolved ground speed at Ground Speed Resolver. Input from Ground Speed Coupler. Automatic Wind Computer failure. Sea Motion corrections to Ground Speed and Drift. Variation input to 920B Interface Unit. Corrected drift angle from Automatic Wind Computer to 920B Computer.	Corrected drift angle from Automatic Wind Computer to 920B Computer. Ground speed output from Automatic Wind Computer to 920B Computer. Time of re-synch analogue and digitally computed position. Lat and long 920B Computer input and digital output. Distance to destination. Command track.
DECCA	Chain selection. Lane ident. Decomater, red, green and purple lanes. Decca ignore. Zone accumulation meter.		Actual aircraft position in Decca lattice or co-ordinates.

TABLE 2 - NIMROD MK.1 SUMMARY OF NAVIGATION SYSTEM INSTRUMENTATION PARAMETERS

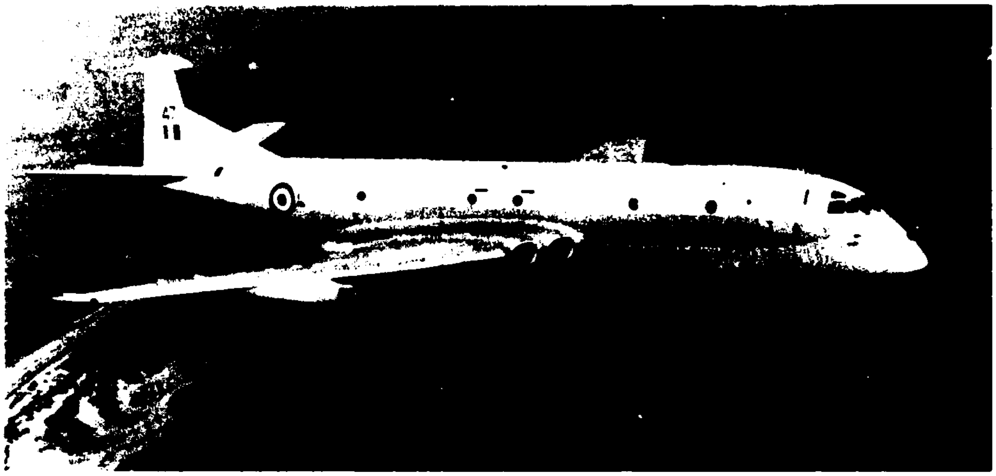


FIGURE 1 - NIMROD MR MK.1

FIGURE 2 - NIMROD MR MK.2

FIGURE 3 - NIMROD AEW MK.3

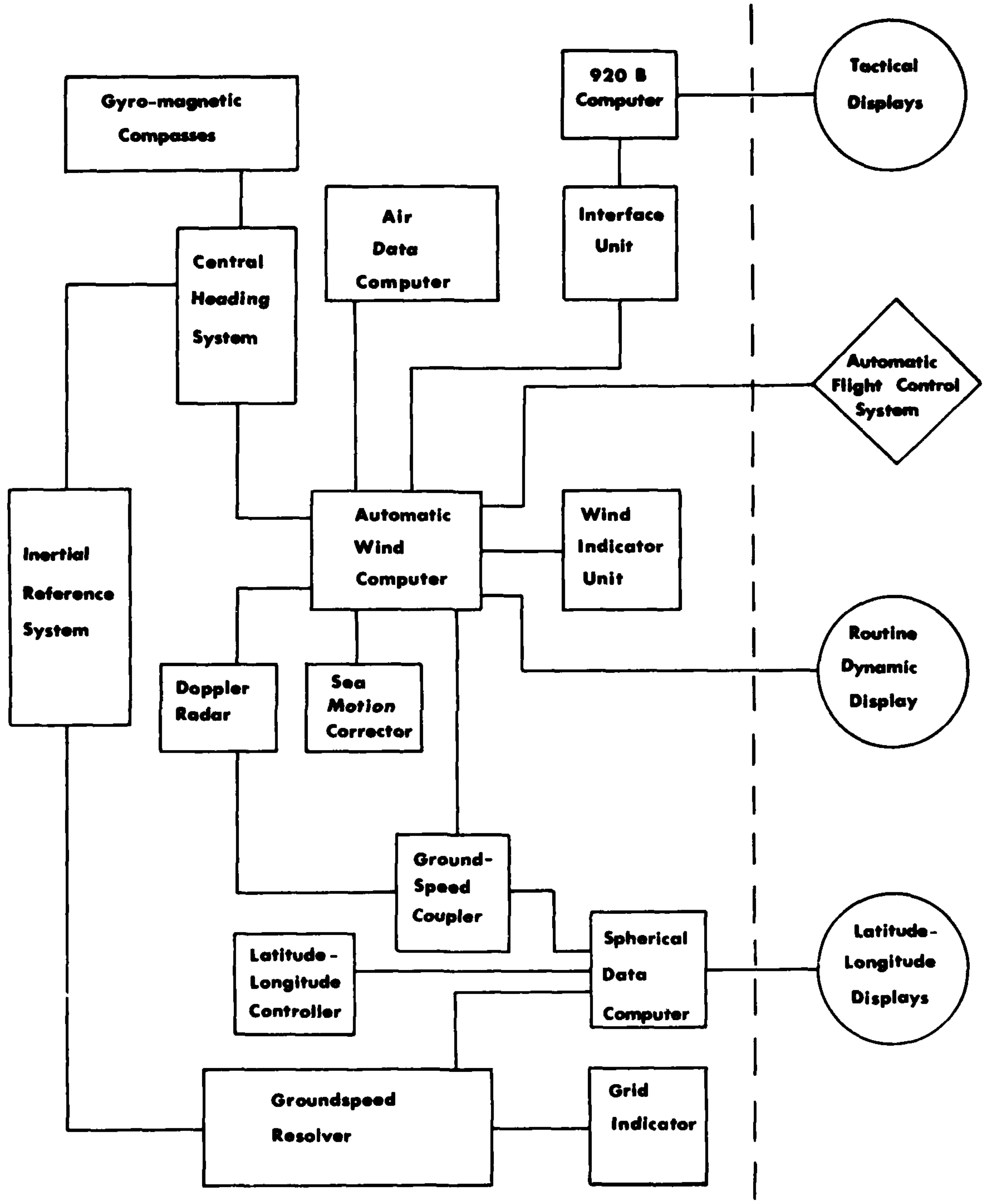


FIGURE 4

NIMROD MR MK.1
NAVIGATION SYSTEM
ESSENTIALS OF DOPPLER/INERTIAL MIX MODE

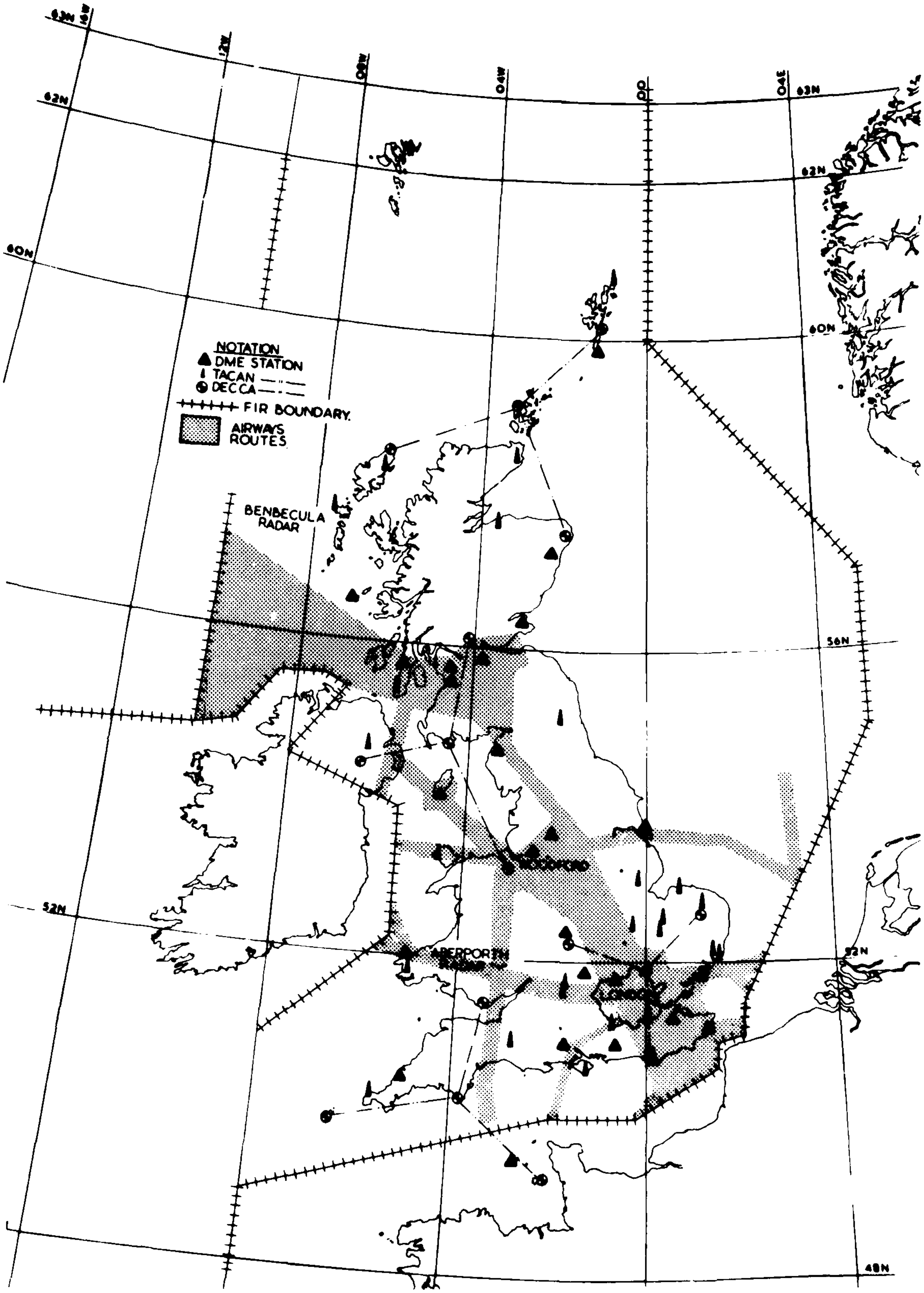


FıGURE 5

CONTROLLED AIRSPACE AND FIR BOUNDARIES AROUND THE UK
SHOWING POTENTIAL NAVIGATION REFERENCES

RANDOM FIXING ERROR (KM) - 68% PROBABILITY

Decca Period (see Time-Season Diagram)	Error in kilometres Contour C	D	E	F
Full Daylight	<0·19	0·19	0·24	0·46
Half Light	<0·19	0·24	0·46	0·93
Dusk	0·24	0·46	0·93	1·85
Summer Night	0·46	0·93	1·85	3·70
Winter Night	0·69	1·39	2·78	5·56

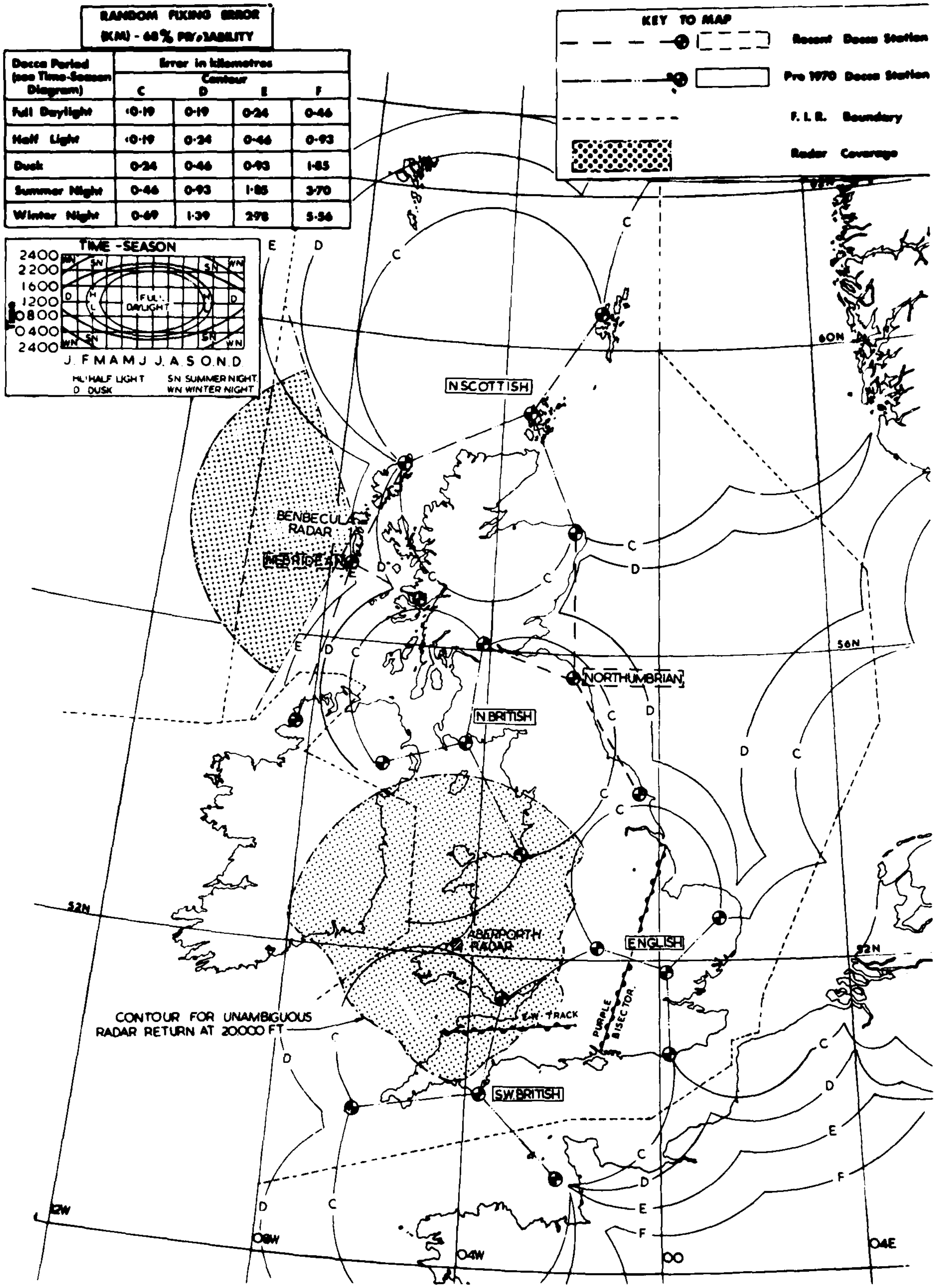


FIGURE 6

DECCA AND PRECISION RADAR COVERAGE OF UK AIRSPACE
USED FOR NIMROD NAVIGATION TRIALS

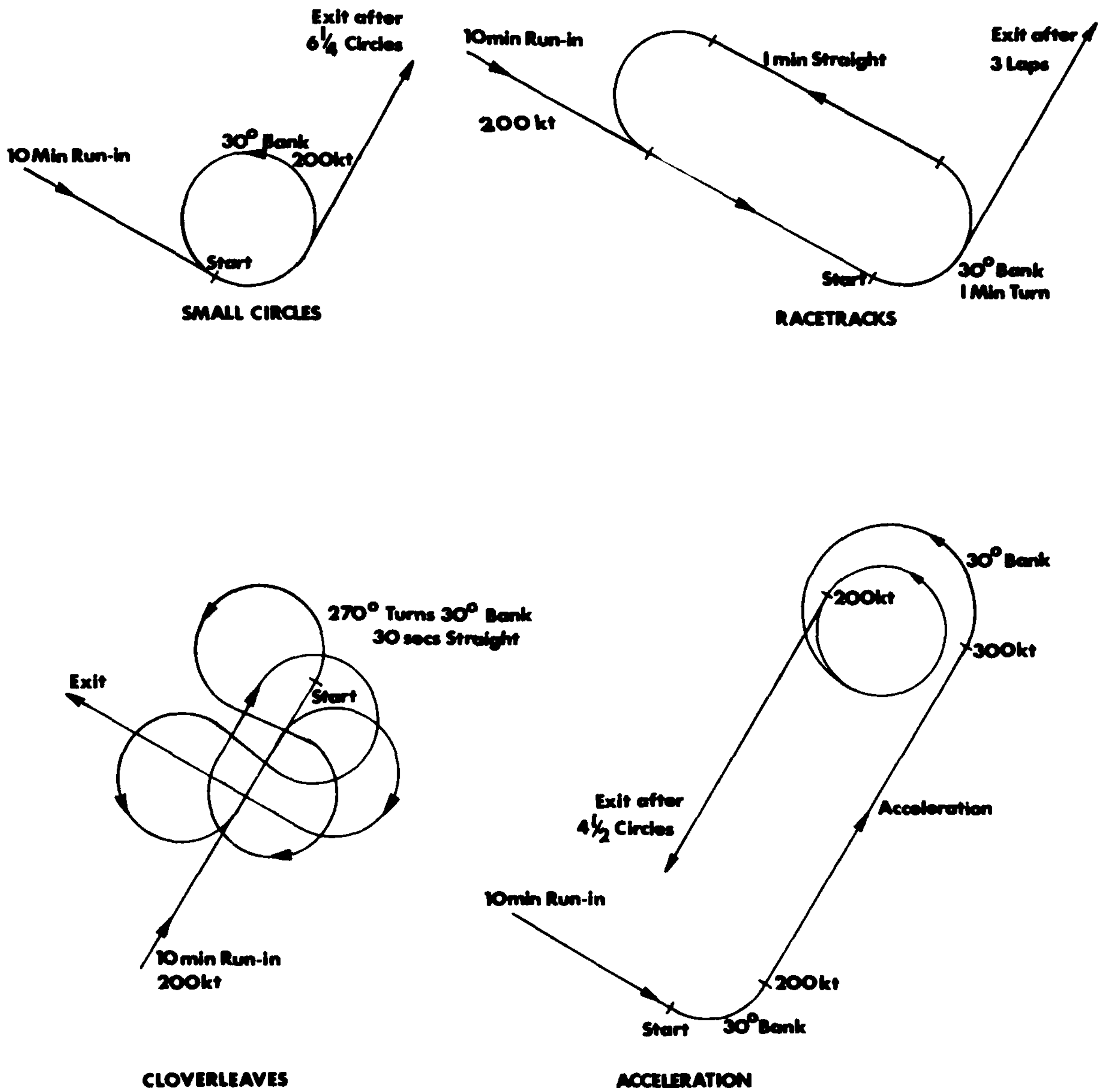


FIGURE 7

NIMROD MR MK.1
SIMULATED TACTICAL MANOEUVRES

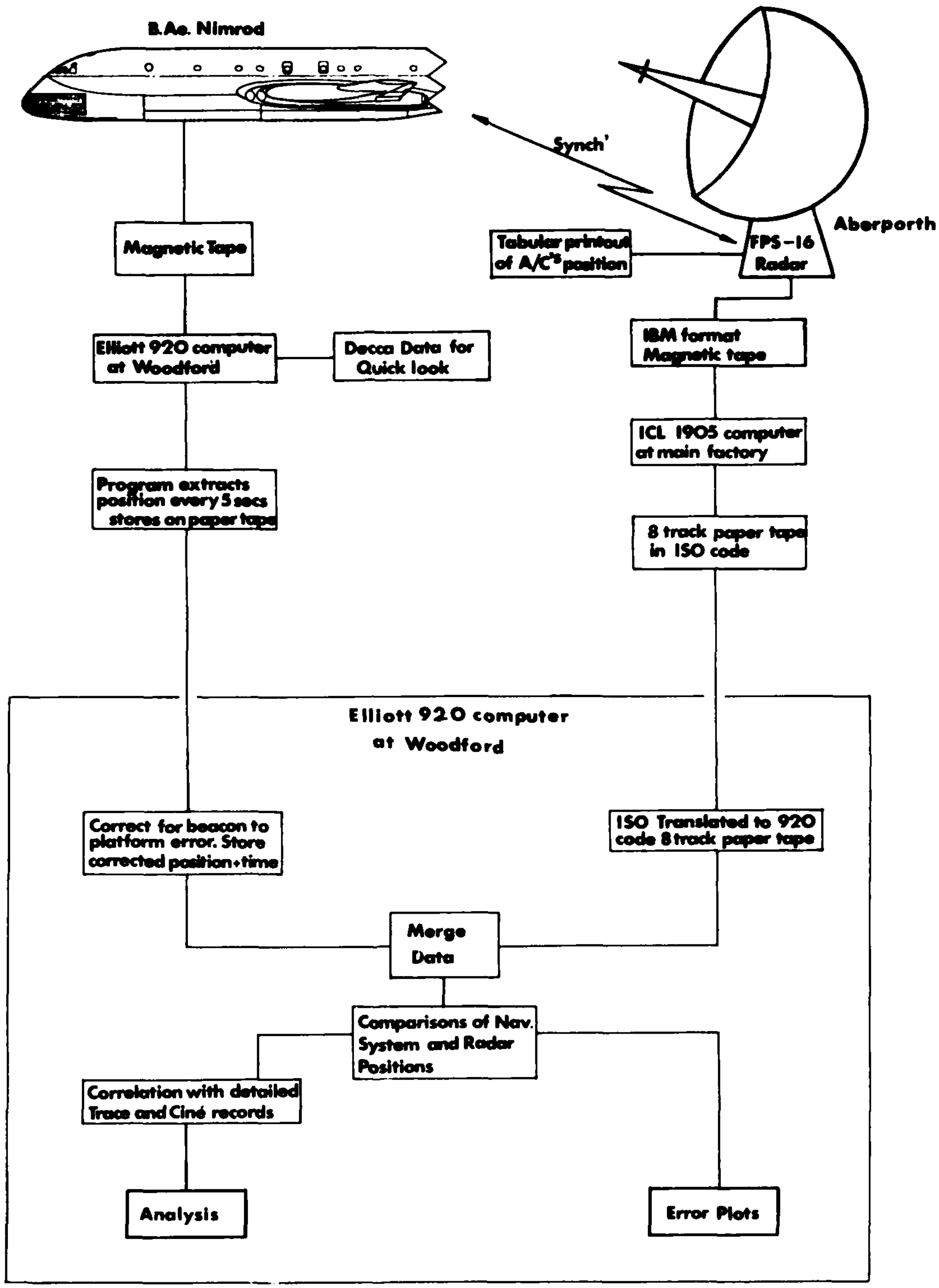


FIGURE 8

NIMROD MR MK.1
DATA RECOVERY FLOW CHART

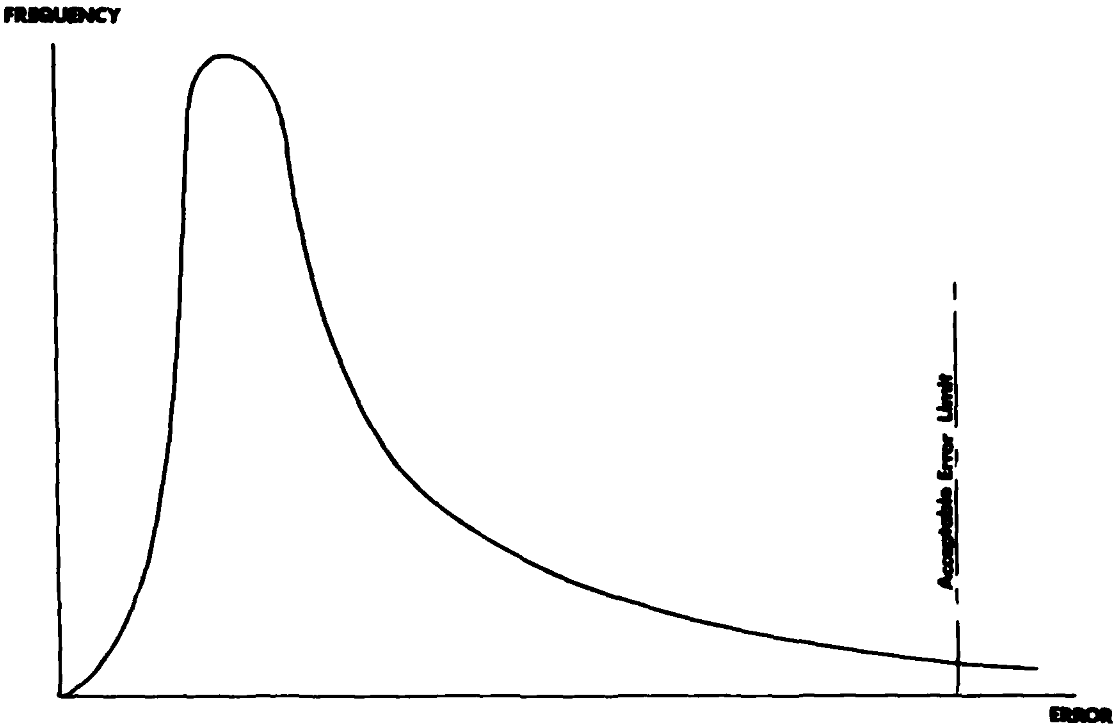


FIGURE 9 - NIMROD MR MK.1 PRODUCTION NAVIGATION SYSTEM RESULTS

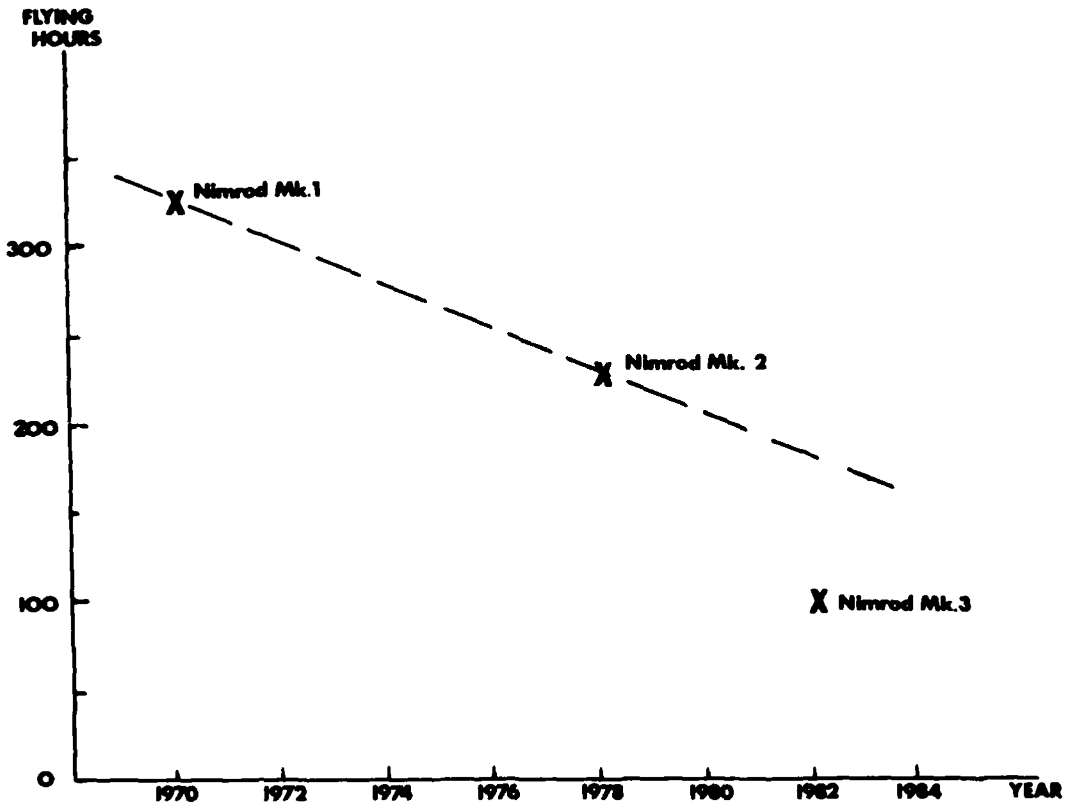


FIGURE 10 - NIMROD NAVIGATION SYSTEMS-FLIGHT TRIALS EFFORT

ESSAI D'UN SYSTEME INTEGRE DE PILOTAGE, NAVIGATION ET VISUALISATION (CARAVELLE ALIS)

par
Alain KLOPFSTEIN
Ingénieur Navigant d'Essais
au
Centre d'Essais en Vol
91220 BRETIGNY/ORGE
FRANCE

RESUME

La complexité des équipements d'avionique a fait apparaître le besoin de systèmes intégrés à bord des avions. Un système expérimental a été installé sur une Caravelle. Il assure les échanges d'informations entre les commandes automatiques de vol, les visualisations de pilotage tête haute et tête basse, et les calculateurs de navigation et de guidage.

Les essais, qui viennent de commencer, porteront sur les améliorations possibles des capacités opérationnelles de l'avion compte tenu de l'association des diverses fonctions et de l'intégration des interfaces avec l'équipage.

1. GENERALITES

1.1. But de l'étude

L'explosion technologique qui a suivi l'introduction du transistor puis de la microélectronique dans les équipements embarqués, et dont l'apparition des mémoires à bulles est la plus récente manifestation, continue à se propager plus vite que la réflexion des utilisateurs potentiels à concevoir des systèmes cohérents. Les divers équipements embarqués se sont sophistiqués séparément, grâce à ces nouvelles possibilités technologiques, et certaines fonctions ou calculs sont souvent réalisés plusieurs fois dans l'avion. (Mesure des assiettes, calculs de rotation d'axes, calculs de vitesses etc ...).

Le but initial de l'étude financée par le Service Technique des Télécommunications et des Equipements (STTE) de la DTCA est de favoriser le développement de systèmes dits"intégrés", conçus pour éviter les duplications inutiles en faisant circuler les données d'intérêt général vers les différents équipements qui en ont besoin, et optimiser la répartition des fonctions communes à l'ensemble de l'avionique.

Mais cette approche, qui consiste à considérer globalement toute l'avionique comme un système, conduit rapidement à de nouveaux concepts d'utilisation opérationnelle des avions. L'intégration des fonctions de visualisation et de commandes qui sont les interfaces entre l'homme et la machine, a des conséquences sur le poste de pilotage de l'avion (planche de bord et commandes de vol) et l'aboutissement naturel de la démarche est l'intégration de l'ensemble de l'avion, de ses équipements ... et de l'équipage, en un système optimisé. Et c'est pourquoi le problème de la charge de travail de l'équipage, abordé au départ en terme de diminution de cette charge, sera peut-être réexaminé au cours des essais, en terme de transfert et de changement de nature de la charge, afin de faire collaborer l'équipage, au mieux de sa capacité, avec le reste du système.

1.2. Choix du système en essais

Le souci d'avoir un support d'étude suffisamment ouvert et non limitatif, a orienté le choix du thème d'étude vers un système d'avion de transport civil long courrier. Les fonctions traitées sont :

- le pilotage (jusqu'à l'atterrissage en catégorie III)
- la navigation en zone océanique et en zones terminales
- les visualisations et commandes associées aux fonctions précédentes ou liées à la gestions des radiocommunication

Le choix du Centre d'Essais en Vol (CEV) pour l'avion banc d'essai s'est porté sur une Caravelle pour des raisons de coût et de facilités de mise en oeuvre. Un chantier de 18 mois l'a transformé en Avion Laboratoire d'Intégration de Systèmes (ALIS).

2. PRESENTATION DU SYSTEME EXPERIMENTAL

Conduite par les Services Officiels (STTE et CEV) l'opération ALIS utilise des équipements existant ou dérivés d'équipements existants, produits par les principaux équipementiers français, coordonnés par l'AEROSPATIALE à qui a été confiée la responsabilité de maître d'oeuvre pour l'avionage du système.

2.1. Architecture du système

L'architecture retenue réalise un compromis entre - d'une part une structure classique par liaison numérique multiplexée (data bus) bien adaptée à l'échange d'informations entre équipements, - d'autre part une structure respectant le principe de "ségrégation" entre équipements dont les pannes mettent en jeu la sécurité de l'avion.

Le synoptique du système est représenté sur la figure n° 1.

La liaison numérique série appelée SIGMA dont la logique de fonctionnement est du type NORME MIL 1153, est composée de 2 bus redondants. (Voir annexe 2).

Chaque bus est géré par un calculateur qui, assurant aussi les calculs de navigation et de guidage, est appelé Calculateur de Navigation et de Guidage (CNG). Les deux CNG qui font office de calculateurs principaux se surveillant mutuellement et chacun peut gérer les deux bus en cas de défaillance de l'autre.

Les bus sont alimentés en données primaires issues des senseurs, par des concentrateurs de données (CD) portes d'entrée du système. Ces concentrateurs effectuent les conversions et les mises au format nécessaires pour mettre les informations au standard SIGMA, et assurent le dialogue avec les CNG. Ils fournissent également les informations à l'ensemble des commandes automatiques de vol (CADV).

Le principe de ségrégation est appliqué aux ensembles de commandes automatiques de vol (CADV) et de visualisation de pilotage, qui participent à l'atterrissage tout temps :

- Les CADV qui comprennent principalement un calculateur Pilote Automatique (PA) monitoré et un calculateur automanette (AM) également monitoré, reçoivent les données par une liaison numérique (ARINC 429) en provenance des CD, sans passer par la liaison SIGMA. (Toutefois pour des phases de vol non critiques (navigation) des données élaborées par les CNG arrivent aux CADV via la liaison SIGMA et les CD).

- L'ensemble de visualisation comprenant deux boîtiers générateurs de symboles (BGS) qui alimentent les deux collimateurs de pilotage tête haute (HUD en anglais) et les écrans tête basse (HDD en anglais), reçoit les données nécessaires à l'élaboration des images par la liaison SIGMA.

2.2. Composition du système

L'architecture du système telle qu'elle vient d'être décrite permet d'avoir un ensemble opérationnel après panne pour les phases d'atterrissage (CADV passivé après première panne + visualisation passivée après première panne et indépendance entre CADV et visualisation).

Pour obtenir un système effectivement opérationnel après panne il faudrait toutefois tripler les éléments communs, et en particulier les concentrateurs de données et certains senseurs.

Pour des raisons financières, on a fait l'impasse sur le triplement de ces équipements, ce qui n'altère en rien les principes et les résultats de l'opération, mais ce qui peut apporter certaines contraintes opérationnelles aux essais en véritables conditions IMC.

En outre on verra plus loin que certains traitements de l'information peuvent éviter le triplement des senseurs.

Les CD sont donc limités à deux, la provision existant néanmoins pour le troisième, et en règle générale les senseurs sont doublés :

2 centrales à inertie (INS)
2 radio altimètres
1 centrale anémométrique (ADC) + provision pour une deuxième
2 ILS
2 VOR
2 DME
2 sondes d'incidence.

Les organes de dialogue entre les opérateurs et le système se composent

- d'un boîtier de reconfiguration système permettant de sélectionner un CNG ou un BGS en cas de panne
- des matériels spécifiques classiques pour les CADV : poste d'engagement, panneau de commande des modes supérieurs et indicateurs de modes et de situation d'atterrissage
- d'une boîte de commande de radio navigation (BCRN) qui permet soit la commande et l'affichage manuel des fréquences, soit le passage en mode de gestion automatique par les calculateurs CNG du système
- pour la visualisation de pilotage et la navigation système (exécutée par les CNG) d'une boîte de commande et de visualisation dérivée des CDU (control data unit) de systèmes de navigation automatisés ARINC 424 MK 1.3 comprenant un tube cathodique et un clavier numérique associé.

Dans une deuxième phase vers fin 1981 sera implanté un organe de dialogue centralisé appelé Poste de Commande Multiplexé (PCM) composé d'un tube cathodique et d'un clavier de touches à fonctions multiplexées, permettant la commande CADV et des fréquences radio en plus des commandes de navigation et de visualisation. Ce PCM sera associé, pour la conduite de l'avion, aux organes déjà existants implantés sur le manche à balai :

- palette de déconnexion reconnexion rapide du PA pour pilotage transparent (type CWS)
- commandes incrémentales des modes de base du PA (assiette et route dans un premier temps, pente et route dans un deuxième temps) utilisant la commande de trim ("sapin" de trim) par impulsions courtes calibrées ou par rampes calibrées (impulsion prolongée).

2.3. Installation d'essai

2.3.1. Groupe auxiliaire de puissance (auxilliary power unit APU)

La Caravelle ALIS a été équipée d'un APU de 40 kw assurant la ventilation au sol des équipement, et la génération électrique de tout le système en essais(dans tout le domaine de vol) indépendemment de la génération normale de l'avion "de base".

2.3.2. Poste de pilotage (voir figure 2)

Le poste de pilotage de la Caravelle a été assez profondément modifié. Les nouveaux organes de commande et de visualisation ont été chaque fois que possible, soit doublés, soit mis à disposition des deux places pilote (2 HUD, 2 HDD + provision pour un troisième, 2 BCRN, un bandeau PA au milieu). Un seul CDU système est implanté en place gauche, mais dans la deuxième phase il est prévu deux PCM en poste de pilotage.

Cependant la place gauche a été plus spécialement conçue pour l'essai du système, et la place droite pour la pilote "de sécurité". En particulier on trouve en place gauche des instruments de pilotage électromécaniques à entrée numérique (ADI, anémomachmètre, altimètre) tandis qu'en place droite ont été conservés les instruments d'origine de l'avion de base, indépendants du système en essais.

2.3.3. Cabine

Les équipements composant le système sont logés dans des armoires adaptées, où ils sont finalement accessibles pour toute intervention.

On trouve en outre en cabine :

- un poste ingénieur d'essai reproduisant partiellement le poste pilote, avec les principaux instruments de pilotage, le deuxième CDU, un HDD, et un moniteur de télévision retransmettant tout ce que voit le pilote à travers son HUD.

Les commandes de la génération électrique d'essai sont aussi regroupées au poste ingénieur d'essai.

- un poste "mise en oeuvre" à partir duquel des vérifications de fonctionnement des divers équipements composant le système sont possibles
- un poste "installation de mesure" qui regroupe les commandes et les visualisations de l'installation de mesure et de dialogue avec le calculateur de mesure, un enregistreur graphique et les visualisations numériques.

2.3.4. Installation de mesure

Un système d'acquisition numérique décentralisé de type B.O.A. permet l'acquisition des informations circulant sur les diverses liaisons numériques du système (bus SIGMA et liaisons ARINC) ainsi que les paramètres numériques ou analogiques entrant dans les concentrateurs de données. En outre sont acquis les principaux paramètres avion.

Les données sont enregistrées

- soit sur enregistreur graphique pour visualisation immédiate
- soit sur enregistreur numérique à cassette pour préexploitation rapide
- soit sur enregistreur numérique classique pour exploitation complète en temps différé.

Une station de mise au point des logiciels du calculateur de mesure (identique aux CNG) qui peut rester à demeure sur l'avion mais est utilisée exclusivement au sol, permet de plus la préexploitation de l'enregistreur à cassette.

En outre 3 magnétoscopes enregistrent

- ce que voit le pilote dans le HUD (symbologie + vision extérieure)
- la visualisation HDD
- les gestes du pilote.

Il est également prévu d'enregistrer dans certaines phases, grâce à un système dit oculomètre, l'endroit précis où se porte le regard du pilote.

3. ESSAIS EN VOL

3.1. Mise au point de la plateforme d'essai

Avant d'évaluer l'apport du système intégré en tant que tel, il faut vérifier que son fonctionnement est bien conforme à ce que l'on en attend.

C'est cette phase de mise au point et de vérification qui est actuellement en cours, et qui inclut l'installation d'essai.

Le fonctionnement individuel des composants du système est vérifié, puis on examine la dialogue entre les équipements, qui constitue en fait le fonctionnement du système.

Ces essais devant couvrir tout le domaine de vol sont assez longs à réaliser. Commencés en juillet 80 par la mise au point de l'APU ils devraient s'achever en mai 81 après la mise au point du pilote automatique et son intégration à l'ensemble, qui constitue la tâche la plus longue de cette première phase.

3.2. Augmentation des capacités opérationnelles

Le couplage des informations de navigation des deux centrales à inertie, avec celles des couples de balises DME - DME ou VOR/DME, fournit une navigation précise(fraction de nautique), surtout en zones terminales, où les balises sont plus nombreuses. Le système, qui choisit le couple de balises le mieux placé par rapport à l'avion pour recaler la navigation inertielle, placera l'avion dans de bonnes conditions pour entreprendre des approches IFR sur terrains faiblement équipés.

L'association d'une visualisation de pilotage tête haute et de commandes incrémentales des valeurs de consignes des modes de tenue de pente et de route du pilote automatique, doit permettre avec l'aide supplémentaire de l'automanette le suivi d'une trajectoire d'approche définie en pente et en route indépendamment de toute information au sol. La hauteur jusqu'à laquelle cette trajectoire pourra être poursuivie avant d'acquérir les repères au sol permettant la poursuite à vue de l'approche et l'atterrissage sera définie au cours des essais.

Pour un terrain sans installations ILS l'objectif est à priori la hauteur de 200 pieds (approche IFR catégorie 1).

Bien évidemment des essais seront également effectués sur des terrains équipés d'ILS de catégorie 1, 2 et 3, où les informations ILS pourront être utilisées par le système, en couplage avec ses données propres, assurant ainsi, en particulier pour les ILS les moins performants, un filtrage des informations.

Le système sera alors utilisé avec les modes supérieurs du PA, la visualisation tête haute servant à surveiller celui-ci pendant toute la phase d'approche et

- soit à assurer la transition et l'exécution de la finale en pilotage manuel à vue (approches catégorie 2)
- soit à aider à la décision de poursuite ou de remise de gaz au passage de la hauteur de décision pour les approches en catégorie 3.

Dans tous les cas précédents, différents couplages, figurations, et lois de pilotage, seront comparés pour optimiser le système et aboutir aux minimums opérationnels les plus bas.

Il faut noter que dans le cas présent de la Caravelle ALIS il ne s'agit pas de refaire des essais de PA ou de HUD qui ont déjà été menés sur d'autres avions du CEV, mais de faire les essais d'un système complet, chaque dispositif d'aide au pilotage ne trouvant sa pleine efficacité qu'au sein du système global.

3.3. Optimisation du dialogue pilote / système

Un des objectifs importants du système est de libérer le pilote de certaines tâches "tactiques" liées aux qualités de vol de l'avion telles que le suivi d'une trajectoire, pour lui permettre de se consacrer aux décisions "stratégiques" telles que le choix de la trajectoire ou la décision de poursuite de la phase d'atterrissage. (D'où l'importance des commandes de type incrémental permettant la modification rapide d'une valeur de pente ou de route, le PA se chargeant de maintenir la valeur choisie ou d'effectuer la modification demandée).

Mais pour atteindre ce but, il est nécessaire de disposer d'organes de commande et de visualisation qui soient facilement utilisables et qui n'apportent pas de charges nouvelles à l'équipage. Il semble particulièrement important que l'affichage d'un paramètre se fasse de façon unique, même s'il intervient dans plusieurs équipements différents : ainsi l'affichage d'une route commandée par le PCM ou par la commande incrémentale sur le manche, doit réagir sur le PA, et sur la figuration HUD et HDD aussi bien que sur l'écran cathodique du PCM.

Les essais du CDU type ARINC MK 1.3 dans la phase transitoire, puis du poste de commande multiplexé (PCM) ensuite, seront orientés vers l'évaluation de ces charges.

On examinera en particulier l'influence de l'intégration de commandes du PA dans le PCM sur la rapidité d'action du pilote. De même la commande des fréquences de radiocommunication et de radionavigation à partir du PCM sera comparée à la commande à partir des boîtiers actuels.

3.4. Optimisation des redondances

La sécurité des aéronefs impose le plus souvent un doublement et même un triplement de certains senseurs ou de certains équipements.

La capacité d'un système intégré à faire circuler des informations et à les traiter peut supprimer certaines redondances, soit en surveillant les évolutions de paramètres dont on connait à priori les limitations en bande passante, soit en recoupant les valeurs de paramètres de nature différente, mais non indépendant entre eux (liés généralement par les lois de l'aérodynamique et de la mécanique du vol pour les paramètres fondamentaux).

Ainsi les calculateurs principaux du système (CNG) effectuent des calculs de surveillance sur l'évolution des assiettes et des accélérations de l'avion, et des calculs de vraisemblance entre l'incidence, la vitesse aérodynamique, la vitesse verticale et l'assiette longitudinale.

Ces traitements permettent lorsque des valeurs entre deux senseurs dépassent un certain seuil, de déterminer lequel est faux, sans recourir à la valeur d'un troisième senseur.

Les essais en vol devront déterminer les seuils praticables et vérifier les capacités de détection d'erreurs.

Si les résultats sont positifs, des études d'extension, des vérifications par calculs de vraisemblance pourront être poursuivies.

3.5. Méthodes d'évaluation

Comme il a déjà été dit une caractéristique importante de l'expérimentation est de s'attacher aux résultats globaux et non aux performances de tel ou tel équipement. Parmi les grandeurs mesurables représentatives de cet aspect global, on s'interessera en premier lieu à la trajectoire de l'avion et à ses écarts par rapport à la trajectoire optimale (roulage inclus), ainsi qu'à la manière dont il parcourt sa trajectoire (incidence, vitesses, facteur de charge).

On mesurera aussi les temps de décision dans les différentes phases de vol, définir par les délais entre l'instant théorique où la décision peut être prise, (arrivée des informations nécessaires sur les interfaces avec le pilote) et le début d'exécution par le pilote des actions consécutives à la décision (action sur les commandes de l'avion ou sur les postes de commande du système, annonces verbales).

D'autres résultats seront d'ordre qualitatifs, comme la nature des décisions prises, le taux d'erreur de manipulation, les avis des pilotes sur la disponibilité de décision, la "mise dans la boucle de pilotage" et la facilité d'exécution des actions.

D'autre part, l'enregistrement des gestes du pilote et du balayage du regard donneront des indications sur la nature et le volume de la charge de travail.

Si des premiers résultats quantitatifs, il est aisé de tirer des mesures de performances, il sera plus difficile d'évaluer, à partir de l'ensemble des résultats, des indications sur les gains de sécurité des vols.

En effet si la probabilité de panne d'un composant du système ou de l'avion peut être évaluée, ainsi que ses conséquences notamment dans les phases de pilotage automatique, il est toujours difficile d'évaluer la capacité du pilote à faire face à un évènement imprévu, et la réalisation au simulateur et à plus forte raison en vol, de telles situations est délicate et ne peut être repétitive si l'on veut garder le caractère imprévu de l'évènement.

Une réflexion sur la nature des informations accessibles au pilote et sur la nature de sa charge de travail doit donc être menée parallèlement aux résultats que l'on peut obtenir.

4. CONCLUSIONS

Après la mise au point en vol de l'outil que représente la Caravelle ALIS, et dont l'achèvement devrait avoir lieu dans le deuxième trimestre 81, une première tranche d'évaluation s'étendra jusqu'à la fin 81.

A l'issue de cette tranche d'essais, un chantier interviendra pour effectuer la mise en place d'équipements non encore disponibles aujourd'hui et en particulier le poste de commande multiplexé et des équipements radiocommunication commandables par le PCM.

D'autres modifications pourront intervenir en fonction des premiers résultats obtenus ou des nouvelles études en cours.

En effet la conception et l'architecture du système permettent d'envisager l'intégration de fonctions nouvelles. Parmi celles-ci on peut déjà citer

- l'approche MLS en pilotage manuel (pour laquelle la figuration d'interception d'axe de type VEGA déjà programmée peut être essayée)
- le vol en croisière économique (Flight Management)
- l'étude de lois de montée optimales adaptables aux avions d'armes (vols à point de polaire constant).

Tous ces développements seront menés avec le souci d'adjonction du plus petit volume d'équipements supplémentaires et d'utilisation maximale des paramètres existants et des calculs déjà effectués.

Des modifications d'architecture pourront également intervenir si certains problèmes de redondance sont résolus (couplage des CADV sur le bus SIGMA).

La capacité du système à absorber sans trop de difficultés ces problèmes nouveaux sera l'un des meilleurs tests de la souplesse et de l'intérêt de la conception du système.

A N N E X E 1

REFERENCES DES PRINCIPAUX EQUIPEMENTS CONSTITUTIFS DU SYSTEME

C N G	- Calculateur de navigation et de guidage - Constructeur SFENA Réf. UMP 7.800
B G S	- Boîtier générateur de symbole - Constructeur TH / CSF Réf. BGS/ALIS
H U D	- Collimateur tête haute (Head Up Display) - Constructeur TH / CSF Réf. TC 125 monochrome
H D D	- Visualisation tête basse (Head Down Display) - Constructeur TH / CSF Réf. VMC 210 B trichrome
C A D V (PA et AM)	- Commandes automatiques de vol - Constructeur SFENA Réf. A 40 A avec calculateur PA/Directeur de vol B185AAM et calculateur Automanette B186AMM
C D	- Concentrateur de données - Constructeur SFIM Réf. UTG101+VAM 25/2
I N S	- Centrale à inertie (Inertiel Navigation Système) - Constructeur SAGEM Réf. ULISS 46
A D C	- Centrale aérodynamique (Air Data Computer) - Constructeur CROUZET Réf. 70 G
D M E	- Interrogation DME - Constructeur LMT Réf. 3574 D
V O R	- Réception VOR - Constructeur EAS Réf. DVR 740
I L S	- Réception ILS - Constructeur BENDIX Réf. RIA 33 A
R A	- Radio-altimètre - Constructeur TRT Réf. AHV 5

A N N E X E 2

RAPPEL DU PRINCIPE DU BUS SIGMA

Le Bus SIGMA est conforme aux recommandations du Comité Technique Intégration (2ème édition Mars 1975). Les principales caractéristiques sont les suivantes :

. Transmission série
. Structure à dérivation sur ligne bidirectionnelle
. Capacité 32 abonnés ou unités
. Rythme de 1 M Bits par seconde (à ± 1%) - Code Manchester biphase (trois niveaux + 3 volts, 0 volt, - 3 volts à ± 0,3 Vprès)
. Support paire torsadée blindée de longueur 100 m au maximum
. Message composé d'octets assemblés en nombre variable
. Protection des échanges par parités croisées
. Protection contre les courts-circuits et isolement par transformateur des dérivations
. Procédures à l'initiative de l'une des Unités (Unité de Gestion) en mode commandes-réponse
. Niveau de redondance acceptable.

1. DEFINITIONS

1.1. Unité Périphérique ou de Gestion (UP ou UG)

Nom donné à toute unité branchée sur la ligne Bus. L'unité de gestion qui assure la gestion travaille en mode Maître, les unités périphériques en mode Esclave.

La gestion peut être centralisée : la gestion se fait par une seule unité ou groupe d'unités si la redondance est nécessaire.

La gestion peut être décentralisée : la gestion est assurée à tour de rôle par différentes unités qui sont à tour de rôle Maître ou Esclave.

1.2. Coupleurs

Nom donné à des ensembles fonctionnels faisant partie des unités décrites précédemment.

1.2.1. Coupleur de bus - C.D.B.

Le C.D.B. assure toutes les fonctions liées à la transmission par ligne Bus.

Il peut être tour à tour Maître (coupleur de l'U.G.) ou Esclave (coupleur de l'U.P.).

1.2.2. Coupleur sous-système C.S.S.

Le C.S.S. assure l'adaptation entre un ou plusieurs sous-systèmes standards et le C.D.B.

Le C.S.S. peut être limité à quelques modules ou ne pas exister si la compatibilité entre le C.D.B. et le sous-système qu'il connecte au Bus est assurée.

1.3. Boîtiers de Dérivation

Dispositif de connexion du C.D.B. sur la ligne Bus.

Il existe deux types :

. B.D. - utilisés pour la connexion de toutes les unités sur la ligne sauf les unités terminales.
. B.D.A. - utilisés pour les connexions des unités terminales, ils comportent le dispositif d'adaptation.

Ces boîtiers sont protégés contre les courts-circuits, en cas de court-circuit sur une dérivation le niveau du signal sur la ligne ne doit pas chuter de plus de 10%. Un isolement galvanique tant à l'émission qu'à la réception doit être prévu.

2. CARACTERISTIQUES DES ECHANGES

2.1. Définitions

. Caractère : Elément constitué d'un nombre fixe indissociables à la transmission.
. Message : Ensemble de caractères transitant sur le Bus en un seul bloc et dans un seul sens.
. Echange : Ensemble de messages transitant sur le Bus à partir d'une seule initiative de l'Unité gérant le Bus.

2.2. Principe des échanges

2.2.1. Initiative des échanges

Dans une phase déterminée, les échanges s'effectuent sous le contrôle d'un maître unique (U.G.) en mode Commande/Réponse. Cela veut dire qu'une U.P. n'a le droit d'émettre que si elle a été interrogée par l'U.G.

2.2.2. Origine et destination

Les transferts d'information se font de trois façons :

1/ - U.G. à U.P.
2/ - U.P. à U.G.
3/ - U.P. à U.P.

2.2.3. Type d'adressage

Deux types sont prévus :

. Adressage individuel : une seule U.P. est destinataire
. Adressage général : tous les U.P. sont destinataires.

2.2.4. Synchronisation

1/ Horloge de base : La ligne Bus fonctionne de manière asychrone en mode "Half duplex". Cet asychronisme implique une horloge dans chaque unité.

2/ Temps de réponse d'une U.P.: Le temps s'écoulant à la suite d'une commande de l'U.G. entre le dernier bit du message reçu et le début du message de réponse doit être inférieur à 20 μs.

3/ Synchronisation de messages : Elle est assurée par un code invalide correspondant au codage d'un bit 1 en code Manchester biphase durant 3 temps de bits.

2.2.5. Format

1/ Format de caractères : Chaque caractère est consittué d'un octet d'information suivi d'un bit de parité transversale tel que le nombre de 1 dans l'ensemble des 9 bits soit impair.

2/ Format des messages : Chaque message comprend :

- un code de synchronisation : SY
- un caractère de procédure : PROC
- un nombre variable de caractères d'informations banalisés (compris entre 0 et n) : INF
- un caractère de parité Longitudinale : PL
- chaque message est suivi d'un blanc d'une durée au moins égale à 4 micro-secondes.

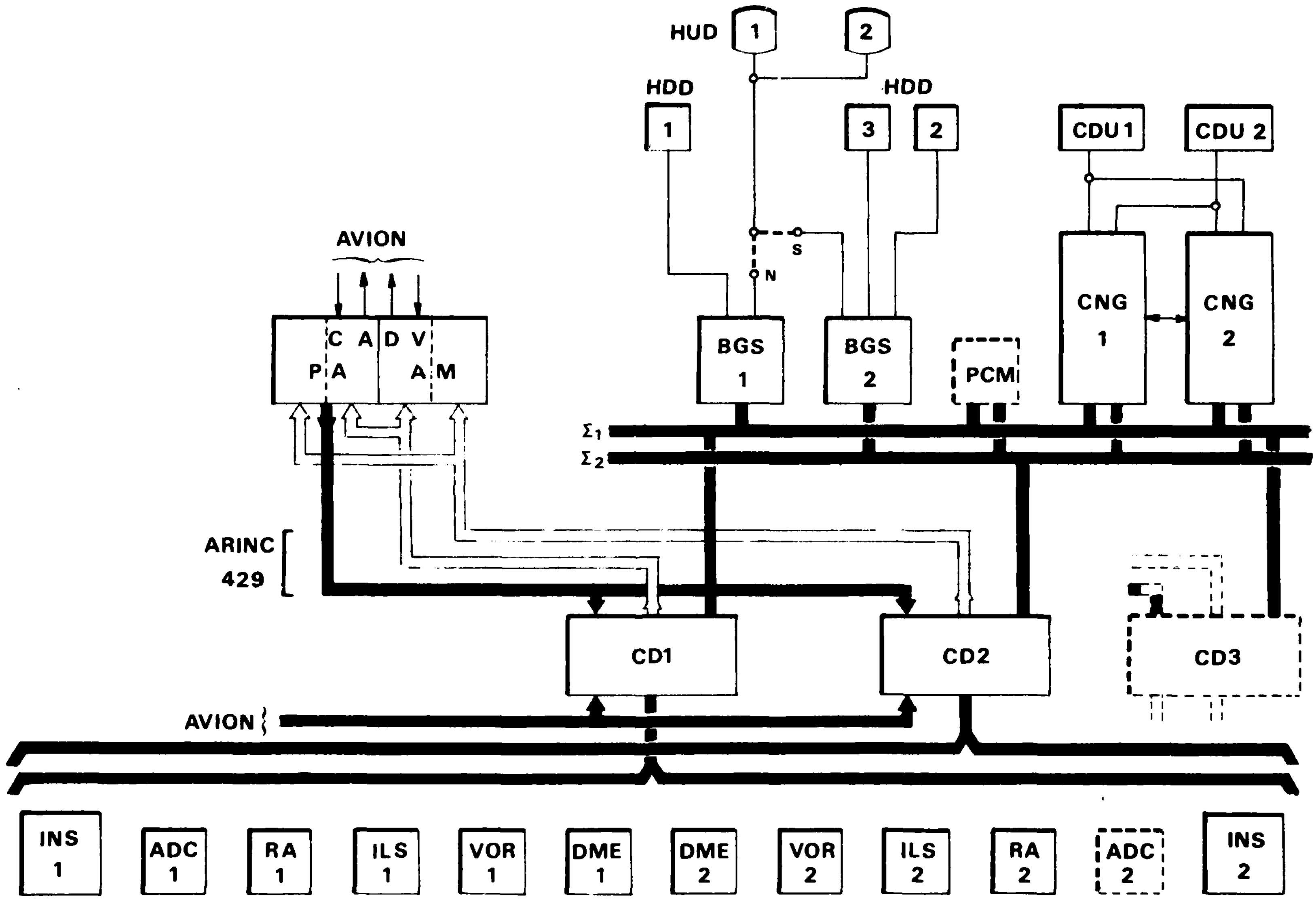


Fig.1 Synoptique du systeme

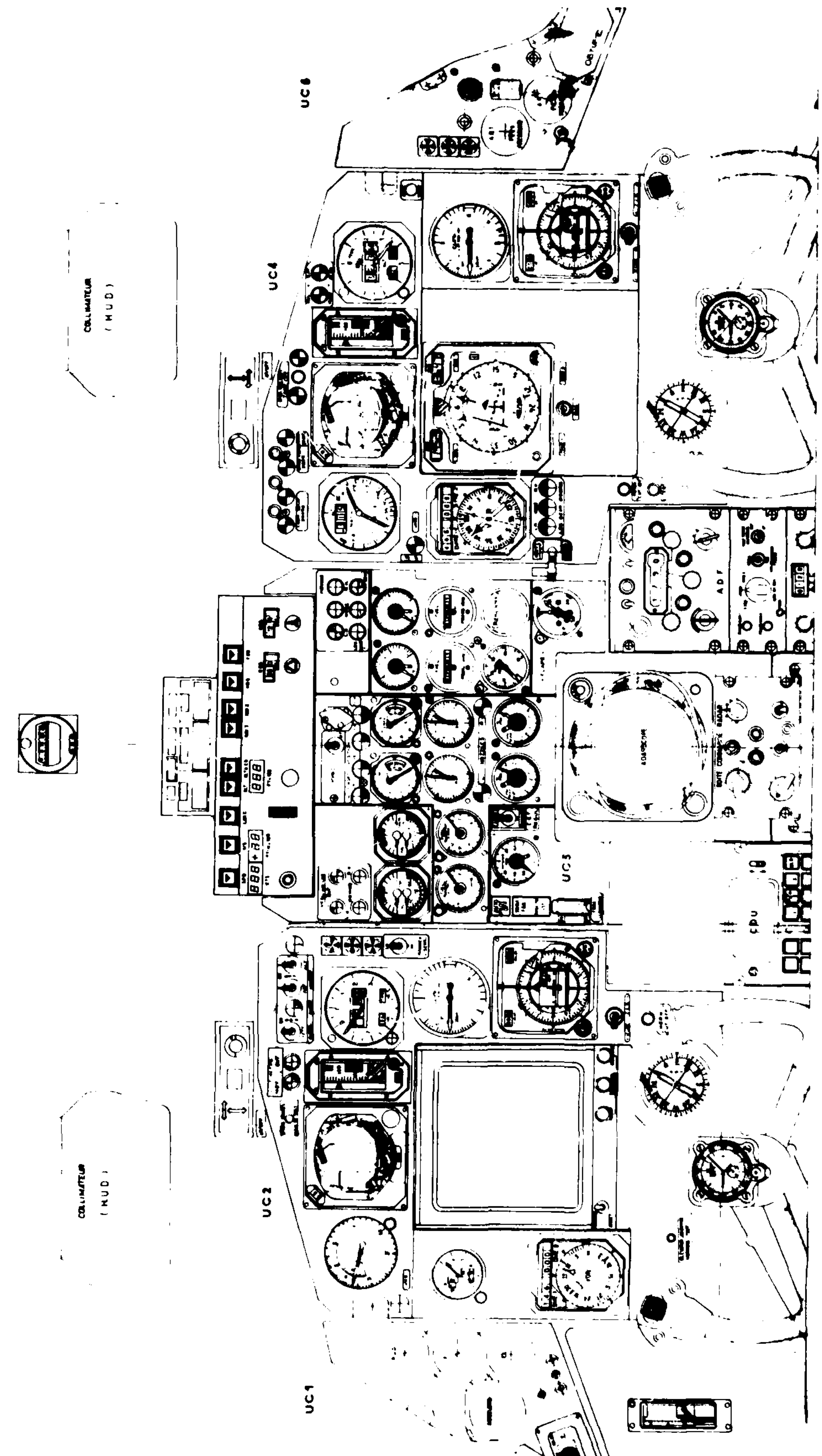


Fig.2 Planche de bord

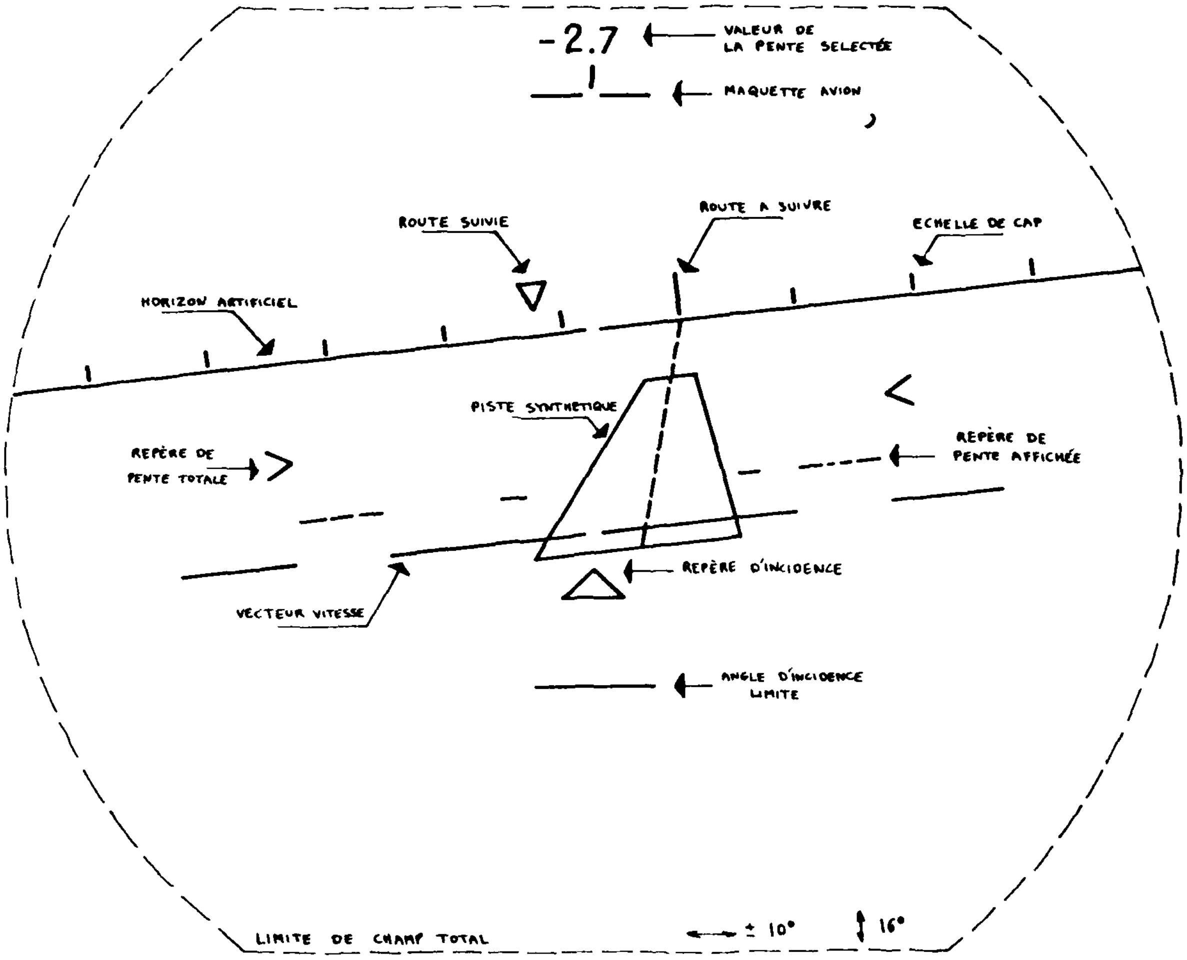


Fig.3 Figuration HUD en approche

ONBOARD AND GROUND TEST OF AN AUTONOMOUS NAVIGATION SYSTEM BASED ON TERRAIN CORRELATION

by

Dipl.-Ing. Horst Dieter Lerche
Vereinigte Flugtechnische Werke GmbH
2800 Bremen 1
Hünefeldstraße 1-5

Germany

SUMMARY

The tremendous progress in microprocessor technology and digital data processing has caused a trend towards high sophisticated avionic systems with decentralized system architecture. A report will be given on the experience gained in developing and testing of a navigation system based upon terrain correlation. A dedicated ground based data management system and special test methods have been developed to perform flight tests. Furthermore typical flight experiments will be discussed.

Contents

1. Introduction

The trend from centralized avionic-processing-system towards so called highly sophisticated systems with distributed microprocessor based architecture in future navigation and flight-control systems has resulted from the tremendous progress made in microprosessor-technology and in development of new mass memory devices. In this context one could remember the availability of high speed single-chip-microprocessors and highly integrated bubble-memory-circuits for application in digital avionic systems.

Classical real-time solutions to problems involving complex mathematical realationships can be performed by parallel computation and parallel processing in a microprocessor system with decentralized architecture. The discussed TERPAC-avionic-system tested in an experimental-program is an example of this system architecture.

The inter-communication between the autonomous microprocessor systems is guaranteed by a high-speed multiplex-data-bus.

A great amount of digital data are stored aboard in a non volatile magnetic-bubble-mass memory, which retains the information through many power cycles.

Further advantages of distributed microprocessor based architecture realized and tested in the following TERPAC-avionic-system, are discussed in Chapter 3.

Since the seperate hardware and software modules are capaple of being tested independently from one another, the simple test capability of this distributed architecture is of great importance, particularly during the integration and experimental phase. Dedicated test equipment which has also been developed in the experimental program is required to perform the pre-flight and in-flight test procedures.

2. TERPAC-system

Navigation and control of aircraft and missiles will be considerably improved by using information of the over-flown terrain stored in the microprocessor based avionic system. Such a terrestrially aided navigation and flight control system like TERPAC (fig. 1) offers the following principle modes of execution:

- inertial mode

- TERCOM-position-fix-mode

- SZEKOR-position-fix-mode

- terrain-following /avoidance-continuous mode

In the inertial mode the kalman-filtered basic navigation-system is responsible for holding of the course when position-fix-outputs are not available.

In the two position-fix-modes the navigation accuracy is improved by an automatic comparison (correlation) of an actual signature of the terrain contour with a reference signature. In the case of terrain-profile correlation (TERCOM) the actual position of the vehicle is computed by matching the scanned terrain signals with digitized terrain values, stored in the TERPAC-mass-memory. In the case of scene-correlation (SZEKOR) the comparison is performed between an actual image of the terrain-contour taken with on image-making sensor and a stored reference image. Wether the position-fix-system is running in a single-fix- or in a quasi continuous mode depends on the significance of the over flown terrain.

On the other hand the terrain-following-system will be provided with the stored terrain-data in the fourth mode in order to optimize the reference flight path against the actual flight path by means of software. The "critical" point on the terrain ahead of the vehicle predicted by the TERPAC-system and the optimal flight path are computed.

The optimal flight-path is as close to all stored terrain-grid-points as practical for controlling the minimum clearance above the terrain. The TERPAC-based terrain-following-system will also be used as back-up-system for a radar-aided terrain - following-system.

The block-diagram in figure 2 shows the principle data flow and the control functions of the TERPAC-system.

The terrain-map mass-memory and the digitized terrain-data stored in it serve as the basic data-source both for the position-fixing-system and for the terrain-following/avoidance-system. In addition, the digitized terrain-data may be pre-processed for projection on a digital-map-display instead of an analogous device or for obstacle warning to the pilot.

The TERPAC-processor as a central processing unit computes in a recursive manner the actual position by correlation-algorithms and in a predicting manner the reference flight-path by optimizing algorithms.

The differential output signals generated in the tracker software module by comparison of the predicted reference flight path with the actual flight path are fed to the flight control system.

Switching over from one mode of action and the inherent sensor subsystem to another will be controlled prior to data pre-procesing.

The second TERPAC- mode (TERCOM-position-fix) is discussed in more detail below, to report about experience gained in an experimental-program to test a combined navigation-system consisting of:

- inertial measurement unit

- radar altimeter subsystem

- data pre-processing modul

- TERPAC-processor

- terrain-mass-memory

- kalman-filter

Fig. 3 shows the typical measurement behaviour of an inertial navigation system without and with TERPAC-fixes. The position error increases proportionally to the time flown after launch. The overall navigation accuracy is considerably improved by feedback of TERPAC-fixes over a kalman filter to the outputs of the inertial measurement unit. The resulting position accuracy is in the order of JTIDS-performance parameters.

Furthermore the error behaviour of a TERPAC aided inertial navigation system is independent of the travelling time and thus of the range in the operational case.

The values of clearance altitude and the altitude above datum-plane are continuously measured by the radar altimeter and the baro aided vertical channel of the intertial measurement unit (fig. 4), and are simultaneously stored in a pre-processing stage. After pre-processing (noise-filtering, data reduction, time synchronisation) the measured terrain profile is generated as differential signal and fed to the TERPAC-correlator stage.

This processing method guarantees that the motion of the vehicle in the vertical plane is eliminated, ensuring that additional terrain elevations cannot be simulated. These additional terrain-elevations could reduce the resulting position-fix accuracy of the TERPAC-aided inertial navigation system.

On principle the TERPAC-system can be applied to manned or unmanned vehicles, but the requirements differ greatly. In contrast to a preplanned mission with preplanned flight path in missile-guidance, the TERPAC-system must guarantee maximum flexibility and will be subordinated to the basic navigation system in manned aircraft.

The following reqirements are typical for application in manned aircraft:

- variable track angles

- no limitted flight-envelopes

- course correction during TERPAC-updates

Fig. 5 shows the principal data-flow of the combined navigation system. The overflown terrain contour is calculated from the differential output signal of baro sensor and the platform vertical channel and from the radar altimeter outputs. The actual position of the vehicle corresponds to the correlation maximum, computed in the TERCOM unit. Not only the accuracy of the position outputs is improved updating by means of kalman filtering, but also of velocity and attitude angle components.

3. Avionic system architecture

The TERPAC- navigation-system (fig. 6) consists of the following autonomous sub-systems:

- altitude measurement unit

- terrain correlation system (mass-memory, correlator)

- baro-sensor

- inertial measurement unit

- kalman filter

Further parts of this communication system are the telemetry and the test terminal.

The test terminal gives a flexible data management capability, which provides the system on board of the aircraft with terrain map data and test stimuli during the pre-flight phase via a data line up to 30 metres. During the in-flight phase the error status of the realtime system is supervised via the telemetry terminal.

The autonomous sub-systems are inter-connected via a multiplex data bus of 1 MHz rate. The data duty cycle is considerably reduced, because only the directly usable outputs are transmitted between the sub-systems. This data reduction is abtained by realtime pre-processing of the independent software-tasks by microprocessors within the terminals of each autonomous sub-system.

Owing to the high data rates during the correlation process on the one hand and the urgency of time synchronization of altitude measurement on the other hand, subsystems one and two are inter-connected via a high-speed correlator bus. The I/O-control between the hierarchically structured multi-data-bus-system is performed by terminal A.

This distributed microprocessor based architecture proved successful in the early development and integration phase of the experimental-program (fig. 7).

The development was efficient by breaking the total function of the TERPAC-avionic-system down into autonomous subsystems with independent software modules processed in parallel by standard microprocessors and by defining simple standarized I/O-interfaces. The constructed modularity allows hardware and software elements to be modified simply and quickly.

Uncritical software timing and asynchronous operation are further advantages of the realized avionic architecture.

The clear arrangement of the hardware and software structure permits a greater degree of subsystem check-out and a simple system testing procedure in the integration and testphase.

The subsystems have been separately tested like stand alone systems in the course of stepwise integration into the total system.

It has thus been possible to isolate the problems of digital signal processing by multiplex-data-bus from those of software computation of the sub-systems.

4. Ground based data management system

Prior to the test flight the ground based data management system (fig. 8) mainly provides the highly sophisticated TERPAC-system with pre-processed terrain-map data and additional test data. Typical flight test data are:

- packed terrain elevation data
- map header data
- way point coordinates
- subsystem test data strips

In addition to these test data test stimuli are transmitted in order to initialize bite-tasks in the application software of the TERPAC-system. The results of the test computation are transmitted back to be valued by the system engineer on a display. In this way it is possible to decide at an early test phase, whether the terrain map data stored in the mass memory of the TERRPAC-system are valid or not.

By insering the way point coordinates into the data management computer, adequate terrain map data are generated from a central terrain data base. The software tasks are:

- selection of test area
- transformation into the navigation coordinate system
- grid point interpolation
- data packing and formatting

Before transferring data to the mass memory of the avionic system, identification bits are added to the digital altitude word for error diagnostics.

The ground based data management system utilizes a stationary general purpose computer und a mass memory unit managing the terrain data base and the data pre-processing.

A portable ground test terminal is interconnected via a telephone data link. The test data can either be directly transferred to the mass memory of the avionic system via a flexible data line and aircraft interface or stored in the bubble memory of the ground test terminal.

Other ground based components are graphic display, plotter and line printer.

The design of the data management systems provides for great flexibility during flight tests starting from various aircraft positions.

5. Flight test methods

The highly sophisticated design of TERPAC-system requires dedicated test methods for performance of the overall acceptance test. In the pre-flight test phase, the functions of the autonomous subsystems such as inertial measurement unit, baro-sensor and radar altimeter are statically testes with approved methods.

The alignment status and indicated present position of the inertial measurement unit, the measured static pressure value above datum plane and the corresponding output signal of the radar altimeter simulated by a delay circuit between the transmitting and receiving antenna are of interest.

Testing of the software implemented in the decentralized microprocessor system, testing of the data processing capa ility via multiplex data bus and a mass memory validity check out are the key to TERPAC-testing. Two methods have mainly been applied: the self diagnostic method and the stored response method.

The self diagnostic method includes outomatic testing of ram and data frame by bit-pattern and in flight worst case computation. For instance, the required correlation time is continuously compared with a limitation value, e.g. in the case of altimeter measurement the travelling time of the transmitted electromagnetic signal. If an error occurs, an error bit is set in the status word. After each data transmission from and to the mass memory via the multiplex data bus a checksum test is automatically initialized.

The purpose of the stored response method is to test the application software by means of simulation. A typical set of subsystem date outputs is derived from simulated flight envelopes and stored in the non-volatile memory of the subsystem terminals. The results of the application software computation are compared with those similarly derived from the simulation.

A simulated platform and profile generator, and simulated measurements implemented in the TERPAC-software are very helpful in checking out the kalman-filter-equations.

Fig. 9 shows the results of a simulated straight and level flight over eight update areas. The position error of the unaided inertial navigation system of low accuracy rapidly increases as a function of time. With kalman filtered TERPAC updates of differing statistical uncertainty the error behaviour of the combined navigation system is reduced dramatically. It is obvious that the resulting navigation accuracy will be independent of time. Updates with greater uncertainty are pointed out in the figure.

6. Flight test experiments

Extensive optical tracking test series have been conducted in order to test the behaviour of the sensor system measuring the radar altitude and the altitude a ove datum plane (fig. 10). A knowledge of these characteristics is important for valuation of TERPAC-accuracy.

The aircraft is flown along a predetermined datum line and tracked by two cine-theodolites which record elevation and azimuth angles simultaneously.

By using the presurveyed distances from the datum line the X, Y, Z - coordinates of the aircraft and the velocity components can be calculated off line.

If the aircraft is out of the tracking range of the cine theodolites, the flight envelope coordinates of the aircraft are calculated from the measurement outputs of a radar tracking device (refer to fig 11). The flight test data and radar signals are simultaneously transmitted via seperate data links and stored on a magnetic tape in the ground station for post flight analysis.

Most TERPAC-avionic-components are installed in a modified pod of the experimental aircraft.

The aim of the experimental program is to test the online processing of the TERPAC system and the influence of terrain contours on the accuracy of TERPAC updates.

7. Conclusions

The requirements concerning the high navigation accuracy in missile guidance and low level flight with manned aircraft can be fulfilled by the discussed TERPAC-avionic-system. A dedicated ground based data management system has been developed to perform test flights and to check out the realtime hardware and software system. The effects of the real terrain and of the dynamic flight behaviour of the aircraft will be tested in a current experimental program.

METHOD

- TERPAC -processor matches measured terrain -signatures against terrain–reference–signatures stored in the mass memory

MODES OF ACTION

- inertial mode
- TERCOM–position–fix–mode
- SZEKOR–position–fix–mode
- terrain–following/avoidance–continuous mode

CHARACTERISTICS

- automatic
- autonomous
- all–weather -, night capability
- high anti jam resistance, low detectibility

Fig.1 TERPAC (TERRAIN – PARAMETER – COMPARISON)

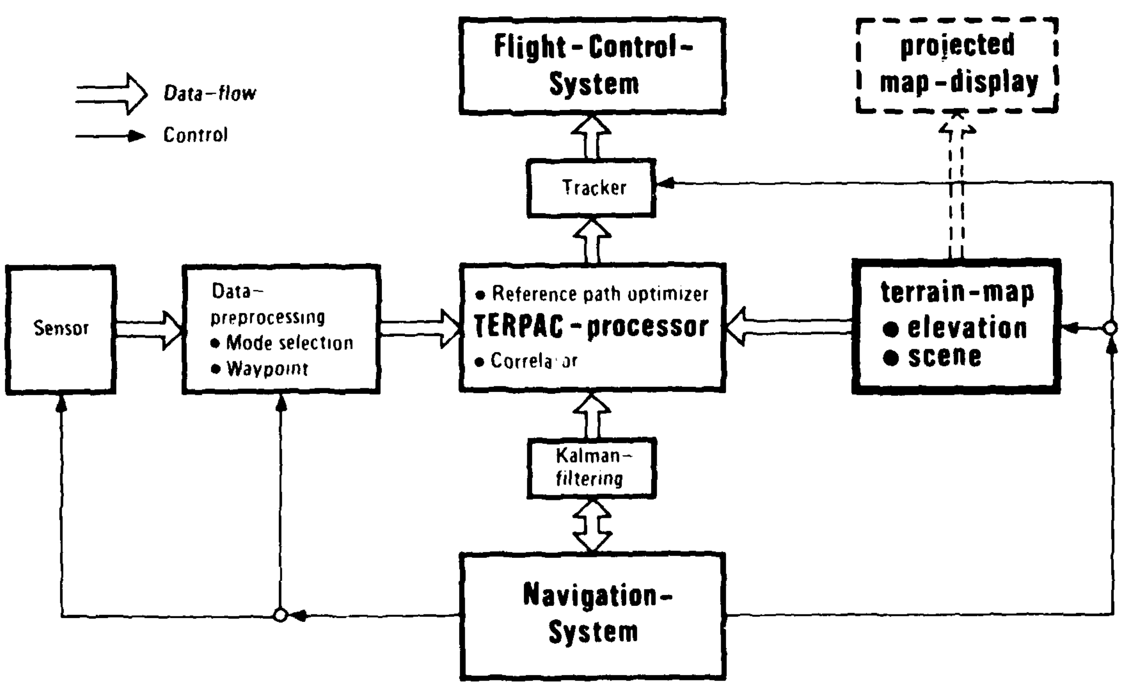


Fig. 2 NAVIGATION-/FLIGHT CONTROL SYSTEM

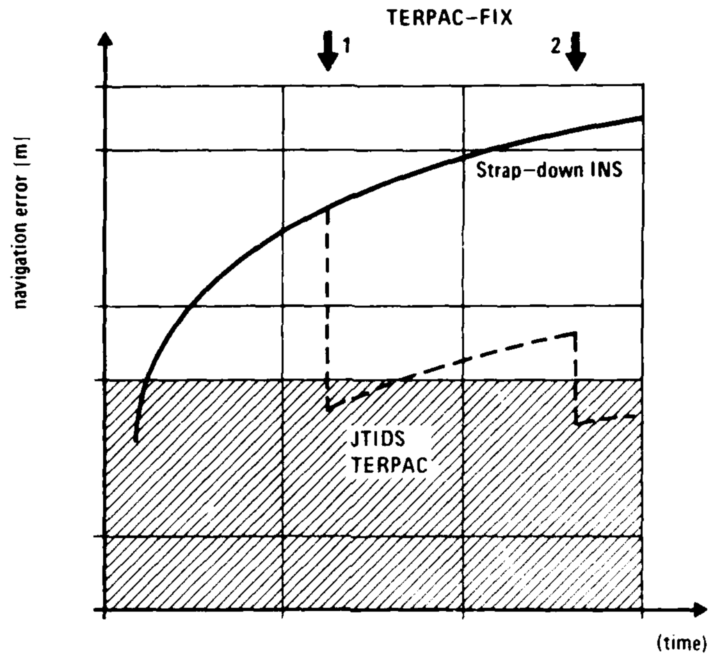


Fig. 3 POSITION-FIX-ACCURACY

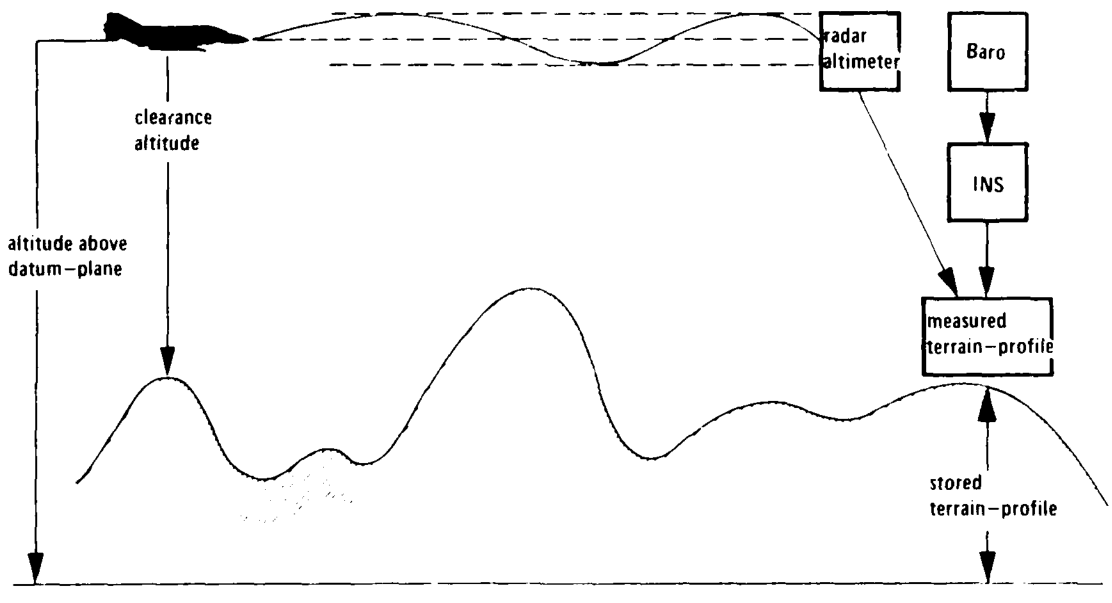


Fig. 4 TERRAIN-PROFILE-MEASUREMENT

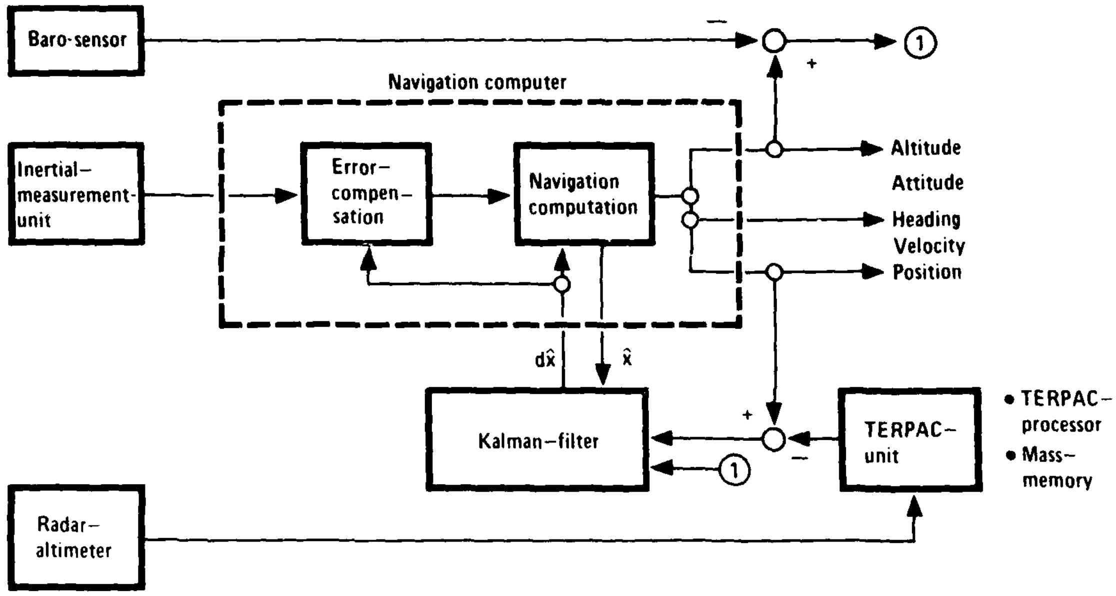


Fig. 5 NAVIGATION-SYSTEM: DATA FLOW

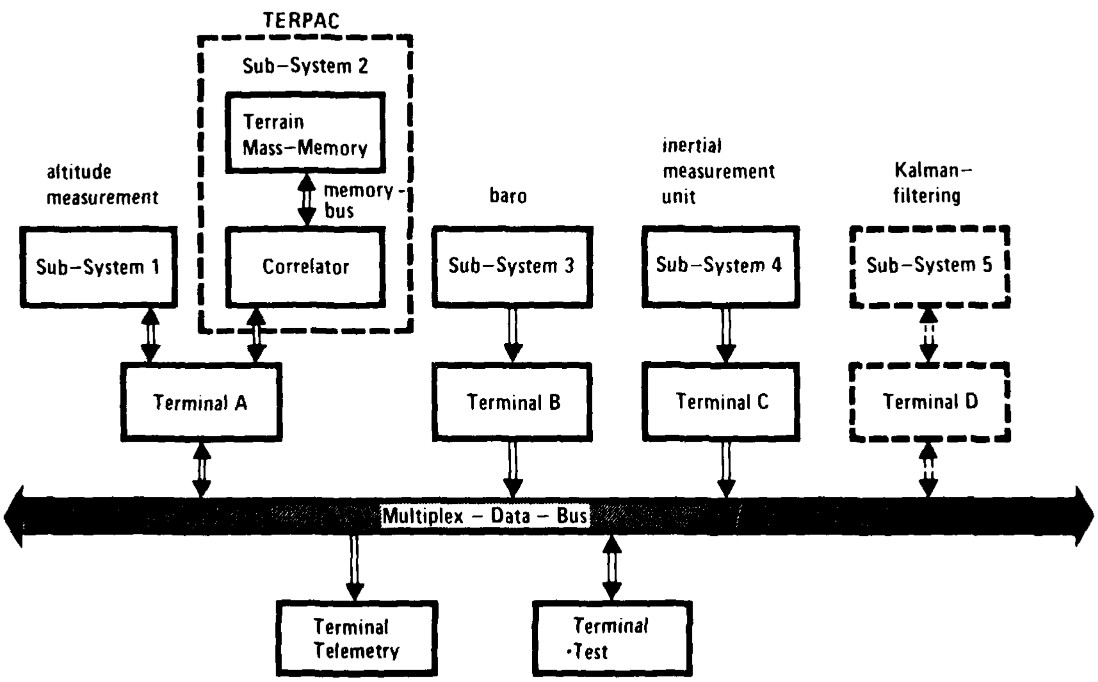


Fig. 6 TERPAC-AIDED-INS: DISTRIBUTED MICROPROCESSOR ARCHITECTURE

- AUTONOMOUS SUBSYSTEMS
- PARALLEL COMPUTATION
- GREAT MODULARITY
- INDEPENDENT SOFTWARE–TASKS
- REDUCED DATA–RATES
- UNCRITICAL ASYNCHRONOUS TIMING
- SIMPLE I/O–INTERFACES
- STANDARD MICROPROCESSOR COMPONENTS
- **SIMPLE SYSTEM–TESTING**

Fig. 7 DISTRIBUTED MICROPROCESSOR ARCHITECTURE (ADVANTAGES)

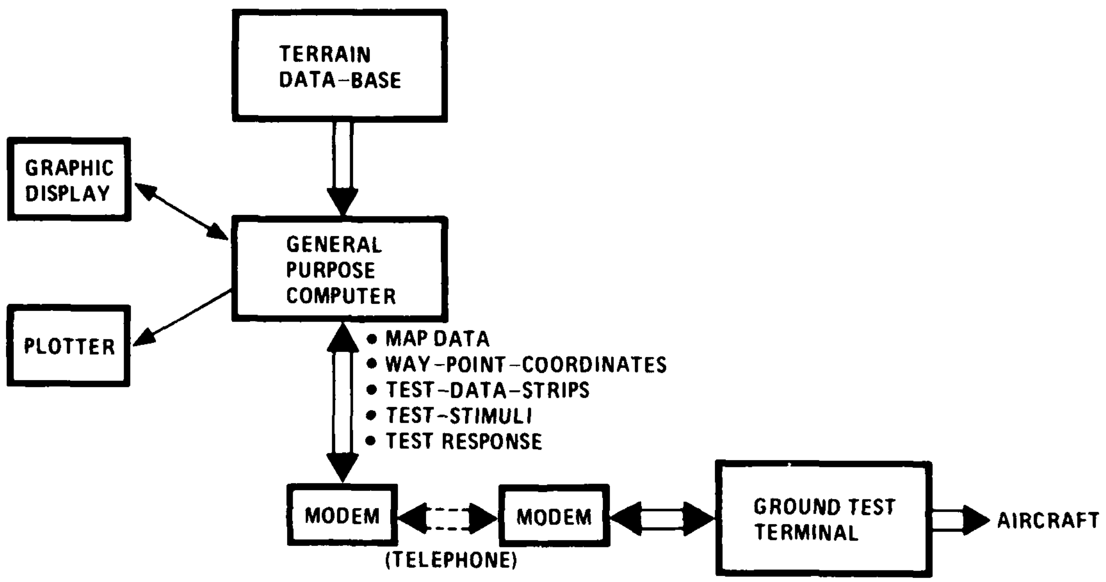


Fig. 8 GROUND BASED DATA MANAGEMENT SYSTEM

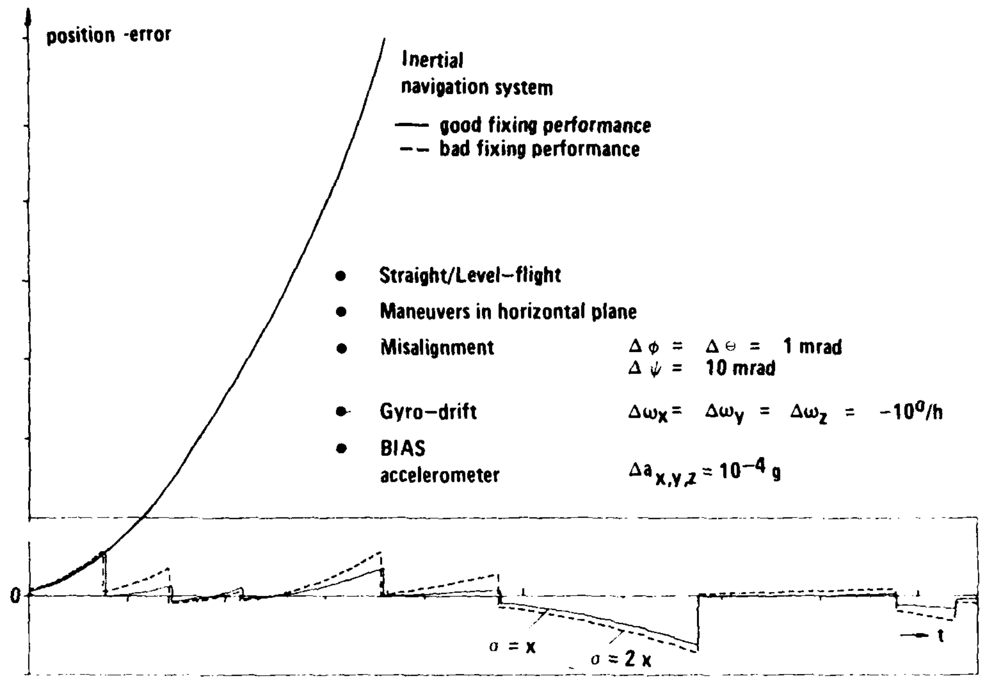


Fig. 9 TERPAC-FIX-ACCURACY (SIMULATION)

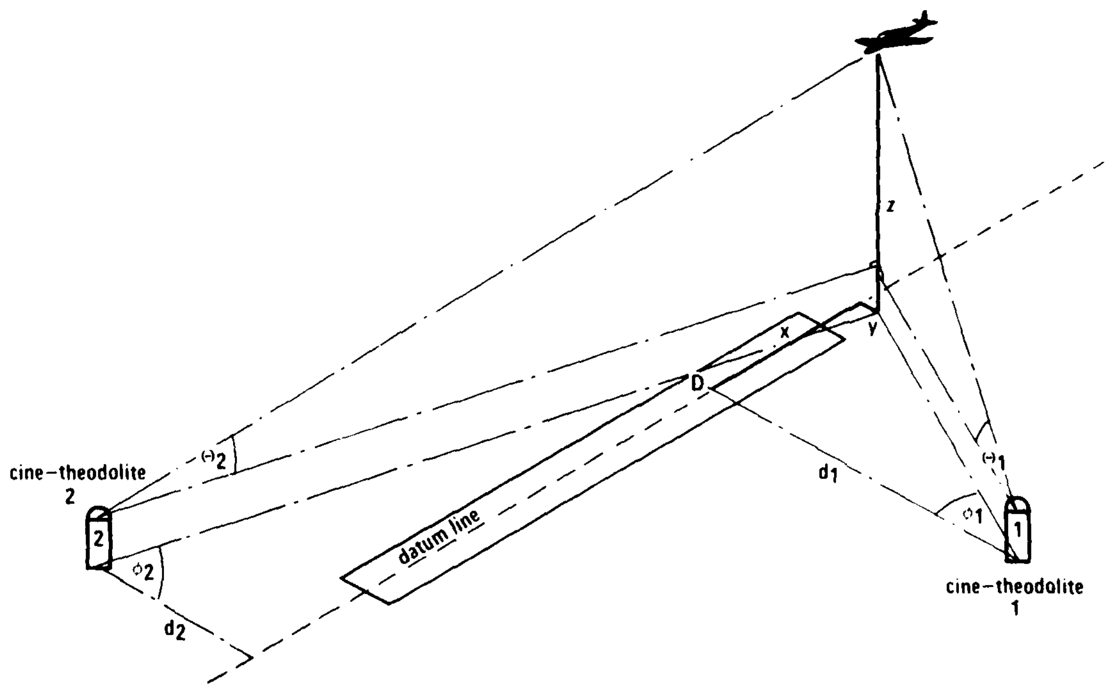
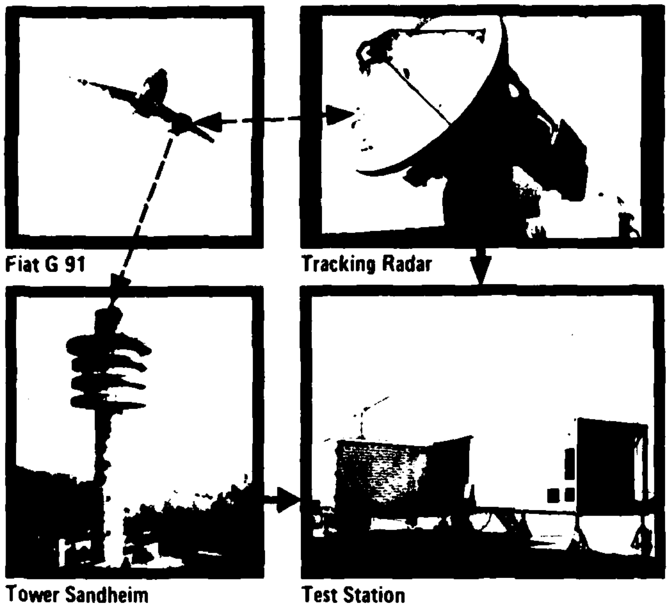


Fig. 10 ALTIMETER CALIBRATION BY OPTICAL TRACKING MEASUREMENT



- MEASUREMENT OF FLIGHT-ENVELOPE

- FLIGHT-TEST-DATA
  - Preprocessing
  - Storage

Fig. 11 EXPERIMENTAL PROGRAM

MISE AU POINT DU SYSTEME DE NAVIGATION ET D'ATTAQUE DE L'AVION ALPHA-JET

par

JÜRGEN LANG
Ingénieur Navigant d'Essais
DORNIER GmbH
8031 Oberpfaffenhofen
ALLEMAGNE

RESUME

On procède à une description du système de navigation et d'attaque, de son but et de ses principes de fonctionnement. L'installation d'essais specifique est evoquée, qui assure la "souplesse" souhaitable pour les essais en vol.
Les tâches principales de la mise au point ainsi que les différentes méthodes de travail au sol et en vol seront décrites. Pour conclure une comparaison sera faite entre les valeurs pré-calculées de précision de tir et de navigation d'une part et les valeurs obtenues en vol, d'autre part.

1. DESCRIPTION DU SYSTEME

1.1 But du système

La version allemande de l'avion Alpha-Jet est conçue pour des missions d'appui tactique léger, et de ce fait elle diffère de la version française, dite "Ecole", essentiellement par son système de navigation et d'attaque. Ce système est destiné à attaquer des objectifs au sol dans des conditions où le pilote voit l'objectif.
En outre, l'avion allemand dispose du mode d'attaque AIR/AIR avec le canon. Le SNA doit assurer la conduite de tir simple et précise ainsi que la navigation autonome.

1.2 Principe de fonctionnement de la conduite de tir

L'élément de base est constitué par le viseur conçu en tant qu'équipement tête haute (HUD) alimenté par un calculateur d'armes et un générateur de symboles.
Un avantage notoire que présente un HUD est le fait que, durant toute la phase d'attaque proprement dite, il permet au pilote de fixer l'objectif, étant donné que toutes les informations nécessaires -tant celles relatives au pilotage que celles relatives à la précision de la conduite de tir- lui sont présentées dans son champ de visée, et cela sensiblement à la distance à laquelle il voit son objectif. Suite à ces conditions, [illegible] des yeux du pilote restera constante, ce qui est un fait décisif si l'on [illegible] le temps disponible entre l'acquisition de l'objectif et le déclenchement de l'arme (~ 5 s). Grâce à tout cela, le pilote n'est plus obligé d'estimer ou de [illegible] les différentes grandeurs (telles que portée des armes, vitesse, altitude [illegible] direction du vent etc...). Le réticule de tir est dirigé sur l'objectif, et [illegible] l'arme est déclenchée manuellement (queue de détente ou poussoir [illegible] manche).

[illegible] selon le principe CCPI (Calcul continu du point d'impact): Le point [illegible] arme sur une surface horizontale est calculé en permanence, puis [illegible] fréquence 50 Hz).

[illegible] Alpha-Jet ce n'est pas une mesure directe de la distance oblique entre [illegible] point d'impact qui est effectuée -cette méthode étant normalement appliquée [illegible] systèmes plus complexes utilisant p.ex. le LASER ou le radar- mais le calcul [illegible]:
[illegible] est mesurée par un radioaltimètre qui détermine ainsi la surface horizon[illegible] le point d'impact. C'est la centrale gyroscopique de cap et de verticale [illegible] information d'assiette longitudinale, ce qui permet, en utilisant également [illegible] mesurées des conditions de vol (telles que vitesse, angle d'incidence [illegible] les valeurs mémorisées de la ballistique d'armes, de calculer la distance [illegible] par conséquent la hausse du point d'impact.
[illegible] on découvre les points critiques du système en ce qui concerne la précision de tir: pas de centrale inertielle, pas de dispositif de mesure directe de la distance oblique. D'ailleurs, les erreurs les plus décisives découlent de la centrale gyroscopique et du radioaltimètre.

Il faut ajouter à cela que ce principe n'est apte à fonctionner que sur un terrain plat. C'est pour cette raison que l'Alpha-Jet dispose, en dehors du mode RDR pur, de deux modes supplémentaires appliquables aussi sur terrain accidenté: c'est tout d'abord un mode mixte appelé mode RDR/BAR, pour lequel il faut connaître la dénivellée ΔH de l'objectif par rapport à un plan proche, cette dénivellée étant prise en mémoire dans le calculateur d'armes. Avant l'attaque, le pilote doit survoler ce plan afin de recaler l'altitude: La valeur de l'altitude radar est mémorisée et utilisée ensemble avec ΔH et les variations d'altitude barométriques qui suivent, pour calculer en permanence, pendant le tir, l'altitude de vol par rapport à l'objectif et ainsi la hausse du réticule.

Le troisième mode, quant à lui, travaille sans radioaltimètre: c'est de ce fait un mode purement barométrique. Les conditions préalables liées à ce mode sont la connaissance de la pression atmosphérique momentanée (QNH) dans la zone d'attaque ainsi que de la hauteur géographique de l'objectif, cette dernière étant mémorisée dans le calculateur d'armes.

Les différents modes opérationnels possibles sont les suivants:

- .Bombardement avec bombes lisses et freinées
- .Tir de roquettes - non guidées
- .Tir canon - AIR/SOL et AIR/AIR
- .Navigation

Pour chaque type d'arme, il est possible de mémoriser 10 ballistiques, ce qui permet d'utiliser p.ex. 10 types de bombe différents.

Enfin, c'est le Doppler et le calculateur de navigation associé qui constituent des composants fondamentaux du système. Ces derniers ont une double fonction: ils assurent la navigation autonome ainsi que le soutien du système de conduite de tir en mode AIR/SOL. Les données du vent en tant que vitesse et direction constituent une information importante pour la conduite de tir. A partir de la vitesse air vraie (TAS), délivrée par la partie aérodynamique du calculateur d'armes, ainsi que de la vitesse sol délivrée par le capteur de vitesse Doppler, le calculateur de navigation détermine le vecteur du vent.

1.3 Principe de la navigation

Dans ce contexte le HUD présente l'avantage suivant: il évite au pilote navigant à basse altitude d'être obligé de discerner ses informations de pilotage dans l'obscurité de la cabine, car ces dernières lui sont projetées en superposition au paysage extérieur. Le calculateur de navigation détermine la distance parcourue par intégration de la composante horizontale de la vitesse sol Doppler. Or, la position présente de l'avion est obtenue en additionnant vectoriellement la distance parcourue à la position de départ de l'avion. La déclinaison magnétique est mémorisée comme polynôme dans le calculateur. Du fait que le calcul de la position présente s'effectue toutes les 50 ms, on a la garantie que les moindres changements de la déclinaison magnétique soient pris en considération. Le calculateur de navigation est également en mesure de calculer la position présente à partir des données fournies par une station TACAN, et de l'indiquer au pilote. Ainsi, le pilote dispose de moyens de comparaison.

En somme, visualisations et commandes du système Doppler de l'avion Alpha-Jet sont pratiquement identiques à ceux d'un système inertiel. Sur le poste de commande de navigation (PCN) le pilote peut mémoriser et sélectionner 10 différents buts de navigation et 4 stations TACAN. Il peut aussi mettre le Doppler sur Standby.
Le PCN pourra fournir au pilote, sous forme numérique, les informations suivantes: soit la force et la direction du vent, soit la vitesse sol et l'angle de dérive, soit les coordonnées de position de l'avion ou bien d'un but de navigation, soit la distance et le relèvement d'un but de navigation, ces deux dernières informations étant également présentées sur le BDHI.

En mode Navigation, le HUD présente au pilote les informations suivantes: cap, relèvement, distance et temps de vol à la station concernée, dérive due au vent de travers, symbole TO/FROM, vitesse sol ou vitesse air, horizon, pente et vitesse verticale.
En comparant la position présente Doppler à une position présente TACAN, ou bien lors du passage à la verticale de points de navigation prédéterminés, le pilote est en mesure de corriger le calcul de navigation. En outre, il est possible de mémoriser en vol par simple pression d'un bouton les coordonnées de 5 buts d'opportunité au maximum. La navigation peut être effectuée ou en coordonnées géographique ou en coordonnées UTM; la commutation est possible à tout moment (Photo 1).

2. INSTALLATION D'ESSAIS

L'enregistrement des valeurs mesurées se fait comme d'habitude par télémesure et sur enregistreur magnétique embarqué. 3 commutateurs PCM et 1 commutateur PAM à 64 voies chacun sont utilisés.

Les aspects particuliers de l'installation de mesure sont liés aux unités suivantes:

- Une centrale à inertie servant de référence pour contrôler
  - .la centrale gyroscopique
  - .la sonde Doppler
  - .le boîtier gyrométrique.
- Le PMS, qui est le système de mémorisation programmable du calculateur HUD. Il s'agit là d'un matériel expérimental optionnel permettant d'introduire, au sol ou en vol, à l'aide d'un lecteur de cassettes, le programme destiné au calculateur d'armes et au générateur de symboles. La bande de la cassette peut être enregistrée et vérifiée en quelques minutes sur le lieu des essais. Dans le cadre des essais en vol cet équipement s'est avéré très efficace pour des changements rapides de la symbologie ou du calcul d'armes.

La caméra de visée fait également partie de l'installation d'essais, étant donné qu'elle permet de mesurer l'erreur de visée au moment du déclenchement de l'arme, d'une part, et de figer la symbologie telle qu'elle se présente au pilote dans sa suite chronologique, d'autre part.

3. ESSAIS A EFFECTUER

Quelles sont les tâches qui se présentent dans le cadre d'une telle expérimentation? Il est possible de définir des tâches partielles qui en fait ne sont pas assurées successivement mais parallèlement.

- Il s'agit de déterminer les conditions d'environnement des équipements installés tant dans le poste pilote que dans les différents compartiments fuselage.
  Cela concerne essentiellement
  . les températures,
  . les vibrations et oscillations (lors du buffeting, tir canon),
  . la compatibilité électromagnétique.

  Ces mesures doivent assurer
  - premièrement que les conditions d'environnement spécifiées par l'avionneur pour les différents lieux de montage soient respectées par lui-même,
  - deuxièmement que les équipements fonctionnent correctement dans leurs lieux de montage et dans le domaine défini par ces spécifications.

- Un grand soin doit être porté à la qualité des grandeurs d'entré qui sont fournies par les sondes et qui entrent dans le système. Ceci concerne la précision et la pureté des signaux.
  Pour une bonne partie il s'agit de valeurs d'étalonnage, concernant par exemple
  . altitude et vitesse,
  . angle d'incidence, angle de dérapage, température d'impact,

  ainsi que des signaux provenant de
  . centrale gyroscopique,
  . radioaltimètre,
  . Doppler,
  . boîtier gyrométrique.

  Dans la mesure du possible, les erreurs déterminées sont mémorisées dans le calculateur d'armes comme valeurs correctives. Ce procédé permet non seulement une augmentation de la précision mais également la cohérence des valeurs indiquées, car les paramètres
  . vitesse,
  . altitude,
  . nombre de Mach,
  . incidence,
  . radioaltitude,
  . cap

  sont présentés deux fois au pilote du poste AV: d'une part dans le HUD, et d'autre part en tant qu'indication classique du type tête basse.

- Infrastructure du système

  Dans ce contexte, il faut surtout parler de la bonne disposition des éléments de visualisation et de commande au poste pilote. Il faut veiller à ce que ces instruments et organes soient logiquement conçus et convenablement montés. Il ne faut pas ignorer que l'agencement de ce matériel ne peut se faire que sur avion en vol, ou lors de maquettages, à la rigueur, mais jamais en réunion au tapis vert. Ignorer ce fait, c'est encourir le risque de perdre plusieurs années au niveau des essais.
  Dans ce paragraphe, il faut également citer le développement en détail du logiciel pour ce qui concerne
  . les vérifications automatiques du logiciel,
  . les modes de secours en cas de pannes d'équipements et les indications de telles pannes,
  . l'acquisition des caractéristiques des armes de série,
  . la précision des calculs aérodynamiques.

- Symbologie HUD

  Les symboles présentés dans le viseur sont à contrôler quant à la luminosité, au contraste, à la précision, et par rapport à l'importance du champ de visée, ainsi quant à l'information elle-même et son utilité et compréhension pour le pilote.

  Nature et forme des symboles - ceci est un vaste sujet, et qui est loin de faire l'unanimité des pilotes. L'expérience européenne en matière de HUD électroniques était pratiquement nulle au début des essais Alpha-Jet. Et l'opinion d'un pilote quant à la symbologie évolue avec ses expériences acquises, ceci dans le sens qu'il peut "supporter" plus d'informations après un certain nombre d'heures de vol effectuées.

  Grâce au HUD programmable, nous avons pu modifier pratiquement toute la symbologie en 6 mois: Pour l'essentiel, nous sommes passés d'une symbologie pseudo-analogique à une symbologie partiellement numérique (Photos 11 et 12).
  La symbologie version série ne correspond plus ni à la spécification MIL applicable (884 C), ni au STANAG. Ces derniers ont probablement été figés eux aussi dans un contexte purement théorique, à une époque à laquelle personne ne possédait l'expérience nécessaire.

- Intégration du système et contrôle de sa précision

  Le fonctionnement correct de l'ensemble des composants du système entre eux, ainsi que la précision de fonctionnement du système au tir et à la navigation ont été acquis en plusieurs étapes, tant aux banc d'essais que durant des vols d'essais. On verra dans la suite comment nos travaux se sont déroulés.

4. METHODES D'ESSAIS

Pour des raisons de disponibilité du matériel, le système a été essayé en 2 étapes:

Stade I : Sans Doppler, ni calculateur de navigation. La force et la direction du vent sont communiquées au pilote avant la passe de tir et c'est le pilote qui affiche ces valeurs sur l'instrument correspondant au poste pilote. Dans cette phase nos activités étaient concentrées sur la symbologie HUD et la précision de tir.

Stade II : Avec Doppler et calculateur de navigation. Grâce à ce matériel le vecteur vent peut être calculé continuellement par le système lui-même. Dans cette phase nos activités portaient sur la précision de la navigation et sur le développement du boîtier de commande pour le calculateur de navigation.

Le système a été soumis à toute une série de tests comme je vous les exposerai dans la suite; en général, les essais ont été effectués dans l'ordre exposé.

- Simulation

  Ces opérations sont effectuées sur un banc d'essais sur lequel le câblage d'origine de l'avion est reconstitué et qui est équipé de tous les composants et capteurs de l'avion. La précision du système est contrôlée en statique et par points de mesure. Des conditions de vol stabilisées sont simulées!

- Stimulation

  Les paramètres enregistrés lors de manoeuvres de vol et les valeurs correspondantes mesurées par les sondes sont introduits dans le système. En procédant de la sorte, il est possible de contrôler la précision en dynamique ("stimulation"). Par comparaison avec les valeurs constatées en statique, on choisit les constantes de temps pour les filtres nécessaires. En outre, cela permet d'obtenir au préalable une idée de ce qui sera présenté au pilote durant la phase d'attaque.

- Passes de tir fictives

  Elles constituent l'étape suivante et sont effectuées avec l'avion. On surveille le calculateur d'armes en enregistrant ses signaux d'entrée et de sortie. Simultanément ce même calcul est fait au sol par un ordinateur dans lequel la valeur d'un paramètre déterminé est remplacée par une valeur de mesure améliorée et plus précise de ce même paramètre. Exemple: L'assiette longitudinale de la centrale gyroscopique est remplacée par l'assiette longitudinale de la centrale à inertie montée sur l'avion comme installation d'essais. L'influence qu'a cette mesure sur la hausse du réticule de visée est ainsi démontrée de façon claire et nette, et par là même l'influence qu'elle a sur la précision de tir. Le procédé est le même pour l'altitude radio-sonde, par exemple. Dans ce cas, la valeur de mesure de remplacement est celle de l'altitude cinéthéodolite. Dès que toutes les erreurs discernables semblent connues, on passe à la phase suivante.

- Passes de tir réelles

  Leur bien-fondé est de démontrer la précision globale du système, si possible de façon statistique, mais sans pour autant devoir déployer des moyens trop importants. En même temps elles permettent éventuellement de mettre en évidence des erreurs systématiques residuelles, cela lorsque le point moyen de tous les impacts n'est pas identique avec le centre de la cible.
  En plus nous avons trouvé que la pilotabilité de la symbologie ne peut être réellement testée que lorsqu'on demande au pilote d'obtenir les meilleurs résultats de tir.

  Les armes utilisées ont été:
  . une bombette lisse dont la ballistique est bien connue,
  . le canon embarqué prévu pour la série.

  Pour les tirs de bombes, la cible visée est la croix centrale d'un cercle de 200 m de diamètre placé sur le champ de tir de Cazaux (France). Jusqu'à présent environ 70 bombes ont été larguées avant, à peu près autant après installation du Doppler. Un tiers de ces bombes ont été larguées lors de vols de tir effectués par les Services Officiels.

  Les tirs canon se font sur un cône flottant sur une piscine mesurant 110 m de long sur 60 m de large, ou sur des filets de 5 m x 5 m dressés à la verticale et marqués d'une croix orange. Une seule salve de 0,5 s environ à 10 à 20 obus est effectuée sur chaque cible. Jusqu'à présent environ 3000 obus d'exercice ont été tirés en 200 rafales. Le but du tir sur piscine est de déterminer la précision du premier obus tiré et le déroulement approximatif des impacts suivants de la salve. Le but du tir sur filet est de relever le nombre de scores, tout en veillant à éliminer tous les ricochets.

Pour les passes de tir nous utilisons les profils de vol pratiqués par la LUFTWAFFE, à savoir: profil d'entraînement avec hippodrome et profil pop-up.

- Les vols de navigation sont essentiellement des vols triangulaires, chacun des 3 tronçons correspondant à plus de 100 NM que ce soit au-dessus de la terre ou de la mer, avec des altitudes et des vitesses différentes.
  Aux différents coins du triangle de navigation les données du système sont comparées avec celles de la plateforme inertielle et avec les points déterminés du terrain survolé. Un recalage des erreurs de navigation n'est pas effectué ceci afin d'éviter que l'imprécision due au passage à la verticale d'un point n'engendre d'autres erreurs.

5. RESULTATS DES ESSAIS

- Symbologie du HUD

  Les photos 2 à 5 montrent des cas typiques de représentation lors d'une attaque avec roquettes ou canon. Le symbole caractéristique de ces modes est le marqueur de domaine représenté sous forme d'un cercle 3/4 autour du reticule de tir (Pipper).
  Ce marqueur domaine apparaît à la distance maximale de tir, se ronge continuellement dans la mesure où l'avion se rapproche de l'objectif, et disparaît lorsque la distance minimale de tir est atteinte, c'est-à-dire au moment ou la croix de dégagement commence à clignoter.

  Les photos 6 à 10 montrent le mode bombes caractérisé par la ligne de chute des bombes (BFL). Il s'agit là d'une aide au pilotage pour le pilote. Il est plus facile pour lui de viser la cible s'il est en mesure de placer le plus tôt possible cette BFL sur la cible. Ainsi il s'assure toujours une marge de pilotabilité assez grande, et il a la certitude de pouvoir finalement placer le Pipper sur la cible. De toute façon, en mode "bombes", le pilote laisse systématiquement passer le Pipper sur la cible -grâce à la BFL-, pour déclencher la bombe au moment de la coincidence Pipper/cible. Par contre, à l'attaque avec canon ou roquettes, le Pipper peut être en principe maintenu sur la cible tant que l'avion est en passe de tir. Cela peut être mieux illustré moyennant un film de tir pris par l'enregistreur de visée HUD.

  On peut ajouter que manifestement le HUD de l'Alpha-Jet est une très bonne acquisition. Tous les pilotes -y compris ceux des Services Officiels et ceux de la LUFTWAFFE- ayant eu jusqu'à présent l'occasion de voler avec le HUD en sont enthousiasmés. Bien que ce viseur ait une optique dont le diamètre de 4 pouces ne correspond qu'à la moitié de celle du viseur de l'avion F-14, à titre d'exemple, il présente un champ de visée bien supérieur en raison de la géométrie de la cabine.

- Précision de tir

  L'exploitation des résultats de tir est normalement effectuée en CEP (Circular Error Probability) correspondant au rayon d'un cercle autour du point central de la cible dans lequel se trouvent 50% de tous les impacts selon la probabilité.

  Quelle est la précision du système selon les valeurs exploitées comparativement aux valeurs pré-calculées?

  a) Pour les bombes (bombettes d'exercice lisses) la précision pré-calculée correspondait à un CEP de 15-23 mrd selon l'intensité du vent (sans Doppler et avec introduction manuelle du vent). Lors des essais en vol le résultat constaté a été de 10,4 mrd; après correction de l'erreur de visée du pilote il a été de 9,7 mrd, c'est-à-dire que les visées ont été effectuées avec une très grande précision. La différence entre les 2 valeurs est négligeable. Dans le cas d'une distance de tir de 2000 m 50% des bombes se trouvent donc dans un cercle avec un rayon de 20 m autour de l'objectif.

  b) En ce qui concerne le canon certaines dispersions supplémentaires ont été constatées, étant donné que les essais ont été effectués avec des canons prototypes.
  Malgré cela les résultats constatés sont également meilleurs que les valeurs pré-calculées variant, en fonction de l'intensité du vent, entre 4 et 5 mrd.
  Les essais sans Doppler ont abouti à un CEP de 2,5 mrd, sans correction de l'erreur de visée, et à un CEP d'environ 1,0 mrd après correction de l'erreur de visée. Autrement dit, dans le cas d'une salve effectuée sur une cible de 5 x 5 m à une distance de tir de 1000 m 50% des impacts touchent la cible.

  c) Avec Doppler, en modes bombes et canon, la précision de tir a été la même que sans Doppler. Ainsi la méthode non opérationnelle constistant à mesurer le vent juste avant le moment du tir à l'aide d'un ballon météorologique est remplacée par la mesure permanente du vent à l'aide du Doppler embarqué, ceci sans perte de précision.
  (Filtrage du vent Doppler avec une constante de temps de 2,7 sec. et du signal angle de dérapage avec une constante de temps de 2,0 sec.)

  En résumé, les résultats d'essais tant en mode "bombes" qu'en mode "canon" ont été comme suit:

  1) Les précisions de tir espérées ont été obtenues.
  2) Il n'y a plus d'erreur systématique dans le système de conduite de tir.
  3) Les résultats obtenus avec le mode RDR/BAR, qui est d'ailleurs plus représentatif de l'utilisation opérationnelle, sont tout aussi satisfaisants qu'en mode purement RDR.

- Précision de navigation avec Doppler

a) sur terre
Le calcul préliminaire de l'erreur de position en tant que valeur CEP en fonction de la distance parcourue avait donné:

.en configuration "sans charges extérieures": 0,93% de la distance parcourue (ou 3,6NM/h)
.en configuration "avec charges extérieures": 1,3% de la distance (ou 5 NM/h).

Ce même résultat fut obtenu au cours des essais en vol; ainsi à titre d'exemple, une valeur CEP de 0,96% fut atteinte lors de vols de navigation sur la terre (en configuration "sans charges extérieures", 32 tronçons de navigation, $\sigma_x = \pm 0,49\%$, $\sigma_y = \pm 1,16\%$).

Il a été constaté que l'erreur de cap résultant de la plateforme gyroscopique est l'erreur principale du système de navigation, et qui, dans la valeur CEP, est prise en compte en tant qu'erreur de navigation transversale.

b) sur mer
D'une part, le système de navigation Doppler nécessite une surface d'eau suffisamment mouvementée, afin qu'un pouvoir réflecteur suffisant (pour l'énergie électro-magnétique) soit donné. Sinon, le Doppler passe en mémoire, et la navigation se poursuit avec la dernière valeur du vent valable. L'erreur de navigation qui en résulte dépend de la durée de cette condition et des variations du vent.
D'autre part, le vent engendre un courant au niveau de la surface de l'eau qui a pour résultat une erreur sur la mesure de la vitesse, étant donné que la vitesse relative entre l'avion et la terre est faussée.

Lors de quelques vols et en fonction des conditions rencontrées, on a obtenu le résultat suivant:

$$CEP = 1,46\% \qquad \left(\begin{array}{l} \sigma_x = \pm 1,4\% \\ \sigma_y = \pm 1,1\% \end{array} \right)$$

Par une estimation du courant à la surface de l'eau avec un vent connu, il est possible de démontrer que le Doppler est tout aussi précis au-dessus de l'eau qu'au-dessus de la terre.

6. CONCLUSIONS

Les études théoriques en vue d'améliorer le système de navigation et d'attaque furent lancées en automne 1975. Un prototype (03) a été destiné aux essais en vol correspondants qui ont eu lieu en deux étapes:

- Le premier stade sans Doppler. Cette mise au point s'est étendue sur une période de 11 mois en 1977 et a nécessité environ 100 vols.
- Pour le deuxième stade avec Doppler la part essentielle des travaux était consacrée aux essais de navigation, qui ont eu lieu entre le mois de mai 1978 et le mois de mars 1979 et pour lesquels environ 100 vols ont été également nécessaires
- Depuis l'été 1979 jusqu'à maintenant les efforts se sont portés vers une prise en compte dans le système final des dernières demandes exprimées par la Luftwaffe relatives à l'utilisation tactique et opérationnelle.
- Les performances théoriques ont pu être démontrées, voire dépassées.
- Ainsi donc, le système a atteint son stade opérationnel et apte pour l'industrialisation.

Pour conclure on peut constater que ces essais ont permis pour la première fois à l'industrie allemande de faire des expériences appréciables et nécessaires dans ce domaine.

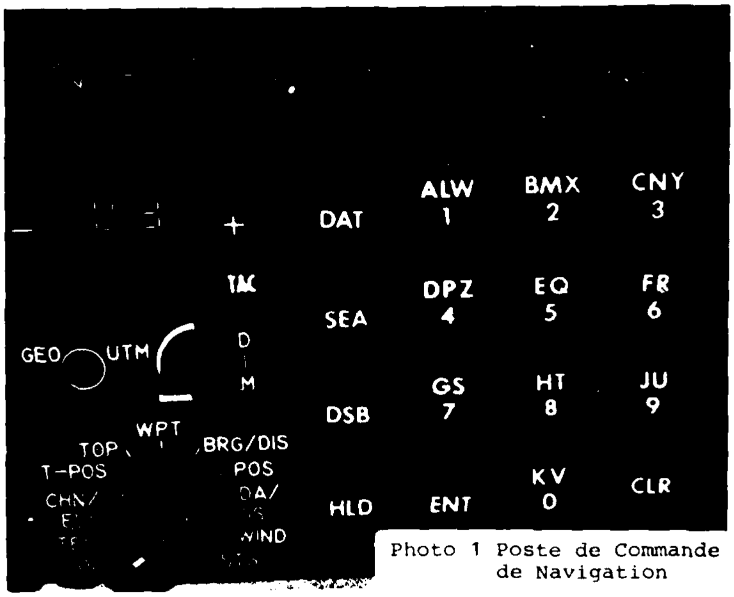


Photo 1 Poste de Commande de Navigation

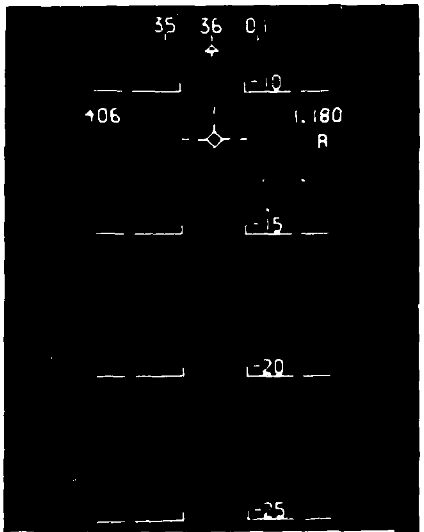


Photo 2 Mode CANON/ROQUETTES Hors domaine de tir

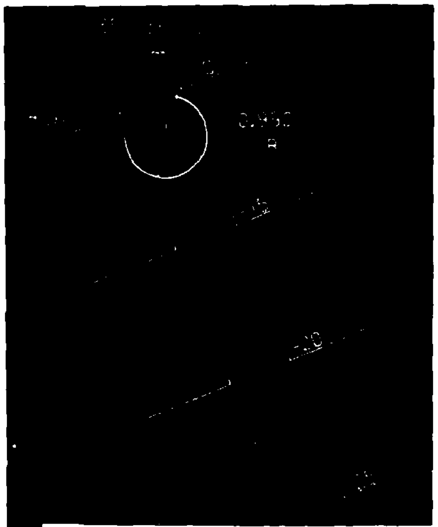


Photo 3 Mode CANON/ROQUETTES Début domaine de tir

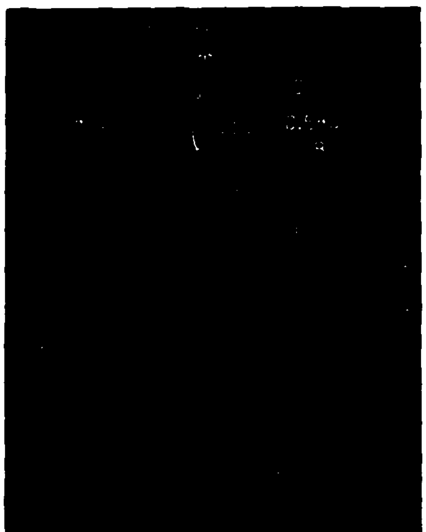


Photo 4 Mode CANON/ROQUETTES Fin domaine de tir

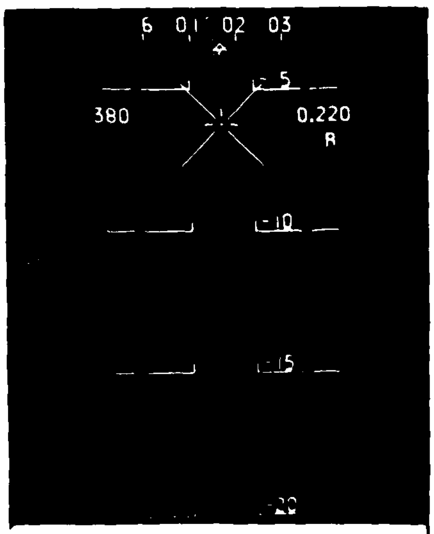


Photo 5 Mode CANON/ROQUETTES avec croix de sécurité

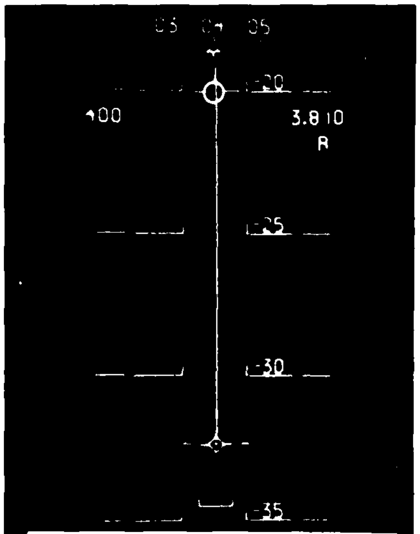


Photo 6 Mode BOMBES
Point moyen de tir

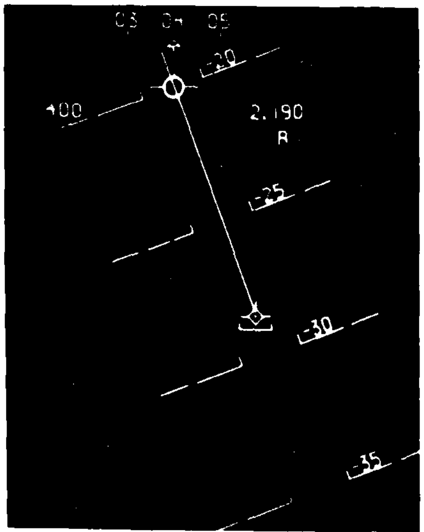


Photo 7 Mode BOMBES
Point optimal de tir

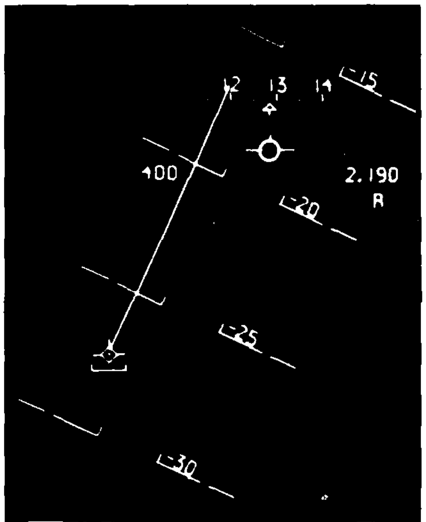


Photo 8 Mode BOMBES
avec vent latéral

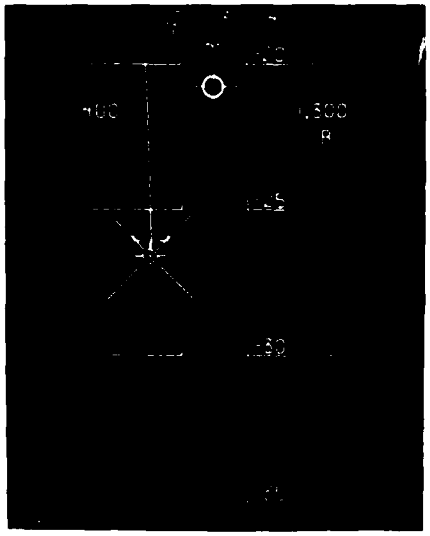


Photo 9 Mode BOMBES
avec croix de securité

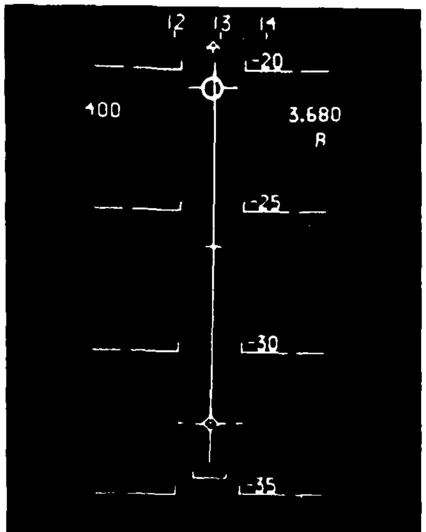


Photo 10 Mode BOMBES
tir en salve

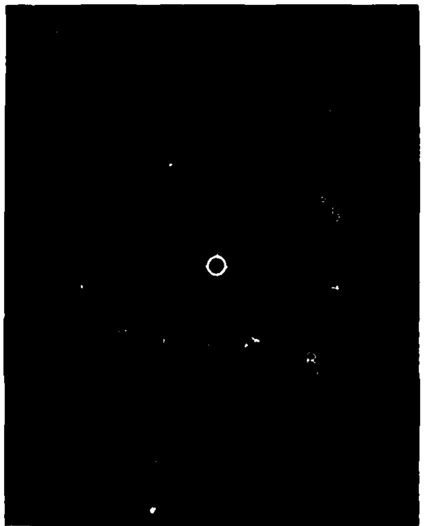


Photo 11 Mode NAVIGATION
symbologie STANAG

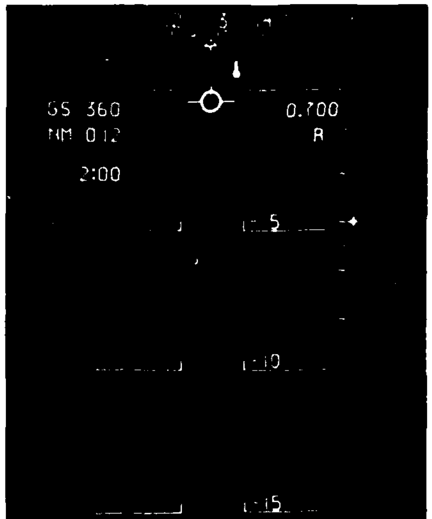


Photo 12 Mode NAVIGATION
écart de route

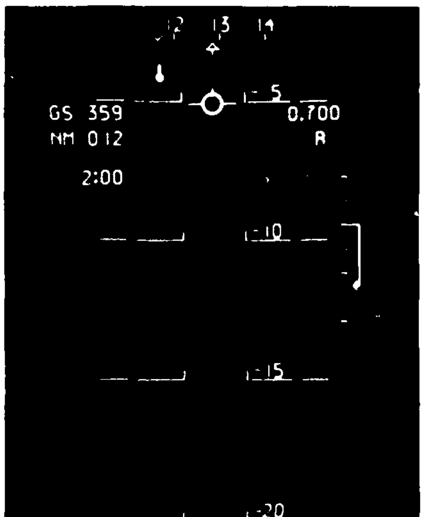


Photo 13 Mode NAVIGATION
allant vers station

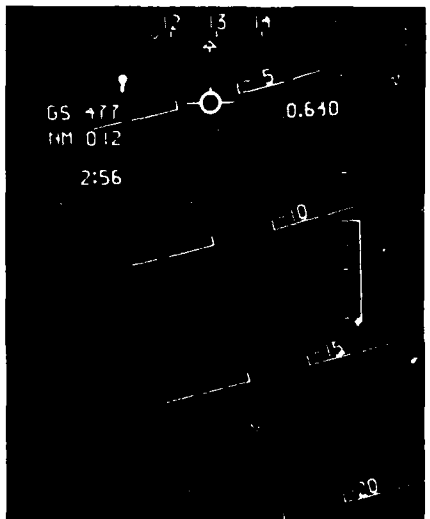


Photo 14 Mode NAVIGATION
venant de station

AIR TO GROUND WEAPON AIMING ACCURACY MEASUREMENT TECHNIQUES

by

H.M. Malley and C.J. Dewhurst

British Aerospace, Aircraft Group,
Warton Division, Preston,
Lancashire, England.

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SUMMARY

The flight clearance programme for a modern air-to-ground weapon aiming system demands that extensive flight trials are carried out to demonstrate that the performance of the system meets its specification requirements. With the increasing complexity of weapon aiming systems in terms of the multiplicity of ranging methods and attack modes provided, ways of minimising the flight trials requirements have to be considered. This paper describes trials methods and associated analysis techniques which are currently used and are designed to maximise the data available while minimising the number of flight trials carried out.

1. INTRODUCTION

The aim of this paper is to describe the trials philosophy and the resulting trials methods and analysis techniques which are used in the testing of current air-to-ground weapon aiming systems. The trials philosophy has been evolved over a number of years from experience in the testing of both air-to-air and air-to-ground weapon aiming systems. During this period the increasing sophistication of weapon aiming systems has demanded a significant increase in the size of the flight trials programme required to be undertaken. This increase together with the rapidly escalating costs of carrying out flight trials has been the major factor in dictating the trials philosophy which has been adopted.

The broad requirements of weapon aiming system flight trials are to evaluate the functioning and accuracy aspects of the system as defined in the aircraft's specification.

> Functioning can be defined as an examination of the operation of the weapon aiming system throughout the specified weapon delivery envelopes to assess its usability and safety. This essentially determines if any unsatisfactory or limiting aspects of system behaviour are likely to occur.
>
> Accuracy can be defined as an assessment of the accuracy with which the weapon aiming system can determine the point in space at which the weapon should be released assuming that the ballistic equations used in the airborne computations are correct.

Before discussing the trials methods and analysis techniques which are used to prove that the weapon aiming system meets its specification, it will be useful to describe the basic principles of a weapon aiming system and in particular the inherent errors which occur.

2. WEAPON-AIMING SYSTEM DESCRIPTION

The fundamental principle of an air-to-ground weapon aiming system is that the weapon is released when the aircraft's distance from the target is equal to the forward throw of the weapon. From this simple definition, the following requirements of a weapon aiming system can be identified:-

> Measurement of the aircraft's position with respect to the target.
>
> Determination of the aircraft's flight condition.
>
> Calculation of the weapon's forward throw.
>
> Generation of a weapon release signal.

These requirements can be fulfilled by a weapon aiming system which consists of four prime elements, namely Sensors, Computing, Displays and Controls, and Aircrew. The Sensor elements enable the aircraft's position with respect to the target and its flight condition to be determined. The Computing element calculates the weapon's forward throw and generates a weapon release signal when the latter is equal to the calculated distance to the target. The control of the attack is carried out by the Aircrew via the appropriate Display and Controls which interface with the Computing element. This in turn positions the relevent sensors during the attack. The mechanisation and integration of these elements varies from aircraft to aircraft. A typical example is illustrated in Figure 1. Although this paper concentrates on manually flown visually sighted attacks, the trials methods and analysis techniques can be equally applied to automatically flown blind attacks.

2.1 Sensors

The sensors provide the input data used in the weapon aiming computations. This data typically consists of positional information of the aircraft with respect to the target, the aircraft's velocity with respect to the ground and the airmass, the aircraft's attitude with respect to the ground and the local airmass characteristics.

In the overall analysis, errors associated with each of the sensor measurements used in the weapon aiming computations must be considered and their effect on the overall weapon miss distance determined. To calculate the magnitude of these errors it is necessary to provide a means of determining the true value of the sensed parameters. Typical sensor parameters considered in the analysis are:-

True Airspeed

Vertical Velocity

Velocity along heading

Velocity across heading

Height

Pitch Angle.

2.2 Computing

This is the heart of the weapon aiming system where the necessary computations to effect a weapon release are carried out. These computations typically consist of calculating the following:-

Weapon forward throw

Distance to target

Time to weapon release

Position of Head Up Display symbols.

With the advent of high speed digital computers the accuracy of the computations and in particular the repeatability has greatly improved. Even so, errors due to the discrete nature of digital computing do arise and although needing to be accounted for, they are no longer of great significance.

2.3 Displays and Controls

The modern Head Up Display provides the Pilot with sufficient information to fly the aircraft 'head up' in all phases of flight. The information displayed includes alpha-numeric and analogue displays of quantities such as height, airspeed, incidence, vertical velocity, attitude and heading. In addition in the weapon aiming phases, the positions of the computed target indicator and weapon impact point are displayed. Selection and control of the displayed symbology are exercised by the Pilot via the various Control Units.

Errors which arise in the positioning of the symbols can be typically due to incorrect display input information, symbol generation inaccuracies or incorrect aircraft/equipment harmonisation. These errors must be determined and their effect on the overall weapon miss-distance evaluated.

2.4 Aircrew

In a visual attack the Pilot is the most significant element in the system. The actions he takes influence the accuracy of the attack. He is interactive with the weapon aiming system via the Control Units in setting up the attack modes, selecting the weapon station and changing the phases as the attack proceeds up to committing the weapon to release. In some attack modes the Pilot can refine the computed target position. It is desirable however to remove him as much as possible from any continuous process and give him minimal discrete actions to perform. The display of the computed target position in relation to the real target at the weapon release point is the measure of Pilot's performance.

3. TRIALS PHILOSOPHY

As stated in the introduction to this paper the prime purpose of weapon aiming system flight trials is to prove that the weapon aiming system operates satisfactorily and meets the accuracy requirements stated in the aircraft's specification. It is the accuracy requirements which dictate the form of flight trials to be undertaken, the operating aspects are capable of being assessed during these trials.

In the aircraft's specification, the accuracy requirements are usually stated as weapon miss distances, i.e. the distance between the weapon impact point and the target which the weapon aiming system should achieve for a given ranging method, attack mode and weapon at defined flight conditions. These miss distance figures are normally qualified by the percentage of weapons which must meet this requirement and are often referred to the aircraft fleet.

The most obvious way to determine the accuracy of the weapon aiming system would be to deliver a single weapon against a target and measure the resulting miss distance. Due to the unique nature of the elements of the weapon aiming system, of the aircraft and of the weapon, it is extremely unlikely that a second or third weapon delivered from the same aircraft would give the same result. It would therefore be necessary to deliver a series of weapons for each combination of ranging methods, attack modes, weapon options and flight conditions to determine the scatter in miss distance which can then be related to the various specification requirements.

This form of trial is termed the "statistical approach" and assumes that the errors which contribute to the overall weapon delivery error are independent. For a modern ground attack aircraft with its multiplicity of ranging methods, attack modes and weapon options, this statistical approach demands that a large number of attacks are carried out to establish the required confidence in the overall results. The answers are however unique to the aircraft and weapon aiming system combinations tested and may not be sufficiently representative of the aircraft fleet due to the data sample available. Another problem with this approach is that the measurement of weapon miss distance gives very little information about the performance of the elements which make up the weapon aiming system, i.e. Sensors, Computing, etc. During a flight trial, even if the weapons hit the target, it is possible to have self-cancelling errors which under other conditions or with different equipment could in the extreme produce additive errors. Thus while the flight trial could indicate satisfactory accuracy, the aircraft fleet accuracy may not be met.

Due to the large expenditure in time and money inherent in carrying out weapon aiming system flight trials, increasing emphasis has been placed on what is termed the "analytical approach" to overcome the deficiencies of the statistical approach. The basis of the analytical approach is that the overall weapon miss distance error can be explained from an understanding of the individual system component errors. This can be achieved by comparing the actual system parameters at the weapon release point with their true values. To carry out this process extensive ground and airborne instrumentation is required.

By relating the analysis to the weapon release point, it is not necessary to release weapons to determine the accuracy of the weapon aiming system. This in turn leads to some simplification in the overall trials operation and a reduction in costs. For weapon release point trials, a mathematical simulation of the weapon is used to determine the miss distances on the ground caused by the system errors present at the release point.

The analytical approach can be used to determine the performance of the various components of the weapon aiming system for each attack undertaken. From an examination of flight trials data for each combination of ranging method, attack mode and weapon option at selected flight conditions, a large sample of data describing the performance of the various system components can be obtained. Equivalent data samples can be obtained from several of the combinations tested, and therefore the need to carry out repetitive trials, as in the case of the statistical approach, can be considerably reduced. In the case of a system with a single ranging method, attack mode and weapon option there would be no difference in the number of attacks required using the statistical or analytical approaches. However for a system with two ranging methods, two attack modes and two weapon options, the analytical approach requires typically one-third of the trials compared to the statistical approach. This reduction in trials can be achieved without affecting the confidence in the results.

4. INSTRUMENTATION

As stated previously for an analytical trial it is necessary to have both comprehensive ground and airborne instrumentation. The following paragraphs discuss typical instrumentation systems which are used for weapon aiming system trials and the initial stages of processing the recorded data.

4.1 Ground Instrumentation

4.1.1 Range Facilities

To enable the actual aircraft attack conditions to be determined, i.e. aircraft velocity, attitude and position with respect to the target, some form of ground instrumentation is required. In the United Kingdom, the Royal Aircraft Establishment provides several Instrumented Ranges. These Ranges have various ground-based tracking facilities including synchronised kinetheodolite cameras which are capable of tracking the aircraft during the final stages of an attack.

In addition the Ranges have meteorological facilities for measuring air temperature and pressure at various heights above the ground. Wind data can be determined by using the kinetheodolite cameras to track meteorological balloons which are released before and after trials sorties. Alternatively wind data can be obtained by using the kinetheodolite cameras to track a smoke flare which is fired from the trials aircraft at the simulated release point. Experience has shown that this latter method provides a more accurate measurement of the wind conditions at the weapon release point but requires additional ground and airborne facilities.

4.1.2 Range Data Processing

The position of the aircraft on the kinetheodolite camera films is subsequently measured and together with knowledge of the kinetheodolite cameras attitude and position, the position of the aircraft in a convenient set of axes related to the target is determined. The trajectory points are normally calculated every one fifth of a second corresponding to the kinetheodolite camera's frame speed. The aircraft's trajectory is usually determined from about 12 seconds prior to weapon release to 3 seconds after release. A typical accuracy of positional data derived from kinetheodolite camera film measurement is better than 1 metre (S.D.). This is considered to be adequate for all weapon aiming system measurement requirements. The velocity of the aircraft is derived from the positional data using a standard differentiation technique.

In the weapon release point error analysis scheme described, the aircraft's position and velocity at points either side of the weapon release instant are used in order that the aircraft's position with respect to the target and its velocity at the release instant can be determined. These values are known as the REFERENCE data.

4.2 Aircraft Instrumentation

It is also necessary to record the values of a number of weapon aiming system parameters onboard the aircraft. The value of these parameters which are obtained from the Weapon Aiming Computer and the Head Up Display are known as SYSTEM data.

4.2.1 Weapon Aiming Computer

To record data from the Weapon Aiming Computer, it is programmed to sample and output the required parameters to the aircraft's instrumentation system. Typically the latter consists of an electronic unit which interfaces with the Weapon Aiming Computer and suitably formats the required data for recording on a magnetic tape recorder in the aircraft. After flight the magnetic tape is replayed on a ground station and the data retrieved for subsequent data processing.

It is essential that all sensor inputs to the weapon aiming equations and all output quantities to the display and weapon sub-systems are recorded. In addition, it is desirable to record some intermediate parameters in the weapon aiming computations. Although these intermediate parameters are not used in the normal weapon release point analysis they can provide invaluable diagnostic data.

As part of the overall data recorded by the aircraft's instrumentation system, a number of weapon system parameters termed 'frozen data' are recorded at the weapon release instant. Since digital computers work in discrete cycles it is normal practice to predict if the weapon release point will occur in the next cycle. When this is determined a timing counter is activated which enables the weapon release pulse to be generated with a high degree of timing accuracy. It is the input parameters to this last cycle of weapon aiming calculations which are the frozen data.

4.2.2 Head Up Display

For all weapon aiming trials, the aircraft are fitted with Head Up Display Cameras which record on 16 mm cine-film the view through the Head Up Display as seen by the Pilot. In order to relate positions of symbols on the display for analysis purposes, a datum symbol is displayed at the electrical centre of the cathode ray tube of the Head Up Display which is in turn aligned to the aircraft's Longitudinal Fuselage Datum. It is therefore possible to relate all measurements taken of Head Up Display symbology or outside world positions directly to the aircraft's Longitudinal Fuselage Datum.

The Head Up Display film is analysed using a film analyser which measures the positions of selected displayed symbols or of ground objects relative to the datum symbol. This data is automatically digitised and stored on an appropriate medium. Typical features which are measured are as follows:-

Computed Weapon Impact Point

Computed Target Indicator

Actual Target.

In the analysis scheme, interpolation of the symbol positions is carried out to obtain their true position at the weapon release instant. The outside world measurements have the appropriate windscreen distortion correction applied (see Section 6.1.2).

4.3 Correlation of Airborne and Ground Data

The ground and airborne instrumentation provide time histories of a series of parameters which are recorded against their own specific time bases. To enable these two sources of data to be used together, it is necessary that a means of correlating them is provided. This is achieved by generating a Radio Tone on the aircraft which is interrupted by the weapon release pulse generated from the Weapon Aiming Computer. The tone is received at the Instrumented Range and recorded against the ground instrumentation time base. The Range uses the interruption of the Radio Tone as the reference for the ground data. The Radio Tone is also recorded on the aircraft's instrumentation, thus enabling correlation of the two sources of data to be achieved.

In addition it is necessary to correlate the information obtained from the Head Up Display Camera which operates at a nominal 16 frames/second with the information recorded from the Weapon Aiming Computer. This is done by recording the time at which each opening of the Camera shutter occurs, on the aircraft's instrumentation.

Figure 2 illustrates how the correlation of ground and airborne data is mechanised.

5. WEAPON RELEASE POINT ERROR ANALYSIS

The objective of the analysis is to break down the overall weapon miss distance into its various components which arise through the errors inherent in the weapon aiming system. This form of analysis enables the performance of the weapon aiming system to be quantified for individual attacks and provides the ability to identify those factors which effect the systems accuracy. It also allows a self-check to be carried out by ensuring that the overall miss distance is fully accounted for by the miss distance components.

The analysis method uses the fact that the actual target and the weapon impact point as calculated by the weapon aiming system are both points in space, which under ideal conditions should coincide, i.e. zero miss distance. However, as this is very unlikely to occur, a number of points can be chosen between the target and the calculated impact point such that the separations between them are accounted for by the various miss distance components of the weapon aiming system. This is illustrated in Figure 3. Each point in the analysis scheme lies on either one of two planes, the Reference plane which is based on the true height of the aircraft or the System plane which is based on the system height of the aircraft.

A basic block diagram of the analysis scheme is shown in Figure 4.

Inputs to the analysis program are as follows:-

Range data consisting of aircraft position and velocity and wind data from either side of the release point.

Aircraft System data consisting of height, speed, etc. frozen in the Weapon Aiming Computer at weapon release.

Head Up Display film data consisting of target, computed weapon impact point and computed target indicator positions from frames either side of the release point.

The analysis scheme determines the overall weapon miss distance and derives the contribution of the following error sources:-

System Height

Specific Sensor (Airspeed, Velocities, Wind, etc.)

System Computing

Release Time Computation

Head Up Display Symbol Positioning

Pitch Angle/Harmonisation

Azimuth Steering

Pilot Aiming

Figure 3 as referred to earlier, shows the analysis scheme points for a visually aimed air-to-ground weapon attack. *These are defined as follows:-*

T is the Reference value of target position and is obtained from Range data.

GR is obtained using a model of the Weapon Aiming Computer's (WAC) ballistic calculations with Reference values of height, speed, wind and dive angle to derive weapon forward throw at target height.

GS is obtained using a model of the WAC ballistic calculations with Reference values of speed, wind and dive angle and System value of height to derive weapon forward throw at System height.

FS is obtained using a model of the WAC ballistic calculations with System values of height, speed, wind and dive angle to derive weapon forward throw at System height.

ES is the System value of weapon forward throw.

DS is the System value of range to the target.

CS is the position of the computed target indicator symbol measured from the Head Up Display film and extrapolated to the weapon release instant; it is calculated using the System value of pitch angle on the System height plane.

HS is the position of the computed weapon impact point symbol and derived as CS.

BS is the equivalent of CS but is calculated using the Reference value of pitch angle on the System height plane.

JS is the position of the computed weapon impact point symbol and derived as BS.

BR is the projection of BS on the Reference height plane.

AR is the apparent target position on the Reference height plane.

X denotes across heading parameters.

The analysis scheme combines the points defined above to produce the following contributions to the overall weapon miss distance:-

<u>Along Heading</u>

GR-T is the overall weapon miss distance.

(GR-GS) + (BS-BR) System Height Error is the miss distance due to the error in the System value of height.

GS-FS Specific Sensor Error is the miss distance due to the error in System values of velocities and wind.

FS-ES System Computing Error is the miss distance due to the difference between the System's calculation of forward throw and a calculation using the same ballistics model with System data.

ES-DS Release Time Computation Error is the miss distance due to the difference between the weapon forward throw calculated by the System and the System's range to target at the weapon release instant.

DS-CS Head Up Display Symbol Positioning Error is the miss distance due to the difference between the System's range to target and the measured computed target indicator position using System pitch angle.

CS-BS Pitch Angle/Harmonisation Error is the miss distance due to pitch angle error, Head Up Display harmonisation, datum dot positioning, etc.

BR-AR Pilot Aiming Error is the miss distance due to the difference between the computed target indicator position and the target on the Head Up Display film.

AR-T Windscreen Distortion Error is the miss distance caused by the optical distortion of the windscreen.

<u>Across Heading</u>

The Overall, System Height, Specific Sensor, Apparent System Computing and Pilot Aiming Errors are defined in the same way as for along heading errors. The remaining errors are as follows:-

EXS-HXS Head Up Display Symbol Positioning Error is similar to the along heading equivalent, but uses the computed weapon impact point instead of the computed target indicator position.

HXS-JXS Pitch Angle/Harmonisation Error is similar to the along heading equivalent but uses the computed weapon impact point instead of the computed target indicator position.

JXS-BXS Azimuth Steering Error is the miss distance due to the offset of the computed weapon impact point from the computed target indicator position in azimuth.

The analysis scheme also calculates the following weapon aiming system component errors:-

- True Airspeed
- Velocity along heading
- Velocity across heading
- Vertical Velocity
- Wind velocity along heading
- Wind velocity across heading
- Height
- Pitch Angle
- Pilot Aiming
- Head Up Display Symbol Position.

An illustration of the data which is input to and output from the analysis scheme is shown in Figures 5 and 6.

6. TRIALS PLANNING

The approach to weapon aiming system trials, the data recorded and the method of analysis of this data have all been discussed in the preceding sections; for completeness a brief review of the trials planning aspects is given in the following paragraphs.

Trials planning can be divided into two phases, namely ground testing and flight testing.

6.1 Ground Testing

Ground testing can be divided between work which is carried out on a Weapon System Rig in integrating the elements of the weapon aiming system and work which is carried out on the aircraft including the specific on-aircraft calibrations.

6.1.1 Weapon System Rig

The Weapon System Rig is an assembly of all the avionic units which form the overall weapon system. The avionic units are connected together as they would be on the aircraft and can be operated as a complete system. As it is not possible to function all of the sensors on the ground, simulation techniques have been developed which can provide dynamic inputs into those elements of the weapon system which require them. This enables comprehensive integration checks of the various elements and functioning of the software in particular, to be checked prior to the installation of equipment in an aircraft.

The use of a Weapon System Rig prior to the commencement of the flight testing phase has been shown to save valuable time and expense by enabling the early identification and correction of system problems to be achieved. In addition it has been found possible to refer problems discovered during flight trials back to the System Rig for investigation under a more controlled environment and at a greatly reduced cost compared with continuing the investigations in flight.

6.1.2 Aircraft Ground Checks

After installing the equipment in an aircraft and carrying out the standard aircraft and equipment checks, harmonisation of the sensors and displays to the airframe axis is carried out, i.e. establishing the angular references of the various system components to that of the aircraft's Longitudinal Fuselage Datum. Trials procedures dictate that the harmonisation of the relevant sensors and displays is checked at regular intervals throughout the flight trials.

From experience it has also been found desirable to calibrate the aircraft's windscreen for optical distortion. This is done using photographic techniques.

6.2 Flight Testing

Using the analytical approach it is necessary to ensure that all the ranging methods, attack modes and weapon options of the weapon-aiming system are flight tested in all regions of the required flight envelope. It is therefore normal practice to carry out a series of attacks at a mid-flight envelope condition for a given attack mode, ranging method and weapon option, with a reduced number of attacks at each of the corner points of the weapon delivery envelope. This is illustrated in Figures 7 and 8. For the prime attack modes it is considered that typically of the order of seven flight conditions should be investigated but in reversionary modes this number is usually reduced. The main criterion in the final choice of the attack parameters is to ensure that a sufficient sample of data is available for the evaluation of the performance of the weapon aiming system components at each flight condition.

7. SPECIFICATION PROVING

In addition to using the ground and airborne instrumentation data to determine the accuracy of the weapon aiming system and its various components at the weapon release point, a more detailed comparison of the data obtained during the weapon aiming phase is carried out. This involves a comparison of the time histories of relevant System and Reference parameters which provides valuable information with regard to the usability of the weapon aiming system. It also enables the dynamic behaviour of the system components to be examined and provides a more detailed statistical description of their performance.

To handle the large quantities of data involved in such a comparison, a computer program called Interactive Trials Analysis System (ITAS) has been developed. This system makes it possible for either trials or design engineers with little or no data processing experience to manipulate flight trials data in order to investigate the performance of the various system components. This consists of examining the performance of the various system components over the full available time history of an attack either on their own or with respect to other parameters.

Figures 9 and 10 illustrate the typical use of ITAS in producing a graphical representation of the system component error against time and also against another parameter. From the former information a quantitative measure of a system component's performance can be determined, while from the latter it is possible to assess the functioning of one parameter against another.

As a result of carrying out a flight trials programme which exercises the various combinations of ranging methods, attack modes and weapon options at the selected flight conditions, a large sample of data relating to the performance of the system components is available. To obtain the best estimate of the performance of the system components the data derived from each attack is combined by statistical methods. In carrying out this process it is necessary to examine the data for two main criteria. Firstly, is the data correlated with any other parameter, typically a flight condition, e.g. speed. If some correlation is found, it is necessary to handle such data in a manner which observes the correlation criteria, but if no correlation is present it is possible to combine the data as a single large sample. Secondly, is the data a random sample. This is obviously more difficult to quantify and becomes a matter of judgement as to whether sufficient aircraft, equipment, pilots, etc. have been used in the trials.

To establish the best estimate of overall performance of the weapon aiming system, the best estimates of the performance for each individual system components are input to an overall weapon aiming sensitivity model which determines the respective miss distance components at any given flight condition. These miss distance components are then added together using a statistical technique to determine the overall weapon miss distance, i.e. the aircraft fleet performance. From the best estimates of the performance of the system components it is possible to check that individual equipments are meeting their performance specifications. In addition from the best estimate of overall miss distance it can be seen if the weapon aiming system as a whole meets the requirement laid down in the aircraft specification.

8. CONCLUSION

This paper has reviewed flight trials methods and analysis techniques which are currently used in the assessment of an aircraft's weapon aiming system and in proving that it meets its specification. These trials form part of the much larger task of clearing an aircraft and its weapon system for operational use. Emphasis has been placed on the analytical nature of the trials which while minimising the flight trials required demands a high degree of both ground and airborne instrumentation.

In the future it may be possible to achieve some improvement in the accuracy of measurement of the Reference data, i.e. aircraft position and velocity. It is considered however that emphasis should be placed on reducing the overall complexity of the trials equipment and data processing required. In particular the use of automatic tracking devices which provide a direct output of aircraft position and velocity would reduce the data processing requirements, in addition to accelerating the availability of Reference data for analysis purposes. Similarly automatic reading of Head Up Display data by using video recorders and auto-tracking replay devices would again improve the overall efficiency of the analysis.

In summary while improvements can be foreseen in the areas of data gathering and data processing, the concept of determining the system component errors at the weapon release point and the reconstitution of the overall miss distance error from the large sample of trials data will continue to be the basis of the analysis techniques for the proving of future weapon aiming systems.

ACKNOWLEDGEMENT

The authors would wish to acknowledge the advice and assistance given by representatives of A&AEE Boscombe Down and in particular Mr [illegible].J. Funday, in establishing the techniques described in this paper.

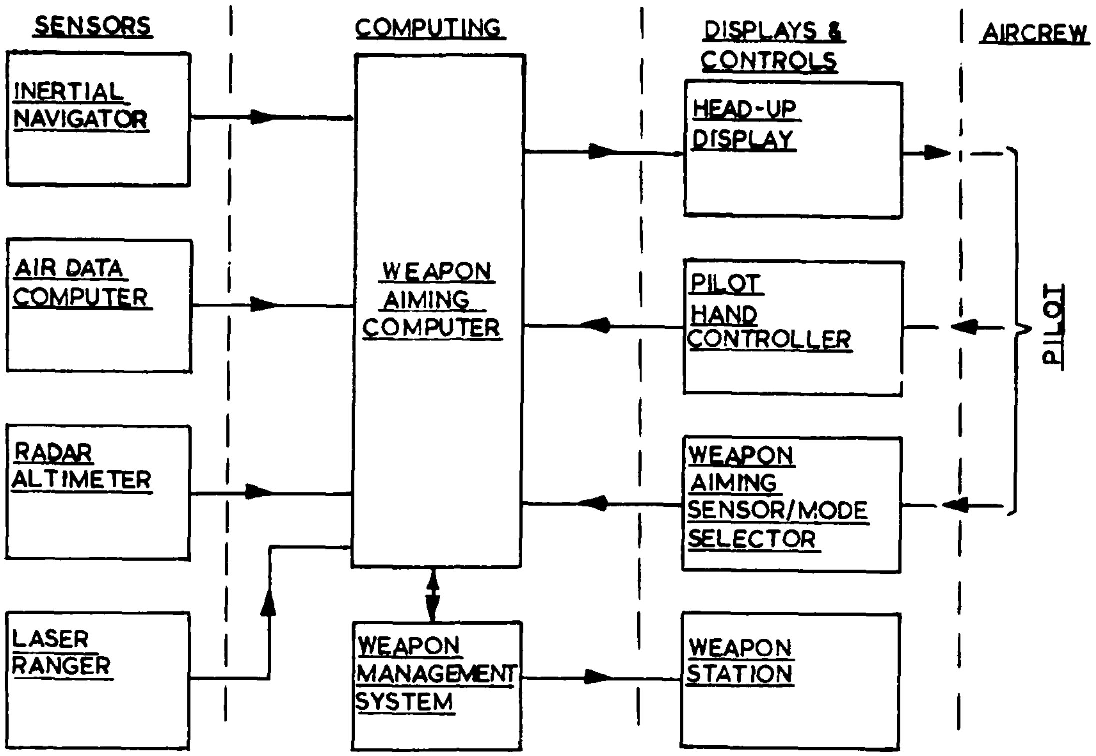

Figure 1. Weapon Aiming System Block Diagram

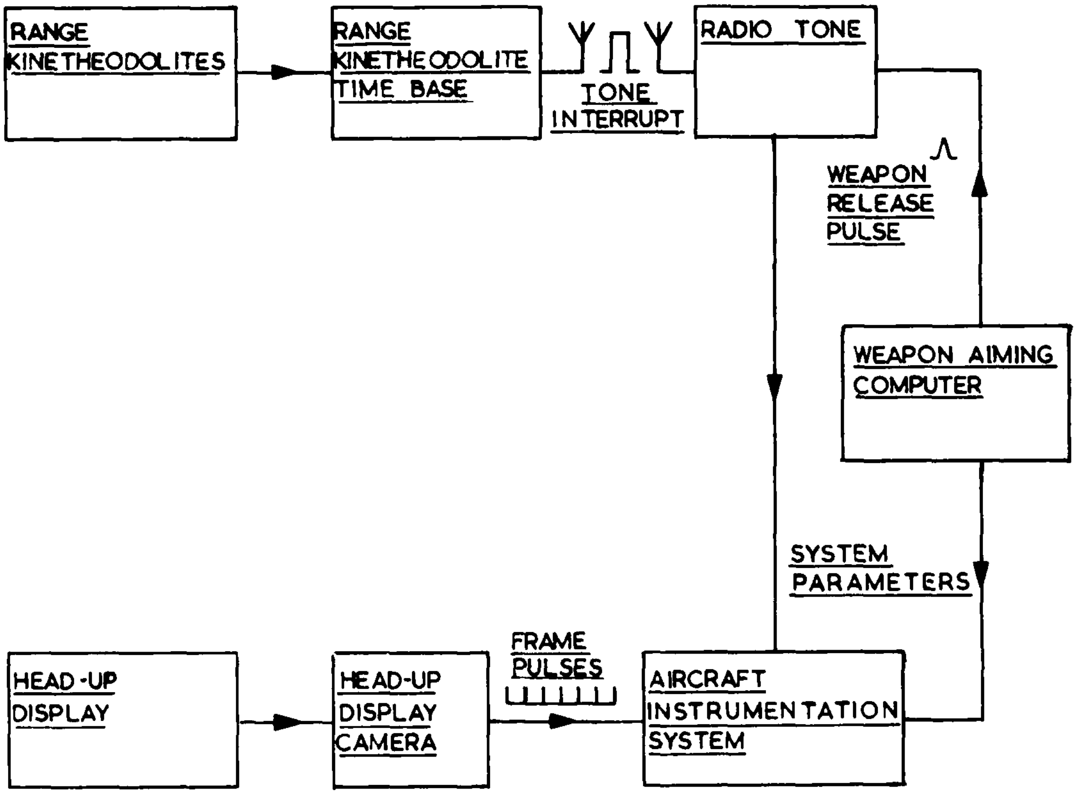

Figure 2. Correlation of Airborne and Ground Data

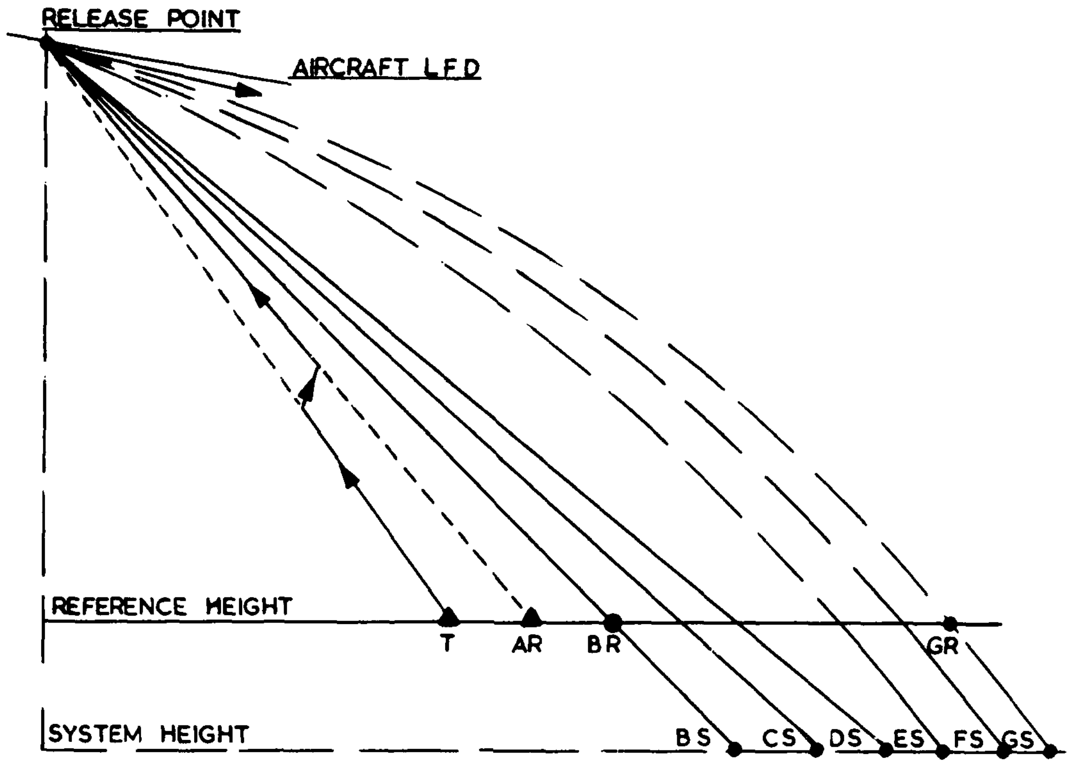

Figure 3. Schematic Diagram of the Miss Distance Components (Along Heading)

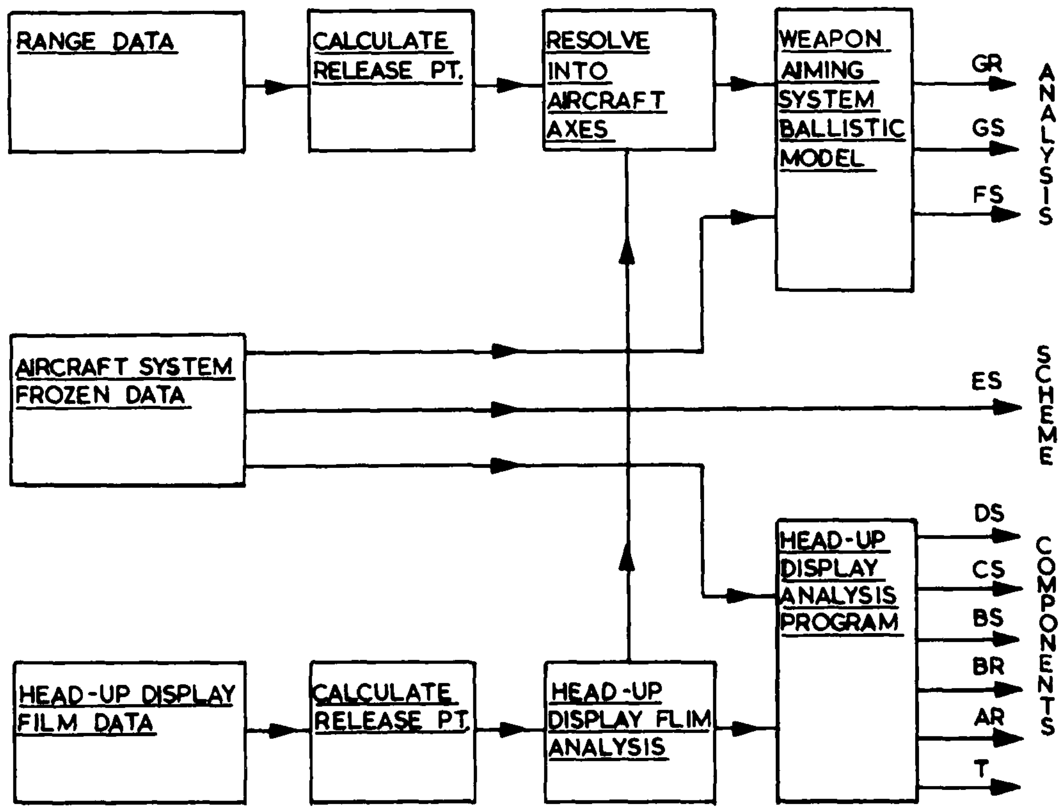

Figure 4. Basic Block Diagram of the Analysis Scheme Components (Along Heading)

WEAPON RELEASE POINT DATA

AIRCRAFT DATA

	ALONG	ACROSS	VERT	TAS
VELOCITY	**	**	**	**
RANGE	**	**	**	
WEAPON THROW	**	**		
WIND	**	**		
HEADING **	PITCH	**	BANK	**

RANGE DATA

AIRCRAFT AXES	ALONG	ACROSS	VERT
POSITION	**	**	**
VELOCITY	**	**	**
WIND	**	**	

HUD FILM DATA

	AZ	EL
APPARENT TGT	**	**
TARGET INDICATOR	**	**
WEAPON IMPACT PT.	**	**
TRUE TARGET	**	**
PITCH ANGLE **	HEADING **	

Figure 5. Analysis Scheme Data Input

MISS DISTANCE BREAKDOWN

ERROR DESCRIPTION	ALONG HEADING	ACROSS HEADING
OVERALL ERROR (FT):	**	**
SYSTEM HEIGHT ERROR (FT):	**	**
SPECIFIC SENSOR ERROR (FT):	**	**
SYSTEM COMPUTING ERROR (FT):	**	**
RELEASE TIME COMPUTATION ERROR (FT):	**	
HUD SYMBOL POSITIONING ERROR (FT):	**	**
PITCH ANGLE/HARMONISATION ERROR (FT):	**	**
AZIMUTH STEERING ERROR (FT):		**
PILOT AIMING ERROR (FT):	**	**
SYSTEM SENSOR ERRORS, ETC.		
TRUE AIRSPEED (FT/S):	**	
VELOCITY (FT/S):	**	**
VERTICAL VELOCITY (FT/S):	**	
WIND (FT/S):	**	**
HEIGHT ABOVE TARGET (FT):	**	
PITCH ANGLE (DEG):	**	
AIMING ERROR (DEG):	**	**
AZIMUTH STEERING ERROR (DEG):	**	
HUD POSITIONING ERROR (DEG):	**	**

Figure 6. Analysis Scheme Data Output

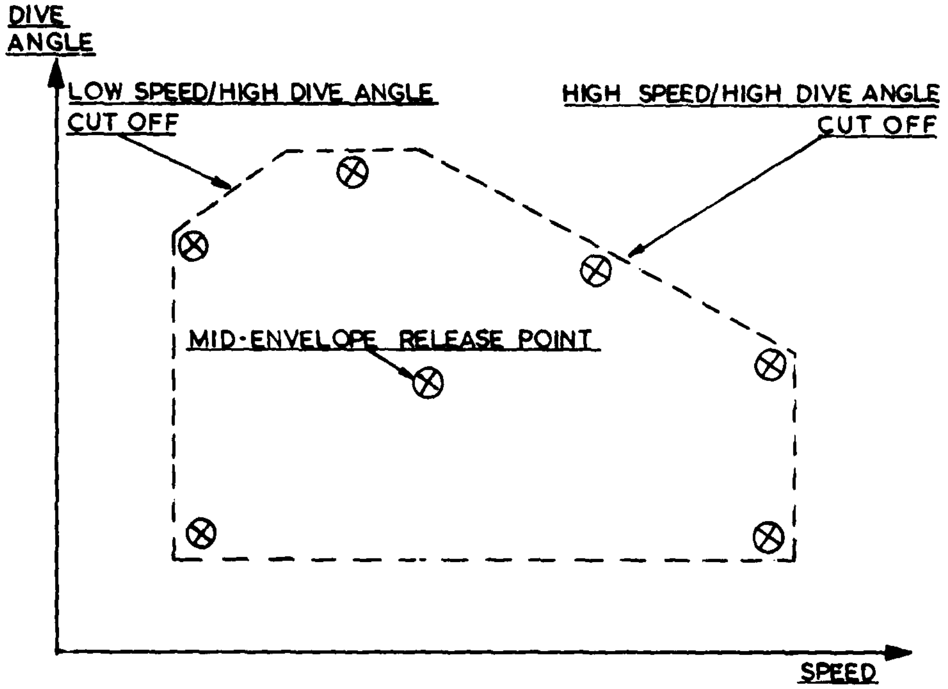

Figure 7. Weapon Delivery Envelope - Freefall Weapons

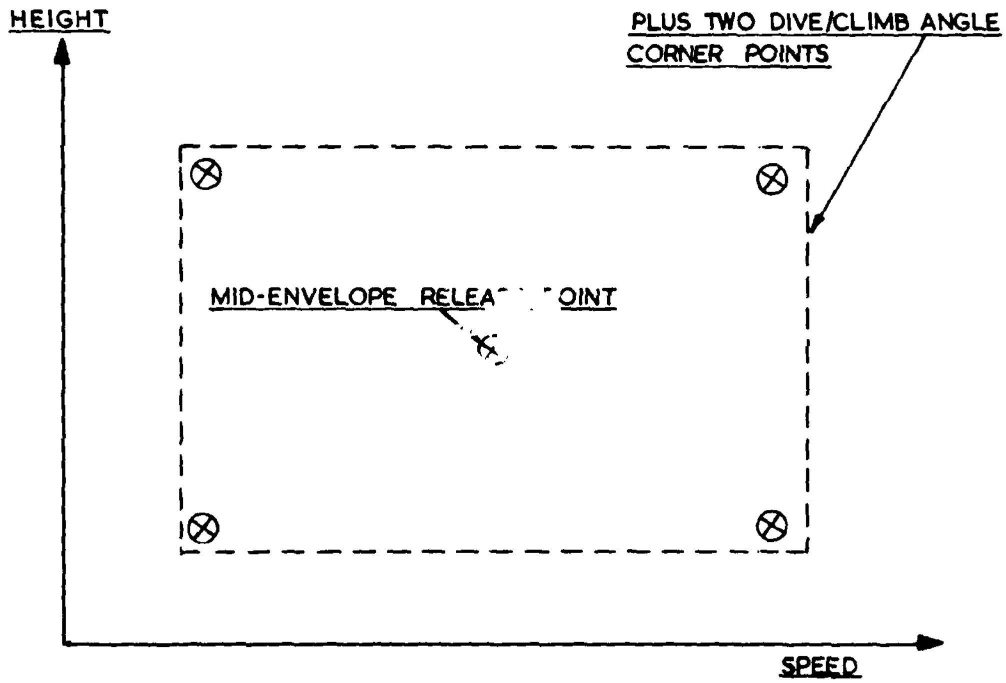

Figure 8. Weapon Delivery Envelope - Retarded Weapons

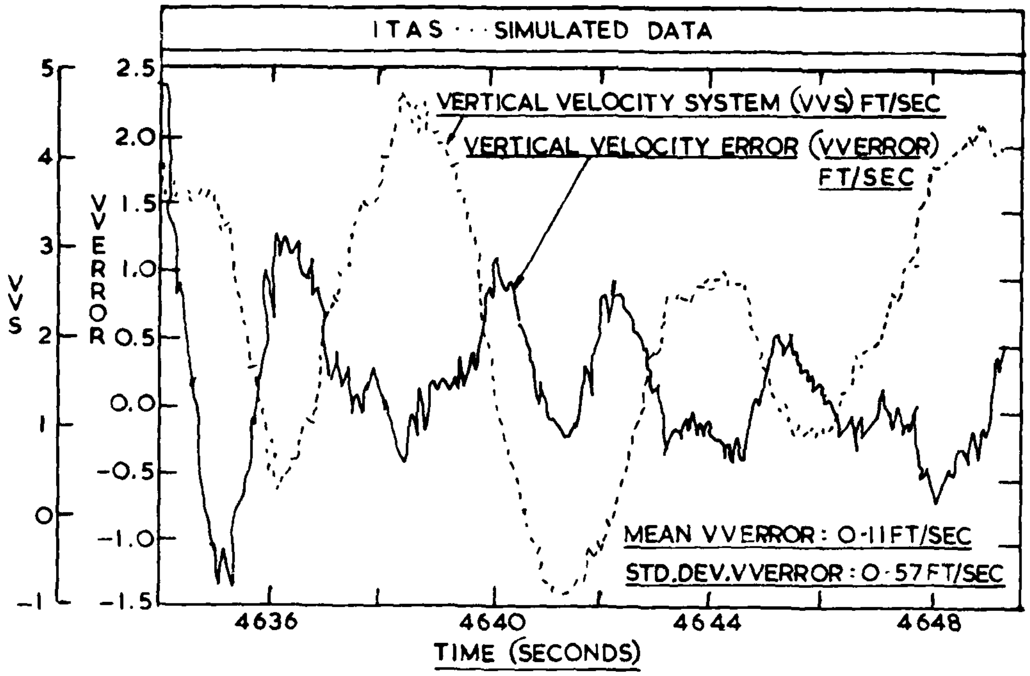

Figure 9. Typical Time History Plot

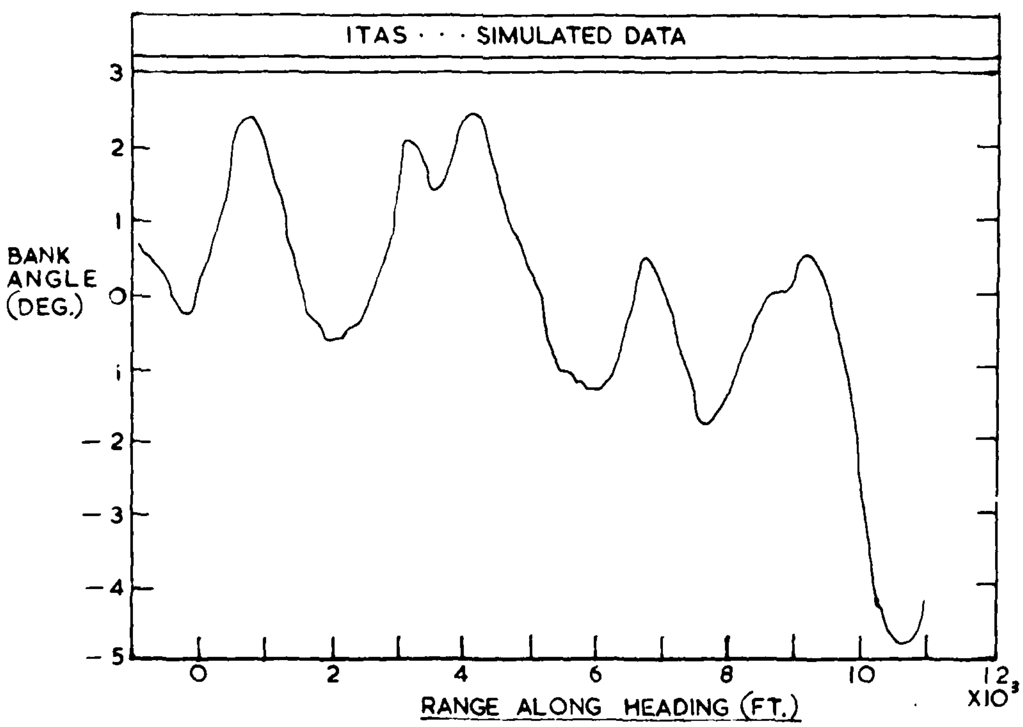

Figure 10. Typical Parameter Cross Plot

DATA ACQUISITION AND ANALYSIS SYSTEM AS A TRAINING DEVICE FOR SIMULATED CONVENTIONAL WEAPON DELIVERY

by

C.F.G.M. Hofman

National Aerospace Laboratory, NLR
Amsterdam
The Netherlands

and

J. Batenburg

Royal Netherlands Air Force

1. INTRODUCTION

The tasks to be performed by the pilot during conventional weapon delivery in high speed fighter aircraft depend on the systems available aboard the aircraft, and on the type of weapon used.
Presently the main part of the air-to-ground training method in the Netherlands consists of the delivery of training bombs at training ranges.
The disadvantages of this method are:
- the site of weapon delivery (and in many cases also the direction from which the attack is initiated) is often the same (learning effect!)
- the performance of the pilot is expressed in one quantity only: the score. The release conditions are not measured
- the high consumption of weapons (expensive!).

In order to avoid these disadvantages, a simple system, called: "Delivery and Impact Analysis System" (DIAS) has been developed and tested by the National Aerospace Laboratory NLR, under contract for the Royal Netherlands Air Force (RNLAF).
This system, based on a photogrammetric method, yields release conditions, the nominal weapon impact position and the weapon time of flight. Simulated attacks on a great variety of realistic targets can easily be evaluated and validated as there is no need to drop training weapons. Furthermore no ground-based instrumentation in the target area is needed. The system consisting of an airborne data acquisition system installed in the aircraft and a ground-based processing and analysis system at the airbase allows a debriefing of the pilot within half an hour after completion of the mission.
This paper gives a description of the system.
Attention is paid to the system requirements, the system evaluation and the implementation in an operational NF-5 squadron of the RNLAF.

2. SYSTEM REQUIREMENTS

The RNLAF using, among others, NF-5 aircraft to carry out low level attacks wished to have a system realized for that type of aircraft which should meet the following requirements:

1. Accurate assessment of pilot performance concerning simulated weapon delivery in an (semi) operational environment by yielding:
   - weapon release conditions and
   - weapon impact positions for each attack run
   - statistical overviews of the mission results in relation to pilot, weapon, target, dive angle etc.
2. Training of pilots concerning weapon delivery by a fast feed-back of the mission results.
3. A simple system characterized by:
   - a minimum of equipment
   - a minimum of personel
   - low costs
   - no influence on pilot procedures during weapon delivery
   - easy handling of the airborne system on the ground
   - user's friendly processing and analysis system.
4. No ground equipment in the target area.
5. No aircraft modifications.
6. No influence on the operational status and handling qualities of the aircraft.

Although the system has no war task and therefore, no back-up system is needed, the RNLAF expressed the following requirements concerning the "mean time between failures" (MTBF):
- airborne system: 100 flying hours
- ground-based system: 1000 working hours.

Based on the number of required "DIAS" sorties per aircraft and the related working hours of the ground-based system one failure every half a year may occur.

3. SYSTEM DESCRIPTION

3.1 Introduction

DIAS consists of two parts, namely:
- an airborne data acquisition system attached to the aircraft and
- a ground-based data processing and analysis system at the airbase.

The airborne system contains a photo camera by which at the moment of weapon delivery two photographs of a once measured target area are taken. The information available from both photographs is processed and analyzed with the aid of the ground-based system and yields the results of the attack carried out.
The ground-based system comprises, among others, a minicomputer, a display terminal and a photo reader.
A "DIAS squadron" has at its disposal several aircraft equipped with an airborne system and one ground-based system.

3.2 Airborne data acquisition system

The airborne system of DIAS contains the following components:
- a photo camera (24 x 36 mm), ROBOT motor recorder 36 C, 24 VDC with an exposure time of 1/500 s. The camera is equipped with an object-glass "Schneider Xenon" f/1.9 -50 mm and a central type shutter
- a release magneto (solenoid)
- an automatic diaphragm control unit of which the electronics were modified and integrated with
- an electronic unit, designed to control the camera
- a housing provided with a front glass and a retractable screen controlled by an electric motor (TRW Globe Motors, type 5 A 548-8).

The camera is attached to the aircraft in an upright position with the optical axis pointing forward, 11 degrees nose down with respect to the aircraft's longitudinal axis. No relevant part of the camera field of view is obscured by any store attached to the centerline pylon.

At the moment of weapon delivery the camera is activated by pressing the bomb button (pilot's action) while for the second photograph the camera is activated automatically [illegible] ms later by the electronic unit which also controls the camera diaphragm and film transportation.
The housing of the airborne system has been attached to the aircraft on the spot of the lower part of the centerline pylon which has been removed. The difference between the original and "DIAS" centerline pylon of the NF-5A concerning the external shape is shown in figure 1.

The housing consists of two parts of which one is attached to the centerline pylon permanently and contains the electronic unit. The second part, the "DIAS nose", is attached to the first part by two "quick release latches" (easy removable) and contains the photo camera (Fig. 2).

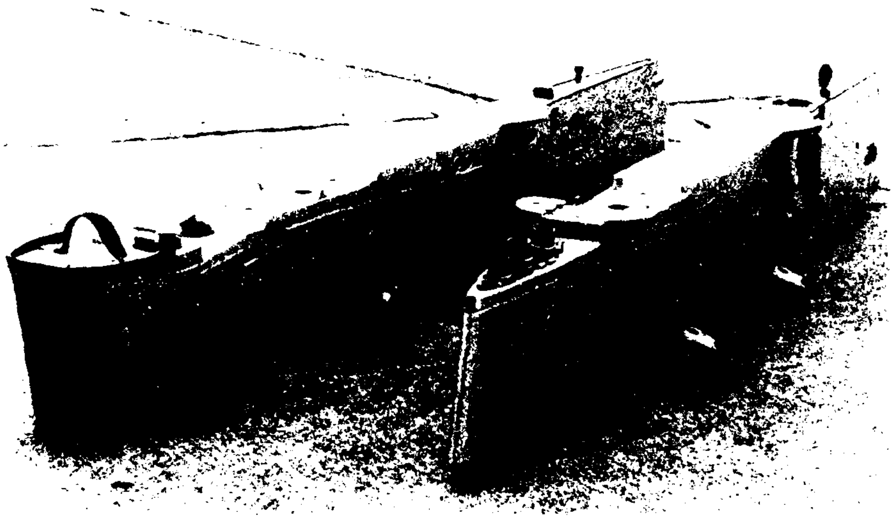

Fig. 1 The original and DIAS centerline pylon of the NF-5A aircraft

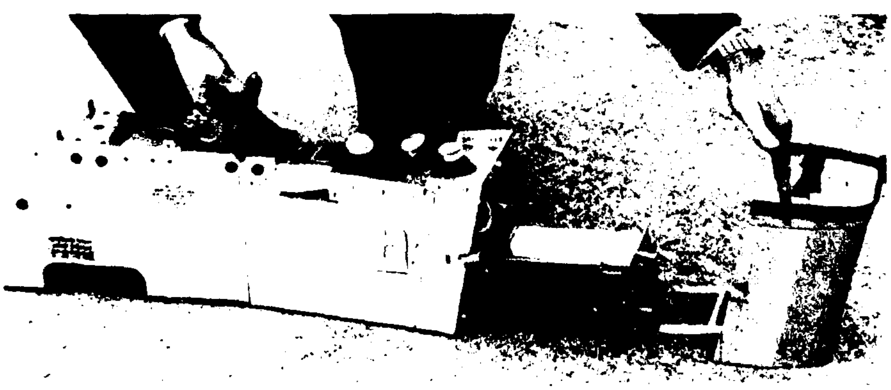

Fig. 2 The DIAS nose removed from the modified centerline pylon

After completion of the flight the "DIAS nose", is brought to the squadron building where the film is removed (easy accessible) and developed.
If no DIAS mission is carried out the real "DIAS nose" is replaced by a "dummy nose" with an identical external shape (Fig.3).

The housing of the airborne system is designed such that all centerline stores can be carried as before (Fig.4).

Fig. 3 The real and dummy DIAS nose

Fig. 4 The rocket bomb dispenser attached to the DIAS centerline pylon

The retractable screen which protects the front glass of the housing against dust and salt (low level missions) will be opened during the weapon delivery phase through the activation (pilot's action) of either the "bomb arming switch" (bombing) or the "armament position selector switch centerline" (rocketry). Pilot's actions during a "DIAS mission" do not differ from those to be carried out normally during an attack mission.
In addition to the modification of the centerline pylon construction a minor modification concerning electrical wiring inside the pylon was needed (power supply, signals to camera). Aircraft modifications are not involved.

3.3 Ground-based data processing and analysis system

The ground-based data processing and analysis system that is situated at the airbase consists of the following components:
- PIP photo reader CAE 71.
- DEC general purpose computer PDP 11/34 (16 bits, 64 K words MOS memory).
- DEC LA-36 printer .
- Lear Siegler ADM-42 visual display unit.
- [illegible] disk units ([illegible] M bytes).

The selection of the components of the ground-based system has been based on, among others, the following requirements:
- Floating point hardware (accuracy and speed of calculation program).
- High order language with real-time facilities (Basic +) (easy accessible and short time for development).
- Interactive programming facilities (editing activities).
- System reliability (mean time between failures is 1000 hours).
- Standardization in the RNLAF.
- Costs.

With the aid of the photo-reader the developed film negatives are projected on a high resolution screen. The once measured reference points (terrain features) in the target area, visible on the projection, are digitized and sent to the computer. On the basis of this information together with briefing data (meteorological conditions, aircraft weight and configuration etc. given by the pilot) and data stored on the magnetic disk (specifications of camera, target, etc.) the analysis of the attack is carried out.

The results of the attack run are displayed on a printer output (Tab.1).

All run results are stored on disk to provide a statistical overview of the results in relation to pilot, weapon, target etc. (Tab.2).

TABLE 1

DIAS RUN RESULTS

SORTIE: RUN: DATE:
PILOT: FLIGHT POSITION:
AIRCRAFT: K- FLAPS: CG: FUEL: (lbs)
CAMERA: TARGET:
WEAPON: DELAY TIME: (s) SIGHT SETTING: (mils)
WIND: (deg)/ (kts) CODE: QNH: (mb) OAT: (°C)

DELIVERY PARAMETERS:	PLANNED	ACTUAL	ERROR
TAS (kts)			
IAS (kts)	______		______
DIVE ANGLE (deg)			
HEIGHT (ft)			(%)
SLANT RANGE (ft)	______		______
NORMAL LOAD (g)			

PITCH ANGLE (deg): BANK ANGLE (deg): ANGLE OF ATTACK (mils):
HEADING (deg): TRACK (deg):
PIPPER POSITION: LONGITUDINAL (mils): LATERAL (mils):
IN PHOTOGRAPH LONGITUDINAL (%): LATERAL (%):

[illegible] IMPACT POSITION: RANGE ERROR (ft) :
DEFLECTION ERROR (ft) :
WEAPON TIME OF FLIGHT (s):
TARGET REF. POINTS NOT ACCEPTED: PHOTO 1: PHOTO 2:
AIRCRAFT REF. POINTS NOT ACCEPTED: PHOTO 1: PHOTO 2:
(CONSIDERING HEADING, TRACK, BANK ANGLE AND CROSS WIND A SIDE-SLIP WAS PROBABLY INTRODUCED)
(ACTUAL DELIVERY PARAMETERS OUT OF BOMB TABLE RANGE)

TABLE 2

STATISTICS OF DIAS RUN RESULTS

DATE OF PRINTOUT: OPERATOR: VERSION:

SELECTED PARAMETER	VALUE	LOWER BOUND	UPPER BOUND
DATE			
SORTIE NUMBER			
PILOT CODE			
FLIGHT POSITION			
AIRCRAFT CODE			
CAMERA CODE			
TARGET CODE			
WEAPON CODE			
PLANNED TAS (kts)			
PLANNED DIVE ANGLE (deg)			
PLANNED HEIGHT (ft)			
LIFE DELIVERY (NO/YES)			

DELIVERY PARAMETERS (NUMBER OF RUNS =)	MEAN	ST.DEVIATION
TAS ERROR (kts)		
DIVE ANGLE ERROR (deg)		
BANK ANGLE (deg)		
NORMAL LOAD (g)		
HEIGHT ERROR (%)		
SIMULATED IMPACT (NUMBER OF RUNS=)		
RANGE ERROR (ft)		
DELFLECTION ERROR (ft)		
ACTUAL IMPACT (NUMBER OF RUNS=)		
RANGE ERROR (ft)		
DEFLECTION ERROR (ft)		
DUD (NUMBER OF RUNS=)		
DUD - RATE = %		

4. SYSTEM EVALUATION

4.1 General

After having carried out an investigation about the feasibility of the photogrammetric method to be applied for DIAS (1), the analysis technique was tested. For the system performance tests, life deliveries on a weapon range were executed (2). Use was made of an instrumented NF-5A test aircraft (3) equipped with a DIAS airborne system (prototype).
The processing of data (photographs) and the data analysis were carried out with the aid of photoreader and computer facilities of the NLR.

In order to establish the airworthiness of the airborne system, tests on the ground as well as during flight were carried out.

4.2 System performance

4.2.1 General

Before DIAS was realized an investigation was carried out to establish the feasibility of such a system. During this study the analysis technique, based on a photogrammetric method was developed. Attention is paid to the analysis method applied in DIAS and information is given about the accuracy of the system. Finally the performance flight tests carried out, to verify the analysis technique are described.

4.2.2 Calculation technique

As already mentioned before mission planning data, available through the pilot, constant system data, stored in the data base of the ground-based system and the information available from two photographs, taken of the target area at the moment of simulated weapon delivery are needed to determine the attack performance.
Table 3 gives an overview of these data.
In the following the analysis method is described in general terms.

The photogrammetric method.

The photogrammetric method, used to determine the position and attitude of a camera with the aid of a photograph, taken of a defined object, is based on the fact that there is only one position from which this object can be seen the way it is displayed on the photograph.
Expressed in mathematical terms:
the camera has to be translated as well as rotated such that the image vectors on the photograph, transformed to the object co-ordinate system, cover completely the object vectors in length and direction.
More background information about the mathematical procedures is given in (4),(5) and (6).

Camera attitude with respect to the aircraft.

The bottom of the forward part of the aircraft

TABLE 3

Data needed for the analysis of a DIAS mission

subject	mission planning data	data base	mission data
aircraft	- tail number - configuration - fuel remaining over target	- position of reference points and camera - configuration data - gun sight position	
target	- target number	- position of reference points including target - target altitude above mean sea level	
camera	- camera number	- focus - measures of negative - time period between photos	- 2 photos
weapon	- weapon type - fuse arming delay time setting	- ballistic table - maximum fuse arming delay time setting - positive tolerance fuse arming delay time setting	
meteorological* conditions in target area	- wind - temperature - pressure (QNH)		
planned delivery data	- dive angle** - airspeed** - release altitude** - sight depression of gun sight		

* Normally forecast data are available.
In case of life deliveries on a weapon range actual data are taken.
In case the pilot experiences other wind conditions than forecasted, and he adjusts his planned delivery parameters, the changed wind conditions will be taken for the analysis.

** These data are not needed for the analysis but are used to indicate pilot's errors.

fuselage has been provided with 4 (aircraft) reference points of which the positions together with the camera position have been measured once in relation to the aircraft co-ordinate system. These reference points, visible on the photograph, are used to calibrate the camera attitude for each attack run with the photogrammetric method.

Position and attitude of the aircraft with respect to the target.

Every DIAS target is selected by the requirement of having terrain features which can serve as target reference points. The position of each point is measured once (infrared theodolite) in relation to the target. With these reference points, visible on the photograph, the position and attitude of the camera and thus of the aircraft can be determined. Having analyzed the photograph, taken at the moment of weapon delivery, the following release conditions are already known:

- release altitude
- heading of aircraft.

Together with the second photograph, taken 340 ms after weapon delivery, the following parameters are determined:

- ground speed
- ground track.

Angle of attack, dive angle and true airspeed.

Because of the fact that the dive angle is one of the most critical delivery parameters much attention is paid to calculate this variable as accurate as possible.
During the performance flight tests it appeared that the calculation of the dive angle by using both known positions, was too inaccurate because very often a curved flight path was flown.
For that reason the dive angle is calculated as a function of the angle of attack, true airspeed

(function of ground speed and dive angle), meteorological and aircraft data and taking into account pitch rate effects.
The dive angle calculation is executed, using an iterative process because both angle of attack and dive angle are unknown.
Note: If the wind information is incorrect the true airspeed will also be incorrect. But, as the pilot plans (and executes) the attack with the wind information available, the results are also based on this information.

The normal load factor.

The normal load factor is derived from the true airspeed and angle of attack taking into account pitch rate effects.

Side-slip indication.

In case the aircraft ground track and heading are not identical, the difference may be caused by either drift or slip or by both.
If the difference does not match with the cross wind component and aircraft speed it is concluded that the pilot initiated a side-slip condition. This is indicated in the run results.

Pipper position.

The pipper position in the ground-based reference frame, at the moment of weapon delivery is calculated by using the position of the gun sight with respect to the aircraft reference frame, the aircraft position in relation to the target and taking into account the angle of attack and the sight depression set by the pilot.

Weapon impact position and weapon time of flight.

For the calculation of the (nominal) impact position and time of flight, use is made of ballistic weapon tables, also used by the pilots to plan the delivery.
In case the actual value of a delivery parameter is not available in the table, an interpolation is performed, using a second degree polynomial.

4.2.3 Accuracy

In this section the most important error sources of the system are described. Furthermore an experiment carried out, to determine the errors and their influence on the results are discussed. Finally the overall accuracy will be given.

The most important error sources are:

- measurement of target- and aircraft reference points
- photo-camera
- read-out of photographs
- delay time of weapon release.

Because the system is used in combination with a flying aircraft some system errors are caused by motion only. For that reason distinction is made between "dynamic" and "static" errors.

"DYNAMIC" ERRORS

Shutter time of camera.

The nominal shutter time of the cameras in use is 2 ms which is tested for each camera. This value appears to be correct.
During the exposure of the film the aircraft covers a certain distance in longitudinal direction.
Furthermore the left hand side of the photograph is exposed 1 ms earlier than the right-hand side causing a position shift of reference points on the left-hand side with respect to the points on the right-hand side.
Taking into account an aircraft speed of 500 kt and a slant range of 2000 ft to the target, this shutter time causes a position error of about 3 ft and an attitude error of about 0.02°.

Elapsed time between both photographs.

The electronics which control the time period between both photographs is adjusted to 340 ms.
Tests on the ground as well as in flight showed that the maximum deviation is 1 ms.
Based on an aircraft speed of 500 kt and a timing error of 1 ms a position error of about 1 ft is caused.

Delay time of weapon release.

With flight tests during which actual deliveries were carried out the delay times between "pickle" moment (pilot's action) on the one hand and the real- and "DIAS" release moment (first photograph) on the other hand were measured. The "DIAS" release occurred 33 ms after "pickle" while the actual release took place 25 to 40 ms after "pickle".
Taking into account both the camera- and actual weapon release time delay a difference of 10 ms exists.
Based on that difference and an aircraft speed of 500 kt an impact error of about 8 ft is caused.

"STATIC" ERRORS

Measurement of target- and aircraft reference points.

The positions of the target reference points (geographical features) with respect to the target are measured with the aid of an infrared theodolite. The accuracy of the measurements is within 10 cm for the relevant distances.

The positions of the aircraft reference points with respect to the photo-camera are measured with the aid of a theodolite. The measurements are carried out with the aircraft levelled (also used for calibration of the gun sight). The accuracy of the measurements is within 1 mm.

Camera focus.

The focal distance (50 mm nominal) of each camera in use is calibrated with an accuracy of o.1 %.

Lens distortion.

To avoid the influence of lens distortion (edges of the lenses) reference points visible on the edge of the photograph (within 10 % of the width) are not used for the analysis.

Read-out of photographs.

The photographs (negative) is enlarged 14 times on the photo-reader screen which means that the projection measures are about 34 x 50 cm.
Read-out tests showed that well defined reference points, visible on the photograph can be read out with an accuracy of 0.2 mm.
The linearity of the photo-reader which has been

tested, is excellently.

Experiment.

In order to establish the total "static" error an experiment was carried out during which with a calibrated camera (fixed position) photographs were taken of an area with 8 *reference points.* The greatest distance between the camera and the reference points was about 2000 ft. With the aid of the DIAS calculation process the camera position and attitude were determined. This experiment showed a maximum position error of 4.5 ft (standard deviation 1.5 ft) and maximum attitude errors of less than 0.1^{o} (standard deviation 0.03^{o}).

OVERALL ACCURACY

Taking *into account* the experimental "static" errors as well as the "dynamic" errors (except for "delay time of weapon release") it is concluded that the overall accuracy of the DIAS position and attitude determination process is given by 8 ft and 0.1^{o} (95 % confidence level) respectively.
Based on the position error and the time period of 340 ms between both photographs the maximum airspeed error (95 % probability) is 10 kt.

Making use of the ballistic tables available and taking into account:
- the position-, attitude- and airspeed errors mentioned before
- a less accurate calculation of the angle of attack and dive angle (accuracy is 0.15^{o})
  and furthermore the variation of:
- weapon characteristics (weight, drag)
- aircraft characteristics (e.g. lift curve slope)
- meteorological data
- delay time between "DIAS" and actual release moment

the following impact errors* (circular error probable; CEP) were expected during the performance flight tests to be carried out:
- bombing CEP = 50 ft
- gunnery CEP = 55 ft
- rocketry CEP = 60 ft.

<u>4.2.4 Performance flight tests</u>

In order to evaluate the DIAS performance, flight tests were carried out with an instrumented NF-5A test aircraft of the RNLAF (2).
During the program 15 missions, consisting of in total 53 attack runs were carried out on a weapon range. Deleveries were made with 20 MK-106's, 28 BDU-33B's and 5 MK-82 R's.
A number of 8 BDU-33B's were delivered under so-called "bunting relase" *conditions* (large deviations from "steady state" conditions, especially side-slip).

In the following, information needed for a good understanding of the flight tests results obtained, is given:
1. The target area was provided with 9 reference points equally distributed over the area. All points were (always visible) taken into account for the analysis.
2. The actual impact positions were measured *accurately.*
3. Four photographs within 1 s were taken, to determine the best time interval for future DIAS photographs.
4. **The release conditions varied as follows:**
   - dive angle : 0 - 15 deg
   - slant range : 1200 - 2300 ft
   - release height : 170 - 630 ft
   - airspeed : 415 - 500 kt
   - normal load : 0.9 - 1.4 g.
5. Use was made of actual meteorological data.

From the performance flight tests the following conclusions can be summarized:
1. The circular error probable (CEP) and standard deviation(S) of the differences between the actual and predicted (DIAS) impact *positions,* were:
   a. BDU-33B
      20 deliveries "steady state" release conditions:
      CEP = 35 ft S = 32 ft
      8 deliveries "bunting" release conditions:
      CEP = 81 ft S = 47 ft
   b. MK-106
      20 deliveries "steady state" release conditions:
      CEP = 42 ft S = 21 ft
   c. MK-82R
      5 deliveries "steady state" release conditions:
      CEP = 43 ft S = 34 ft.
2. The values of the delivery parameters *obtained* by the aircraft special PCM instrumentation system and those obtained with the DIAS method agreed within the measurement accuracy of both systems.
3. During the flight tests it was found that two *photographs taken* with a time interval between 300 and 350 ms would be most appropriate to the analysis.
4. Although DIAS is able to analyse (by approximation) non-steady state deliveries, it has to be emphasized that the results obtained in those cases must be considered with some reservation.

During the flight tests no gunnery and rocketry attacks were carried out. It is, however, expected that the impact errors for both attack types are within the accuracy mentioned in the previous section.

<u>4.3 Airworthiness of the airborne data acquisition system</u>

Ground-based tests.

To establish *the airworthiness* of the airborne system the following tests were carried out:
- altitude test
- temperature/humidity test
- vibration test
- electro magnetic interference (EMI) test.

An EMI test with the system attached to the aircraft was also carried out.
Te tests were executed for the prototype *system* as well as for the series production.

In summary it can be stated that the airborne system has withstanded the tests excellently.

Flight tests.

As is customary when new aircraft configurations are to be certified, the following subjects are investigated:
- structural strength
- flying qualities

*difference between the actual and predicted impact positions.

- flutter
- aircraft performance
- store separation.

Structural strength.

The DIAS centerline pylon nose has to withstand the loads, that will be imposed under the flight condition listed in table 4.

TALBE 4

acceleration (g)			airspeed	Mach
longitudinal	lateral	vertical	(kt)	number
± 2.5	± 1.5	-1.5/+7.2	720	1.7

The structural strength of the nose is considerably in excess of the requirement, as stiffness, necessary to obtain a good camera platform, was the dominant factor in determining the dimensions of the construction. Hence, it was decided not to evaluate the structural strength aspects by means of flight tests, but only to demonstrate structural integrity, by flying up to the limits, imposed by the aircraft itself.

Flying qualities.

The location of the DIAS nose is such, that no changes in the distribution of the lift over the wing will occur, nor will the airflow over the elevators be affected. Thus, no changes in the overal flying qualities of the aircraft due to aerodynamic effects will occur.
The mass, added due to DIAS (30 lbs) results in a shift in the centre of gravity of the aircraft, but the shift will be small (less than 0.1 % m.a.c.). Furthermore, the shift will be in a forward direction, which is beneficial for flying qualities.
Based on these considerations it was decided that a special in flight evaluation of flying qualities was not necessary. No anomalies were noted during the flights carried out so far.

Flutter.

Past experience (extensive analysis of various NF-5 configurations, verified by flight tests) has shown that the amount of mass, installed at the centerline pylon hardly affects the flutter behaviour of the aircraft. As the mass addition due to DIAS is rather low, it was decided that no flutter analysis was required.

Aircraft performance.

To establish the increase in drag number, performance measurements were carried out and compared with measurements executed during flights with the normal centerline pylon.
The results of the tests showed that the effect of the DIAS nose on aircraft performance is quantified by an increase of the basic configuration drag number (7) with 5 counts which is negligible.

Store separation.

The change in geometry of the centerline pylon will affect the air loads acting on captive stores. Thus the behaviour of these stores during separation may be affected. To evaluate the magnitude of the change measurements were carried out using the air load measuring store described in (8). The results were compared with data obtained earlier in combination with a normal centerline pylon.
The normal and lateral force components as well as yawing and rolling moment coefficients were identical in both cases.
With the normal pylon, no effect of Mach number on the pitching moment coefficient was present, whereas with the DIAS nose there is. Furthermore with the DIAS nose the moment coefficient is more nose up at low angles of attack and low Mach numbers.
With the knowledge of the separation behaviour of empty centerline tanks (low density) which is most critical (9) and because of the measured small increase of the pitching moment coefficient (0.011) caused by the DIAS nose it has been concluded that the effect of the change in local air flow due to the DIAS nose on store separation is negligible so that store separation flight limits are not affected.

Subsequently demonstration flights, covering the program of the airworthiness test flights (excluding performance items), a "cold soak", followed by flight in a warm, moist atmosphere, while operating the DIAS system, were carried out. A proper operation of the system during the most extreme conditions was demonstrated.

5. IMPLEMENTATION OF DIAS IN AN OPERATIONAL NF-5 SQUADRON

Because of the results obtained the RNLAF decided to realize DIAS for "operational" use.
For the present one NF-5 squadron of the RNLAF has been equipped with DIAS.
The squadron has at its disposal a ground-based system and 5 aircraft equipped with an airborne system.
In total 1000 DIAS sorties, consisting of 2000 to 4000 attack runs, will be carried out per year. A large number of targets are available.
Dependent on the results to be obtained during the first half year, more aircraft will be involved and other NF-5 squadrons will be equipped with DIAS.

At the moment the squadron is not yet "DIAS operational", so no more information herein can be given.

6. REFERENCES

1 van Sliedregt, J.M. — The feasibility of a delivery and impact analysis system (DIAS), based on a photogrammetric method. NLR TR 77068 C (Confidential), June 1977.

2 van Eck, M.A.A., Hofman, C.F.G.M., Nieuwpoort, A.M.H. — Evaluation of the delivery and impact analysis system (DIAS) with the NF-5A test aircraft K-3001 of RNLAF. NLR TR 78083 C (Confidential), June 1978.

3 Hollestelle, P.M.N., Alders, G.J. — The NF-5A K-3001 test aircraft. NLR Memorandum VV-77-017 L (Limited), April 1977.

4 Fletcher, R., Reeves, C.M. — Function minimization by conjugate gradients. Computer journal 1964 vol. 7(2) pp. 149-154.

5	IBM	System/360 scientific subroutine package (360A-CM-03X), version III. Programmers manual (page 225-228). Publication number H20-0205-3.
6	van Eck, M.A.A.	Theoretical background of the Delivery and Impact Analysis System (DIAS) (to be published) (Unclassified).
7	Anon.	EO-05-NF-5A-1, Aircraft Operating Instructions. Revised 23-V-1978.
8	Alders, G.J.	In-flight measurement of aerodynamic loads on captive stores. Description of the measurement equipment and comparison of results with data from other sources. NLR MP 77028 U (Unclassified), September 1977.
9	Alders, G.J., Rijzebol, K.R.	Separation of empty and partially full external fuel tanks from NF-5 aircraft at high airspeeds. NLR TR 78065 C (Restricted), May 1978.

SUMMARY OF SESSION I - NAVIGATION/ATTACK SYSTEMS TESTING

The seven papers presented in the area of "Navigation/Attack Systems Testing" seemed to indicate that, from a pure test standpoint, the current navigation testing procedures are well defined and aided by modern measurement and recording equipment. Themes, and questions, indicated that if anything remains it is the need to pre-plan complementary ground-based simulations prior to flight testing, and interest still exists in failure aspects during testing and redundancy concepts.

In respect to attack system testing, a clear trend of analytical models to help locate system problems, or reduce testing costs for scoring students during training, emerged. The analytical tie included both off-line computations and ground-based simulator work to expand the overall data base and validate the analytical methods. Questions here were directed at the details of the more advanced systems (e.g., terrain-following performance, specific display formats, etc.) rather than test methods per se.

May be it would have been better to have had a lead paper like the seventh paper which gives at the same time a general view and enough details about the subject to prepare the audience for the other papers.

In all, it was a good session with quite adequate audience participation. The attack system testing aspects appear to be only the tip of the iceberg in terms of the many sophisticated systems just coming into being. For both systems the need for a "global" view of them was emphasized, i.e., treat them as part of the total system rather than isolated sub-systems.

AIR-TO-AIR GUNNERY SYSTEMS TEST AND EVALUATION

by
John A. Wiles
U.S. Air Force Flight Test Center/SES
Edwards AFB, CA 93523
U.S.A.

SUMMARY

In the past few decades, the test and evaluation of aircraft guns and gunsights has been generally qualitative. In 1978, several years of effort at the Air Force Flight Test Center (AFFTC), Edwards Air Force Base, California, culminated in the development of a flight test system for air-to-air gunnery evaluation. The system is theoretically capable of evaluating any gunnery system installed on an instrumented aircraft. It was developed and tested using a scorable tow target system, an analysis software package, and three independent flight evaluations of current fighter aircraft. Use of this flight test system over a two-year period led to the preliminary conclusion that the combination of ballistic prediction software and tow targets with miss distance measurement capability can provide analysis results accurate to within three milliradians (reference 9-1, 9-2, 9-3). The system has operational test and evaluation application as well as use for developmental testing.

INTRODUCTION

The flight test system for air-to-air gunnery evaluation now in use at the AFFTC is composed of hardware, software, and test procedures. The hardware consists of aircraft flight test instrumentation, time code generators, film/video recorders, and airborne targets for both live and dry firing tests. The software consists of a series of computer programs for the following: (a) reading target position and gunsight parameters from film or video data, (b) reading aircraft/fire control parameters from instrumentation tape, (c) calculating trajectory data and probability of hit, P_H, and (d) printing/plotting the results. The procedures consist of ground test and harmonization, tracer line matching, air-to-ground validation firing, and envelope determination/expansion tests. This paper will cover each of these areas in sufficient detail so that an armament engineer can use the method. It will, however, be general enough for modification to accommodate the hardware and software available.

The paper begins with a short review of the gunnery lead angle problem and the gunsight types currently in use. The test equipment is then discussed. The methodology is presented in the step-by-step manner used to facilitate an orderly conduct of the flight test program. Finally, the methods of data analysis are outlined and sample results given.

BACKGROUND

Before the subject of gunnery test and evaluation is discussed, a review of the fire control problem is presented for the reader. The development of the lead angle equation is neither didactic nor rigorous. It is presented only to familiarize the reader with the author's use of terms.

Lead Angle Computation

The problem of aiming the gun in air-to-air combat is somewhat more complicated than in air-to-ground strafe. Not only is the firing aircraft (called the shooter or fighter) generally maneuvering under high g and high q (dynamic pressure) conditions, the target is maneuvering to avoid being fired upon. Therefore, the fighter pilot must consider his own maneuver as well as that of the target when attempting to aim the gun. Three factors are of primary importance when determining the lead angle required for correct gun aiming: time of flight (TOF) of the bullet from the gun to the target, line of sight (LOS) rate of the target with respect to the gun muzzle, and gravity drop.

Time of flight can be calculated by integrating the bullet state vector from the time of firing until the bullet reaches the target. Target motion must, of course, be calculated. The integration is complex and time consuming. Therefore, time of flight is generally estimated based upon range to the target, range rate, fighter velocity, muzzle velocity of the gun and a simplified drag model of the bullet. The lead angle, λ (lambda), required for correct gun aiming is shown in Figure 9-1. In order to correctly aim the gun, the fighter pilot must predict where the target will be one time-of-flight later. If the gun is fired at an angle, λ, in front of the target, collision between the bullet and target will occur in one time-of-flight. If target motion is constant, the lead angle can be approximated by dividing the range (R) into the distance traveled by the target [velocity (V_T) * TOF]:

$$\lambda \approx \frac{V_T * TOF}{R} \qquad (9\text{-}1)$$

But the target velocity is generally unknown. LOS rate (ω_T) can, however, be estimated on board the fighter. We know that:

$$\omega_T = \frac{V_T}{R} \tag{9-2}$$

Thus,

$$\lambda = \omega_T * TOF \tag{9-3}$$

Under tracking conditions, ω_T can be approximated by the pitch rate of the fighter and used in the calculation of λ. This is the simplest form of the lead angle equation. All other terms are for the improved accuracy of implementing this equation and presenting it to the pilot. In addition to correct lead angle, the pilot may need an estimate of gravity drop, especially for longer firing ranges. To illustrate, the gravity drop at a range of 300 meters (m) (TOF = .35 seconds) is only about .6m, or 2 mils. Gun dispersion would easily make up for this minor error. At 750m (TOF = 1 second), gravity drop is 5m, or 6.5 mils. Even perfect aiming on a fighter-size target could result in a complete miss. Gravity drop in mils is approximated by the expression:

$$\text{Drop} = \frac{9.8 * (TOF)^2}{2 * \text{Range}} \times 1000 \tag{9-4}$$

Again time-of-flight becomes important. There is one further complication in displaying the lead angle and drop to the pilot: his own orientation. In air-to-air combat, the pilot does not know which way is up in the world around him, only in relationship to his moving reference - his fighter. Thus, the lead angle required (which is determined in inertial space) and gravity drop (which is determined relative to the real "down") must be summed vectorially, transformed into the aircraft coordinate system (through the Euler angles), translated to the Head Up Display (HUD) which is not at the same point on the fighter as the angle sensor unit, and finally converted into HUD coordinates for display. This simple development neglects gun and HUD orientation with respect to the airframe, angle of attack, sideslip, bullet tip off, gyro precession, jump, swerve, gun location, and a multitude of errors.

Lead Computing Optical Sight (LCOS)

The LCOS is the simplest means of implementing the calculation and display of the lead angle required for correct gun aiming. The basic components of an LCOS are a two-degree-of-freedom gyroscope and a small computer (analog or digital). The underlying assumption in the LCOS equation derivation is that the fighter will be using smooth, constant body rates (in particular, pitch rate, q_b) to track the target. Tracking is an attempt by the fighter pilot to reduce the relative LOS rate of the target to zero. In other words, the ω_T of the target is exchanged for q_b. As the fighter turns, the gyro produces torque proportional to the turning rate (gyro precession law). This rate can then be multiplied by TOF in the computer and the resulting angle displayed to the pilot.

If the relative LOS rate is not zero and/or the body rate is not constant, the gunsight is not "settled". The pipper does not, therefore, indicate the predicted impact point of the bullet. (In reality, the impact point would be on a plane normal to the line of sight at a distance equal to the range to the target.) An LCOS, however, has the characteristic of always moving toward the settled, or perfect, solution. The pipper position and velocity are such that the bullet will arrive at target range at the same relative point as the pipper projection after one time-of-flight. In other words, under smooth (not necessarily tracking or constant) flight conditions, the pipper shows where a bullet fired one time-of-flight earlier would be relative to the target, at target range.

Director Sight

The director sight is considerably more complex. The target is tracked by a sensor device on the fighter, generally a radar. From the sensor measurements, relative target motion is computed. From on-board measurement units, the fighter motion is computed. By vector addition, absolute (inertial) target motion is calculated. Now, through ballistic calculations, TOF can be determined and the predicted position of the bullet at target range can be computed. The motion of the fighter and target can be predicted by integrating current target motion for one TOF. The future relative position of the bullet is displayed as the pipper. Thus the pipper represents the point at target range through which a bullet fired now will pass one TOF from now (relative to where the target will be at the same time). The pilot must only superimpose the pipper on the target and fire.

In order to predict the target position in one TOF, the assumption must be that the target state vector does not change during that time. If the computer uses a linear predictor (velocities only), this assumption can result in errors when firing at longer ranges and under dynamic conditions. Relative velocities can change rapidly in a dogfight environment. Preferably, a second order predictor will be used. Accuracy should be higher because it is more unlikely that relative accelerations will change drastically during one TOF. Even this, however, is becoming more likely. Therefore, the future may call for third order predictors. The problem with higher order predictions is the increased computer capability (memory and speed) required. A second order prediction takes approximately four times the computer capability as a linear prediction; a third order may take as much as nine times as much capability. As a compromise, second order approximations are sometimes used, although the improvement with third order can be significant.

Historic Tracer ("Hotline")

The historic tracer, or "hotline", gunsight displays a simulation of the stream of bullets to the pilot. No target position information is required for the calculations. Every point on the "hotline" represents the relative position of a bullet fired one TOF ago. The historic tracer is implemented by integrating the ballistic equations and fighter motion simultaneously. The pilot maneuvers the fighter such that the "hotline" falls across the target. One disadvantage of this sight is that the fighter pilot does not know which portion of the "hotline" to place on the target. Thus, the advantage of no target information required carries a serious disadvantage. This can be overcome, to some extent, by the range-to-target input being displayed on the "hotline". The pilot still must estimate relative target motion in order for the solution to come together at the appropriate time, one TOF after trigger squeeze.

TEST TOOLS

The tools of the trade for the air-to-air gunnery test and evaluation engineer are many and varied. The primary tools are aircraft instrumentation, target instrumentation, event recorders, analysis software, and flight folder. Ground instrumentation will not be discussed here because it is not often used. Generally speaking, ground-based radar and phototheodolite operators do not like aircraft firing guns over their heads.

Aircraft Instrumentation

Instrumentation on the aircraft generally includes measurement and recording devices for time, airborne radar parameters, aircraft flight parameters, and aircraft state vector parameters.

Time is provided by a time code generator, synchronized before flight with IRIG standard time. This time is recorded simultaneously with all instrumentation parameters so that correlation of events can take place. Time is normally recorded in Greenwich Mean Time and contains computer words for Julian date, hours, minutes, seconds, and milliseconds. Accuracy is very good (± 2 msec).

Airborne radar parameters are provided by tapping the electrical signals in the fire control computer and recording them on tape. Although many parameters are available, target range and radar look angles are considered mandatory. With these parameters, the relative target position can be determined. By differentiating these parameters, relative target velocity (and acceleration, if necessary) can be calculated. The accuracy of radar range is on the order of 10m and is one of the severest limitations on gunnery analysis. Angle accuracies are in the range of 25-50 milliradians. If the radar is the only source of target position, analysis of the results is difficult. At 600m, for example, the radar position may have an uncertainty of 15m; the gun dispersion will have a radius of only 2m. Perfect shooting by the pilot may look to the analyst like a complete miss.

Aircraft flight parameters are considered to be those measured with the pitot-static system and relative air flow vanes. Parameters like airspeed, altitude, Mach number, outside air temperature, angle of attack, and sideslip angle are the primary flight measurements. Required for analysis are true airspeed, density altitude, and angle of attack. Although sideslip is not always available, it is very important; its effect is a three mil bullet error for a .02 radian sideslip error. The accuracy requirements for airspeed and altitude are not stringent. Angle of attack errors can, however, be significant and the measurement should be accurate to within .005 radians. This accuracy is often difficult to achieve, but should be used as a goal.

The aircraft state vector as used here is its position, velocity, and angular orientation in inertial space. These parameters are generally available within the inertial navigation system (INS). The heart of the INS is a gyro/accelerometer unit which measures pitch, roll, heading, and three-axis accelerations (N-S, E-W, vertical). From these measurements, the INS computes position, velocity, and angular rates in inertial space. These parameters can also be tapped and recorded in flight. The required measurements for gunnery analysis are pitch, roll, true heading, and acceleration in the aircraft vertical axis. In the case of director sight analysis, fighter velocities and accelerations in all axes must be known. Generally the accuracies available from a standard INS are adequate for analysis, if the INS is functioning properly.

An on-board recorder is generally installed in the test aircraft to signal condition all of these inputs (plus a trigger-on signal) and record then on magnetic tape for postflight processing.

Target Instrumentation

In comparison to fire control technology, target instrumentation lacks sophistication. The primary purpose of a target is to provide an aiming point for test measurement purposes and an indication of the success of a firing pass. The best that can be expected with any reliability these days is the number of bullets passing within a certain distance of the target. There are two types of targets, scorable and non-scorable. Since gunnery targets are currently limited to the towed variety, this discussion will be similarly limited.

A scorable target widely used at this time is the Secapem 90/B with SFENA MAE-15 scorer. The Secapem is an acoustical target system. The strength of the shock wave of the passing bullet can be translated into approximate miss distance. More discussion of this calculation is contained in the Data Reduction/Analysis section of this paper. Other targets/scorers may be forthcoming in the future. The system nearest to being operational uses doppler radar scoring. Miss distance can be estimated by counting the number of doppler standing waves penetrated by the bullet. What is needed is a scoring system (not necessarily on a tow target) which will provide a time history of the state vector of each bullet fired. For now, the scorable targets are used to confirm the validity of the analysis software and to give physical results to those who distrust software.

Among the non-scorable targets are the training targets: darts, FiCATs, and banners. There are those who say these can be scored. If so, the scoring is, in general, extremely inaccurate. These targets are only valid for hit/no-hit operations analyses, not for systems error analysis. The advantages of these targets are large size or relatively low cost.

Event Recorders

Test event recorders include HUD video and/or gun cameras. The parameters available from this data source range from relative aim error and trigger-on signal all the way to a full set of analysis data including score. Primarily the video or film is used to record pipper position, target position, and trigger-on time (event). The various measurements made from the film are placed on computer cards for use during analysis. The trigger signal is used to time-correlate the film or video frames with corresponding data samples on the instrumentation tape. The accuracy of this correlation may not be good (sometimes +50 msec) and is another severe constraint to the analyst, especially during non-tracking shots.

Using film or video, the angular position of the target (relative to the fighter) can be determined very accurately, on the order of 1-2m. This method reduces the uncertainties of 15m available from radar.

Currently, film provides better resolution and greater accuracy. In the future, as video improves, more video will be recorded and used for gunnery analysis.

Analysis Software

The primary software tool used by the AFFTC analysts at present is the Air-to-Air Gunnery Assessment System (ATAGAS). This system reads the film cards and instrumentation data tape, fires theoretical bullets while the actual trigger is on, and determines the expected number of rounds which will impact within a certain target size. The results are produced in tabular form and plots. ATAGAS has not been completely verified, but has been used with a fair amount of success on several aerial gunnery projects. Once ATAGAS is shown to be working on a given project, the results can be used to provide inputs to various statistical routines to determine errors: biases, dispersion, and CEP.

A further discussion of software is contained in the Data Reduction/Analysis section.

Flight Folder

The last tool to be mentioned is probably the most vital: the flight folder. The aerial gunnery flight folder is the single place for all information gathered during pre-flight briefing, test conduct, quick-look data reduction, post-flight data reduction, and final data analysis. The aerial gunnery flight folder should contain, at a minimum:

Flight card (used by the pilot during flight)
Flight test engineer's notes
Test pilot's notes
Target scores/rounds fired
Event numbers/times
Miss distance output
ATAGAS input/output
All analysis notes, thoughts, reflections, and results

Although the flight folder may seem like an unnecessary administrative burden, it is an invaluable asset.

TEST METHODS

Now that the tools are ready to use, the test may proceed. Like many tests, air-to-air gunnery testing is conducted by starting with the simplest task and working toward the most complex. Basically, a properly conducted aerial gunnery test will contain ground tests, night/dusk tracer firings, air-ground validation, dry-fire flights, live firing flights, and gunnery envelope expansion flights.

Ground Test

Ground testing starts with an accurate wet boresight of the gun installed in the test airplane. The procedure for wet boresight is generally contained in aircraft manuals and technical orders or local operating instructions. Before boresight adjustments are made, the gun should be fired for 200-300 rounds to assure proper functioning and mount setting. Allow for at least three days to conduct this operation. It should not be hurried. Assure that the aircraft jacks are tightened down and do not slip. The alignment of the aircraft with the target is an iterative process. Once the alignment is close to that required, it is easier to move the aircraft than the target stand. Roll angle should be as close to zero as possible. Small deviations in lateral or vertical alignment, however, can be accounted for. Every attempt should be made to align the gun to within 1/2 mil of the correct point.

During the conduct of the final phases of ground testing, actual film and instrumentation data should be run to assure that the processing will be smooth and that all data will be available for reduction. In addition, the data can be checked against the analysis software for impact verification. Bullet dispersion can be measured during wet boresight.

If the aircraft instrumentation is such that measured inputs can be simulated, an end-to-end calibration should be attempted in order to determine the biases and statistical precision of the measurements. With this information, sensitivity studies of the analysis software can be made. Other tests should be conducted to measure delay times in trigger-on signals, event lights, gunfire voltage, etc.

A windscreen parallax test should be conducted on all new systems. Optics engineers can work with the armament engineers to quantify the error between target and pipper position as viewed by the pilot versus the position as recorded by the HUD film or video. If the canopy or windscreen is curved, corrections for canopy distortion can also be measured by noting differences in pipper position with and without canopy windscreen in place.

Night Tracer Firing

The next test is used to confirm the aircraft parallax constants and other software inputs. The test is to fire tracer bullets with high contrast lightin, conditions (dusk or later), to see if the analysis software and/or the gunsight accurately predict the bullet path under actual flight conditions. The analysis software will provide a bullet stream which should match the actual bullets as recorded on gun camera film. The tracer fall line should also coincide with a hotline gunsight, if incorporated; and/or pass through an LCOS pipper, if incorporated. This test does not account for range errors, but should indicate if there are parallax errors or data errors, or if the analysis software or aircraft unique constants are in error. The maneuvers performed during this test should include wind-up turns, barrel rolls, roll reversals, rudder doublets, and pitch doublets. Both the gunsight and the analysis software should predict the bullet paths to within 2-3 mils.

Air-to-Ground Firing

The third test is for analysis software checkout. Strafe runs can be analyzed for mean impact point comparisons with the reference software. Ranges at open fire have to be slightly longer than normal to allow for sufficient gun camera film over-run. The analysis software should be able to statistically predict the results of each strafe run. Strafe with a fixed, manually depressed pipper is acceptable.

Dry-Fire Flights

After all preliminary tests are accomplished, a few dry fire test flights should be conducted under relatively controlled conditions. The tests should be conducted with phototheodolite coverage of both the target and the shooter. Both aircraft should be instrumented, if possible. Sample passes from each of the planned flight test points should be accomplished. Simulated firing should take place. A simulated trigger (or other event indicator) should be recorded on instrumentation tape, film, and transmitted to a ground station for time correlation. All data should be reduced as it would be for a live firing. In addition, the phototheodolite data should be reduced to enable fighter radar input and fire control computer accuracy to be determined. Radar range is a critical parameter. Any biases determined in this test should be corrected in the fighter. If that is not possible, the corrections must be made to the analysis software so that it may accurately model the firing pass.

Live Firing

When all preliminary tests have been completed, live firing can be accomplished. Everything has to work properly for a successful mission: the target has to work, the gun has to fire, the film and data must run, the film and data must be readable, and the pilot must know what to do. Each test point should have been practiced on previous missions. Each item of test equipment should have been double checked.

The test should be conducted in an approved aerial gunnery complex. The aircraft towing the target will clear the range, deploy the target, and set up the appropriate flight pattern for the firings. The most common patterns are Racetrack (Figure 9-2), Figure-Eight (Figure 9-3), Butterfly (Figure 9-4), and Combat Dart (Figure 9-5). Sometimes special patterns are designed to fly a certain difficult test condition with a high degree of repeatability.

Planning for a live firing test must include considerations of bullet impact area, sun angle (not in the pilot's eyes or reflecting off the HUD), time available for tracking, pilot experience, and many other things. Generally, a test program is planned to begin with easy test points: medium to short range, low angle-off (or aspect angle), low closure rates, and 1 1/2 - 2 g's. Even if the pilot never gets the pipper near the target, the data can still be used. Or, if the pilot destroys the target, the data should provide system validation. And the rest of the flight can be used for practice. Depending on the test objectives, several flights with differing conditions will be needed. Low angle-off firings are relatively simple to set up; high angle-off shots will take some practice. Under relatively benign conditions, tracking is usually possible and the pilot can open fire whenever he is satisfied with the pipper placement. Under more dynamic conditions, everything is happening faster. The pilot must estimate target relative motion and anticipate an open-fire situation. The most common error made in high angle-off shots is firing too late. For example, a 1200m open fire range on a 300m/sec target requires a total lead angle of over 280 mils at 90 degrees angle off. In most aircraft, the pilot cannot even see the target under these conditions. The primary purposes of the live fire phase of testing are to validate the analysis software by obtaining results similar to the target score and to give those who distrust software a warm feeling. All data should be reduced. Passes where known data problems exist may be discarded during analysis.

Again some cautions for the engineer: in the early phases of testing, brief the pilot to put the pipper on the target until you and he/she are convinced that the gunsight is in error. Then small incremental pipper offsets should be made until hits are scored. This will allow a logical analysis of systems errors without providing false preliminary information. This analysis should be orderly. The gunsight software should not be changed until the analysis is complete.

Envelope Expansion

After the analysis software has been verified, many dry firing passes may be made to generate measures of system capability: hit probability envelopes may be constructed; system error versus flight condition plotted; probable error determined for non-cooperative, fighter-sized targets; and qualitative system evaluation performed without the hazards of live firing. All these results can be provided using the analysis software.

DATA REDUCTION/ANALYSIS

Analysis of aerial gunnery data requires an understanding of the gunsight, the target, the gun and ammunition, the fighter dynamics, the instrumentation, the tactics and scenario, the pilot, the gun camera, and the software.

Event Data

Film reading or video reading is the measurement technique usually employed when determining target/pipper angular relationships. Some fixed reference point (usually a gun cross) is chosen as the zero in both azimuth and elevation. This point must not move in the aircraft reference frame and must be visible in every frame. Also chosen is some fixed size object of known milliradian dimension, such as the reticle. The measurement process is ready to begin. The first step is to determine the size of the fixed object in reader units, or counts. For example, the number of counts from the left side of the reticle to the right side may be measured as 1430 counts. We may know that the reticle is 50 mils across. Therefore, the scale factor is 1430 ÷ 50 = 28.6 counts per 1 mil. This is assumed true throughout the film frame and for the entire flight.

When the location of the item of interest is to be measured, the reader crosshairs are placed on the zero reference point, and the counters set to zero. The crosshairs are then moved to the item of interest (pipper, target, lagline, etc) and the new counts noted (usually punched on computer cards). If the zero reference point were the gun cross and the readings to the pipper were azimuth +1000 and elevation -3000, the conversion factor could be applied and the actual location of the pipper relative to the gun cross could be computed as 35 mils right, 105 down.

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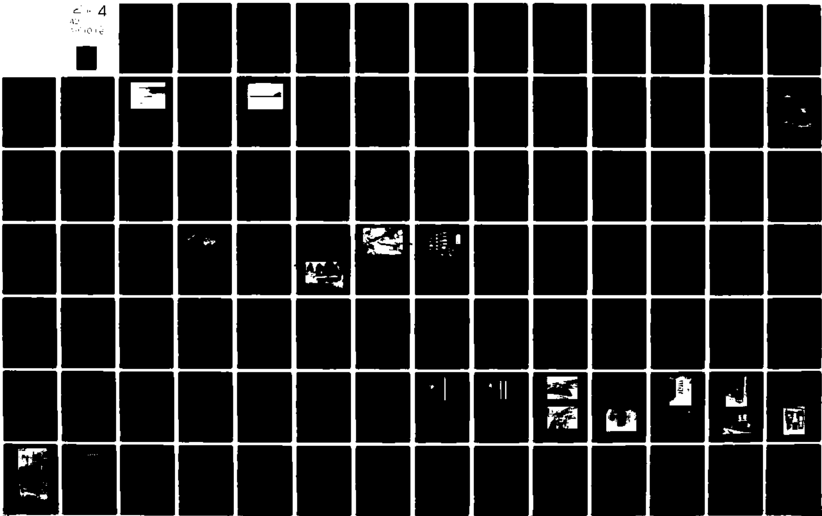


The measurements themselves are normally made by technicians, not the armament engineer. The film given to them must, therefore, be accompanied by a complete description of what frames are to be read, what is to be measured, and what format is required for the output information. All these things are preplanned by the analyst when he/she previews the gun camera film.

Target Scoring Data

One target system currently available which provides bullet miss distance is the Secapem. A discussion of the scoring mechanism and the miss distance method is contained in AFFTC-TR-78-26, Secapem 90/B and SFENA MAF-15 Target System Evaluation (Reference 9-3). The method described in the referenced report is summarized in Appendix 9-A. A computer program can easily be written to reduce Secapem data. The input file should contain target speed, altitude, and the measurement of shock wave strength transmitted from the target. The measured strength of each shock wave can be converted into miss distance, using the method contained in the Secapem report and printed out. Some statistics can also be computed and printed for each pass.

Analysis Software

The heart of aerial gunnery analysis is a software system currently known as ATAGAS. Basically ATAGAS takes the fighter conditions at trigger on and fires a theoretical bullet. After one time of flight, it compares the computed position of the bullet with the present position of the target. Various tabulations, plots, and statistics are available as output. ATAGAS is described in its own documentation (Reference 9-4).

The inputs consist of film/video from a reader and an instrumentation tape of onboard data. The tabulated outputs available include all the inputs; theoretical bullet positions; single-shot probabilities of hit; angle-off; and various target-pipper, target-bullet, and pipper-bullet relationships. The output format is extremely flexible and can be easily changed to suit the analyst. The computer plots consist of any combination of the variables above plotted against time, or cross-plotted against each other. These are also readily changed by the analyst.

The most useful outputs are the flight conditions of the fighter, relative target conditions, aiming errors, and bullet positions. Only aiming error and theoretical bullet position make helpful plots. If there are serious gunsight or data problems, all varieties of printed and plotted outputs are available to aid in the discovery of anomalies. Generally, the assistance of a computer programmer familiar with ATAGAS will be needed to obtain other data items.

A final piece of output data, which is very useful, is the expected result, or P_H. This is generally composed of the number of expected rounds fired and the number of expected hits within a certain size target. Although the discussion here was based on the use of ATAGAS, any general purpose ballistic simulation would suffice so long as the input formats were compatible with the instrumentation and the output format were usable by the analyst doing the evaluation.

Gunsight Simulation

A very useful tool in gunsight error analysis is a simulation of the actual sight algorithm and/or of a "perfect" sight of the same type. This is especially beneficial in LCOS analysis. The gunsight algorithms are coded into FORTRAN and run with the instrumentation data as input. These algorithms are run concurrently with ATAGAS. The output is sight display commands (azimuth, elevation, TOF, etc). The theoretical aimpoint (the simulation) can be compared with the actual pipper position as recorded on film for display errors or sight inadequacies (too many assumptions, approximations, etc).

Error Analysis

In air-ground accuracy analysis, many measures of merit are available. Onboard measurements can be compared to ground based data. The position of the target is surveyed. The actual impact of the weapon is surveyed. Aerial gunnery accuracy analysis is somewhat more subjective and less well-defined. In fact, sometimes it is only an educated guess.

There are primarily three errors: total, aiming, and system. Total error is defined as how far the bullet missed the target. This is the bottom line, the actual measure of success. It is mathematically defined as:

$$\vec{TE} = \vec{BP} - \vec{TP} \tag{9-5}$$

where the arrow refers to the vector quantity and TE is total error, BP is the bullet position, and TP is the target position (see Figure 9-6). All variables must be in the same reference system (generally HUD coordinates) and must be specified at the same time: bullet arrival time. Total error is not necessarily the largest error. The bullet position out of ATAGAS is assumed to be mathematically correct. Before this assumption can be made, however, the ATAGAS deck set-up for the specific test vehicle must be verified by flight test points. This verification is discussed in a later paragraph.

Aiming error is simply defined as how far the pipper was from the desired aimpoint. The error may be due to intentional offset to achieve better results; it may result from a lack of pilot skill; or it may exist because the pipper is too difficult to track with, perhaps due to mechanization. Aiming error is mathematically defined as:

$$\vec{AE} = \vec{PP} - \vec{TP} \tag{9-6}$$

where AE is aiming error, PP is pipper position, and TP is target position (see Figure 9-6). Again, all variables must be in the same reference system and specified at the same time: bullet firing time. This error can be determined directly from the film.

System error is defined as how far the bullet missed the point that the pipper indicated that it would hit. This error is a combination of errors in the on-board sensors, errors in the algorithms and/or calculations, and display/presentation errors. System error is mathematically defined as:

$$\vec{SE} = \vec{BP} - \vec{PP} = \vec{TE} - \vec{AE} \tag{9-7}$$

where SE is system error and the other symbols are as before (see Figure 9-6). The variables in this instance may be in different coordinate systems and certainly are at different time. Thus the bullet position generally must be converted back into the firing reference frame to compare it with the pipper at bullet firing.

If it is possible to measure any of the subsystem errors, they can be vectorially removed from the system error. For example, if a flight computer printout shows commanded pipper position, the difference between this position and the pipper position read from the film would be display error. Any difference between commanded pipper position and the position calculated by the equivalent ground simulation would be a mechanization error. Differences between ground simulations and "perfect" gunsight algorithms should be considered as sight implementation errors: the errors due to approximations and assumptions.

As many of the above errors as are measurable should be determined for every valid gunnery pass. In fact, these data points may be taken for each data sample during a pass (usually 20 per second). The only restriction is that the measurements be taken when the pilot have the intent to fire (trigger on indication). During a 50 round - 20mm burst, for example, there will be approximately twelve sets of error measures.

Statistical Analysis

Individual error data points do not have as much meaning as do several points properly combined. Likewise, individual error measures cannot be meaningful unless all are present. One exception is total error. If a system error is such that a pilot can compensate empirically by offsetting his aim, the sight may still be usable.

Ordinarily, several data points are lumped together in major categories and statistical analysis is performed. The categories of firing range, range rate, target g, and aspect angle are the most meaningful. A representative group of firing parameters may be picked. For example:

Range =	600m ($\pm$60m)
Range Rate =	75m/sec ($\pm$15m/sec)
Target g =	30m/sec/sec ($\pm$5m/sec/sec)
Aspect Angle =	25 deg ($\pm$5 deg)

Assume there are eight valid passes within these parameters, each having twelve data points. There would be a set of data like this:

Pass/Bullet	TE az/el	AE az/el	SE az/el
01/01	+2.4/-5.0	-1.8/-3.2	+4.2/-1.8
01/02	+2.0/-4.8	-1.4/-2.7	+3.4/-2.1
.	.	.	.
.	.	.	.
.	.	.	.
.	.	.	.
08/12	-7.2/+1.8	-11.8/+4.6	+4.6/-2.8

The mean error and standard deviation for both azimuth and elevation could be calculated for each error type. Means and standard deviations for total error give a measure of the overall effectiveness in putting bullets on target. Aiming error statistics indicate ease of tracking. System error indicates accuracy of the sight itself. Circular error probable (CEP) can be calculated from the standard deviation. Total CEP and system CEP can be meaningful numbers to operations analysts and contract monitors. Since the combined sample is large (96), the confidence in the results is reasonably high.

During an earlier section, there was discussion of sensitivity analysis. After a result is determined, a certain confidence must accompany that result. One way to indicate that confidence is with a one sigma uncertainty. For example, assume that the system bias was calculated to be +7 mils in elevation and the root-sum-square of the effects of instrumentation errors was 2.5 mils. There is therefore, a good chance that the true value of bias could be as low as +5 mils or as high as +9 mils. Similarily, if the total CEP were 15 mils, the true value may lie somewhere between 13 and 17 mils.

Also mentioned earlier was verification of ATAGAS. One good way to do this is by comparison with physical evidence. The night tracer firing is an excellent source of physical evidence. Another form of physical evidence is the Secapem target. A general procedure which was successfully used for determining correlation between ATAGAS and Secapem is as follows:

1. For each valid pass (all data good), compute the Secapem percentage hits by dividing the number of bullets within 5m by the actual number of rounds fired.

2. For the same passes, compute the ATAGAS probability of hitting a target with a 5m radius by dividing the expected number of hits by the expected rounds fired.

3. Each pass is now a data set: one ATAGAS percentage and one Secapem percentage. Plot each data set on a graph to see if the data are good. Perfect data would lie along a 45° line (see Figure 9-7). Calculate the correlation coefficient. If x = Secapem score and y = ATAGAS score, then a least-squares fit of the data sets, minimizing y, could be determined from the equations:

$$\hat{y} - \bar{y} = M\,(x - \bar{x}) \tag{9-8}$$

$$M = \frac{\Sigma (x-\bar{x})(y-\bar{y})}{\Sigma (x-\bar{x})^2} \tag{9-9}$$

where $\hat{y}$ is the estimated value and the bar over a variable represents the mean for the set. The correlation coefficient, r, is then:

$$r = \frac{\Sigma (x-\bar{x})(y-\bar{y})}{\{\Sigma (x-\bar{x})^2 \Sigma (y-\bar{y})^2\}^{1/2}} \tag{9-10}$$

The nearer to unity this value is, the better the correlation. A statistical table of "test for significance" can be consulted. It can be entered with r and the number of samples to determine the likelihood of accidental agreement. If the agreement is good, (hypothesis is valid) then ATAGAS can be used for all the error determinations indicated above. If there is inadequate agreement, ATAGAS constants, data inputs, and time correlation errors should be checked and corrected if possible. If no agreement can be found, old-fashioned qualitative methods will have to be used. Another possible method of data correlation is by comparing the hits scored by the target system to the probability distribution of expected hits from the analysis software.

SAMPLE RESULTS

Dusk Tracer

The verification of tracer fall line by comparison to ATAGAS output is a very simple, but necessary process. Many facts can be discovered. The dusk tracer mission (usually only one is required) can verify the instrumentation and data reduction compatibility, along with the correctness of the film reading procedure. This mission has the additional benefit of assuring that all the test vehicle hardware is functioning properly. The data are usually plotted in HUD coordinates for one instant in time: "hotline" (if available) or LCOS, actual tracer rounds, and ATAGAS theoretical bullets (see Figure 9-8). Comparison can be made between the actual bullets and the ATAGAS bullets to assure that the instrumentation and physical aircraft constants are correct. The actual bullets should be within 2-3 mils of the theoretical bullets. The position of the actual bullets can be compared to the "hotline" in the HUD. During a recent flight test program, a bad yaw rate gyro was discovered with this phase of testing. A similar test is possible with an LCOS. Under stable flight conditions, the actual bullets should appear to pass through the LCOS pipper at some preset manual range or the default value for target range (with no radar lock-on). Unless there were system errors, the above testing generally resulted in agreement between ATAGAS and actual tracer rounds with a 1.8 milliradian standard deviation. The figure is nearly identical to the standard deviation for 20mm gun dispersion.

ATAGAS Accuracy

The accuracies and uncertainties in the ATAGAS solution (theoretical bullet position) were studied separately. For normal flight test firing conditions, ATAGAS output errors have a combined standard deviation of less than three mils (Reference 9-1, 9-2). The errors in ATAGAS output are primarily due to instrumentation and timing correlation inaccuracies. Results from these reports have been extracted and are contained in Appendices 9-B and 9-C.

Live Fire

After a live fire mission, results include quick-look (target scores and qualitative film assessment) and post-flight (miss distance, film reading, and ATAGAS). The quick-look data can be used to provide a gross analysis of gunsight effectiveness. For example, on a recent flight test mission, the first firing pass was made with the pipper on the target - virtually perfect aiming - and yet the target registered no score. On the next firing pass, the pilot placed the pipper approximately five meters behind the target. The result was the scoring of 21 hits (out of 23 rounds fired). Since the firing range was 670m, it appeared from quick-look data that there was an +8 mil bias in the gunsight. Post-flight analysis using ATAGAS showed that the bias was +6 mils. The advantage was that the quick-look data were available within 24 hours, whereas the more accurate (and more complete) data were not available for five days. Post-flight data reduction, although it does take longer, can provide the data analyst with significantly more information. The results can be plotted against time on one graph to see trends and to assure agreement (Figure 9-9). In this figure, one can see how all data sources provide similar results when functioning properly. Supporting data for this figure are contained in Table 9-1.

Finally, the target data and ATAGAS data have to be compared to assure that there is, in fact, agreement between the two independent measurement sources. During the early phases of testing with the methods described in this paper, 32 valid gunnery passes were compared. What is meant here by a valid gunnery pass is one where all instrumentation was performing properly so that the data could be processed. The results are shown in Figure 9-7. Supporting data are contained in Table 9-2. The results showed a correlation coefficient of 0.75 with a significance of greater than 99%. Based on this early result and the agreement between qualitative and analytical work on gunsight accuracies, the methodology was pursued.

CONCLUSION

With the methods presented here, three air-to-air gunnery test programs were completed, evaluated, and documented. The techniques are continuing in use on two additional programs with two more ready to begin in the immediate future. The methods are flexible. As new target systems become available, they can readily be used. As more sophisticated software becomes available, it can take the place of ATAGAS. As a new gunsight becomes available, one can determine how much better it really is (if it is). Additionally, using these methods, all these improvements can be proof-tested while they are in use, since the test methods cross-check themselves.

Qualitative assessment of aerial gunnery may still have its place as fighter pilots brag to each other about their prowess. However, quantitative results, such as can be obtained through these techniques, are mandatory for test and evaluation.

REFERENCES

9-1. Johnson, Gary G., Accuracy Study for the Air-to-Air Gunnery Assessment System (ATAGAS) Ballistics Model, AFFTC-TIM-79-1, Air Force Flight Test Center, Edwards AFB, California, July 1979.

9-2. Wiles, John A., Major, U.S. Air Force Reserve, Air-to-Air Gunnery Scoring Parameters Evaluation, AFFTC-TIM-79-2, Air Force Flight Test Center, Edwards AFB, California, August 1979.

9-3. Killebrew, Kerry E., Captain, U.S. Air Force, and John A. Wiles, Secapem 90/B and SFENA MAE-15 Target System Evaluation, AFFTC-TR-78-26, Air Force Flight Test Center, Edwards AFB, California, January 1979.

9-4. Staggs, Darcy and Jud Watts, ATAGAS: A Flight Test Data Reduction System for Air-to-Air Gunnery Assessment, AFATL-TR-79-34, Armament Systems Inc., Eglin AFB, Florida, April 1979.

APPENDIX 9-A

EXTRACT FROM REFERENCE 9-3

Miss Distance Determination

The data reduction method !for determining bullet miss distancel was derived using technical information supplied by the manufacturer from French flight test (see Figure 9-A1). A nominal set of flight conditions was chosen to represent a baseline for miss distance calculations. The conditions were 350 KIAS and 20,000 feet target airspeed and mean sea level altitude, respectively. The miss distance vs. pulse duration curve was approximated with an exponential:

$$\text{Miss Distance} = 6.465 \times e^{(-.307 \times \text{Pulse Duration})} \qquad (9\text{-A1})$$

Empirical adjustments were made for altitude and airspeed variations from the nominal. The resulting data reduction equation was:

$$\text{Miss} = 6.465 \left\{ e^{\left[(-.307PD)\right]} + \left[\frac{(0.5 - 0.1PD)(350-V)}{350}\right] + \left[\frac{(1.3 - 0.25PD)(20000 - ALT)}{10000}\right] \right\} \qquad (9\text{-A2})$$

where

Miss = miss distance in meters

PD = pulse duration in milliseconds

V = velocity in KIAS

ALT = pressure altitude in feet MSL

APPENDIX 9-B

EXTRACT FROM REFFRENCE 9-1

Accuracy Assessment

Most effective fighter aerial combat bullet times of flight occur between 0.5 and 1.5 seconds. However, in the case of rear hemisphere firing (e.g., bomber defense), a 2 second time of flight is reasonable. In this light, the horizontal and vertical angular differences between ATAGAS and Eglin 6DOF (six degree-of-freedom model derived by the Air Force Armament Laboratory at Eglin AFB, Florida) trajectories were computed for 0.5, 1.0, 1.5, and 2.0 seconds time of bullet flight. The following accuracy statements of the ATAGAS ballistic model (TRACER) can be made for the conditions evaluated:

a. 0.5 and 1.0 seconds time of flight accuracy assessments:

(1) ATAGAS was within .15 mil of Eglin 6DOF model in the horizontal plane at 0.5 seconds time of flight (no angle of attack or sideslip).

(2) ATAGAS was within .25 mils of Eglin 6DOF model in the horizontal plane at 1.0 seconds time of flight (no angle of attack or sideslip).

(3) For all practical purposes, ATAGAS matched the Eglin 6DOF model exactly in the vertical plane at 0.5 and 1.0 seconds time of flight (no angle of attack or sideslip).

(4) ATAGAS, at the worst, was within 0.8 mils of the Eglin 6DOF model in the horizontal plane at 0.5 or 1.0 seconds time of flight when an initial angle of attack or sideslip was other than zero. On the average, ATAGAS was within 0.5 mils of the Eglin 6DOF model in the horizontal plane at 0.5 or 1.0 seconds.

(5) ATAGAS, at the worst, was within 0.7 mils of the Eglin 6DOF model in the vertical plane at 0.5 or 1.0 seconds time of flight when an initial angle of attack or sideslip was other than zero. On the average, ATAGAS was within 0.5 mils of the Eglin 6DOF model in the vertical plane at 0.5 or 1.0 seconds.

(6) On the average, ATAGAS to Eglin 6DOF horizontal angular difference was less than 0.25 mils at 0.5 or 1.0 seconds time of flight for all the cases examined. The average vertical angular difference was less than 0.15 mils for all the cases examined at 0.5 or 1.0 seconds time of flight.

b. 1.5 and 2.0 seconds time of flight accuracy assessments:

(1) ATAGAS was within 0.5 mils of Eglin 6DOF model in the horizontal plane at 1.5 seconds time of flight (no angle of attack or sideslip).

(2) ATAGAS was within 0.6 mils of Eglin 6DOF model in the horizontal plane at 2.0 seconds time of flight (no angle of attack or sideslip).

(3) ATAGAS was within 1.0 mils of Eglin 6DOF model in the vertical plane at 1.5 or 2.0 seconds time of flight (no angle of attack or sideslip).

(4) ATAGAS, at the worst, was within 1.0 mils of Eglin 6DOF model in the horizontal plane at 1.5 or 2.0 seconds time of flight when an initial angle of attack or sideslip was other than zero. On the average, ATAGAS was within 0.5 mils of Eglin 6DOF model in the horizontal plane.

(5) ATAGAS, at the worst, was within 0.75 mils of Eglin 6DOF model in the vertical plane at 1.5 or 2.0 seconds time of flight when initial angle of attack or sideslip was other than zero. On the average, ATAGAS was within 0.5 mils of Eglin 6DOF model in the vertical plane.

(6) On the average, ATAGAS to Eglin 6DOF horizontal angular difference was less than 0.40 mils at 1.5 or 2.0 seconds time of flight for all the cases examined. The average vertical angular difference was less than 0.20 mils for all the cases examined at 1.5 or 2.0 seconds time of flight.

APPENDIX 9-C

EXTRACT FROM REFERENCE 9-2

Estimate of Uncertainty

The errors found through parameter variation in this study were grouped by similarity and fit with the following empirical relationships. The approximate mil errors are:

ERROR SOURCE	1σ EFFECT (mils)
(1) Combined independent errors	1.6
(2) Combined errors dependent on time of flight (T_f)	$2/3 \times (T_f)^2$
(3) Target position errors	$\sqrt{4/3^{2} + [12 \times (\text{LOS rate})]^{2}}$

The errors above can be determined and combined for any flight condition to estimate ATAGAS uncertainty.

For example, in a center-of-the-envelope pass, typical of those flown at the AFFTC, the ATAGAS uncertainty would have a one sigma value of approximately 2 1/2 mils. With a timing correlation error, the uncertainty might be greater. The uncertainty could be computed as follows:

Assume co-speed, 750 fps, (230m/sec), medium altitude (20,000 ft) (6000m), target and fighter range, 2000 ft (600m), angle-off, 20°; tracking shot. Therefore, LOS rate is approximately .10 radians per second and T_f is 0.64 seconds.

ERROR SOURCE	CALCULATION	APPROXIMATE 1σ EFFECT	$(\sigma)^2$
1	--	1.6	2.6
2	$2/3 \times (.65)^2$	0.3	0.1
3	$\sqrt{(4/3)^2 + (12 \times 0.1)^2}$	1.8	3.2
TOTAL	$\sqrt{5.9}$	2.4	5.9

TABLE 9-1

Data for Figure 9-9

ELAPSED TIME (msec)	RADIAL ERROR (mils) PIPPER	ATAGAS	SECAPEM
0*	18.2	16.1	11.2
2	--	--	9.7
5	16.1	11.7	10.4
7	--	--	10.0
8	--	--	10.9
10	14.0	10.0	--
11	--	--	11.2
12	--	--	10.0
14	--	--	13.4
15	11.1	8.7	9.8
16	--	--	11.0
17	--	--	9.0
18	--	--	7.8
19	--	--	8.7
20	7.4	7.8	9.0
21	--	--	7.4
22	--	--	6.1
23	--	--	6.0
25	6.3	7.3	6.3
26	--	--	5.9
27	--	--	5.6
28	--	--	5.5
29	--	--	6.2
30	5.7	7.2	6.3
31	--	--	10.0
32	--	--	5.3
34	--	--	7.0
35	8.5	8.1	--
36	--	--	10.1
37	--	--	15.1
38	--	--	13.2
39	--	--	11.5
40	11.4	10.9	10.3
41	--	--	13.7
42	--	--	14.9
43	--	--	18.4
44	--	--	17.1
45	15.6	15.3	--
46	--	--	16.5
47	--	--	16.3
50	20.0	17.4	19.7

*Time = 0 represents trigger on.

TABLE 9-2

Data for Figure 9-7

TEST NUMBER	PERCENT HITS SECAPEM	ATAGAS
1	46	26
2	26	36
3	52	26
4	53	64
5	78	84
6	34	34
7	93	100
8	24	44
9	86	76
10	42	17
11	19	27
12	4	0
13	22	5
14	2	0
15	91	92
16	78	45
17	24	32
18	67	52
19	34	32
20	81	89
21	53	96
22	71	98
23	61	23
24	2	0
25	6	0
26	16	67
27	9	23
28	61	71
29	11	33
30	73	95
31	19	31
32	52	58

LEAD ANGLE GEOMETRY

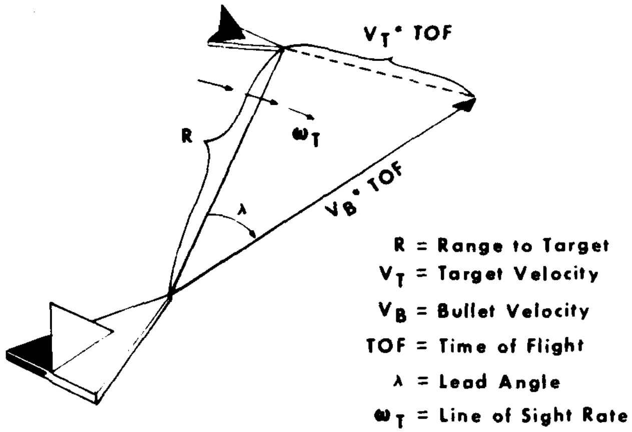


Figure 9-1

RACETRACK TARGET PATTERN

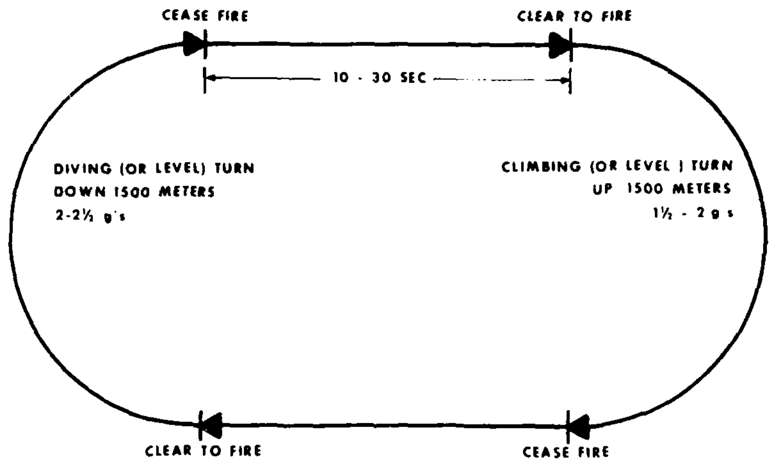


Figure 9-2

FIGURE EIGHT TARGET PATTERN

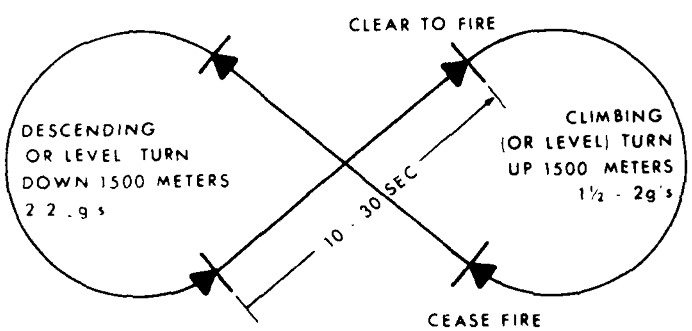


Figure 9-3

BUTTERFLY TARGET PATTERN

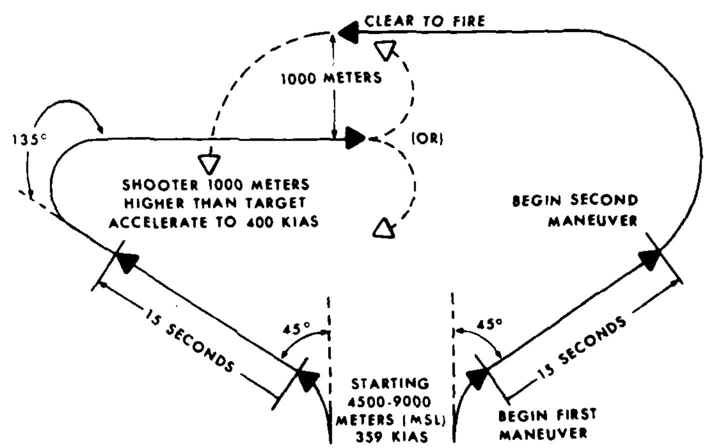


Figure 9-4

COMBAT DART TARGET PATTERN

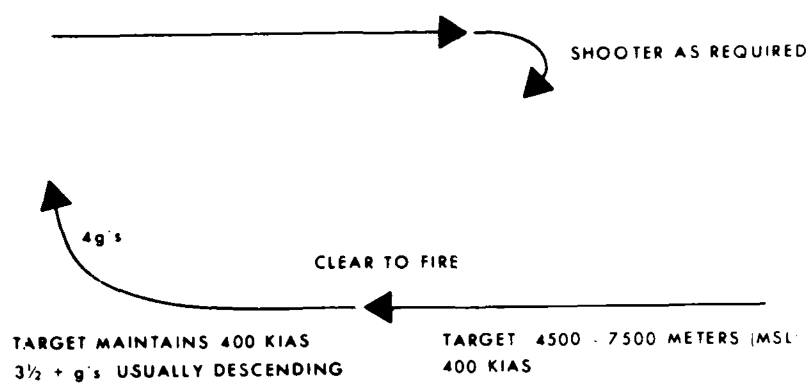


Figure 9-5

ERROR GEOMETRY

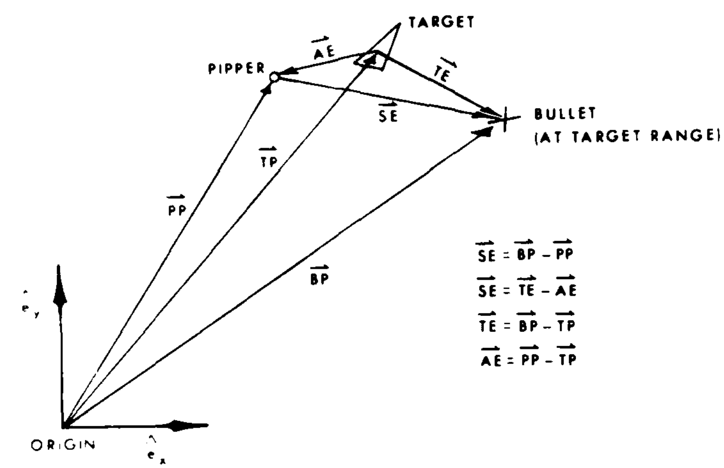


Figure 9-6

DATA CORRELATION

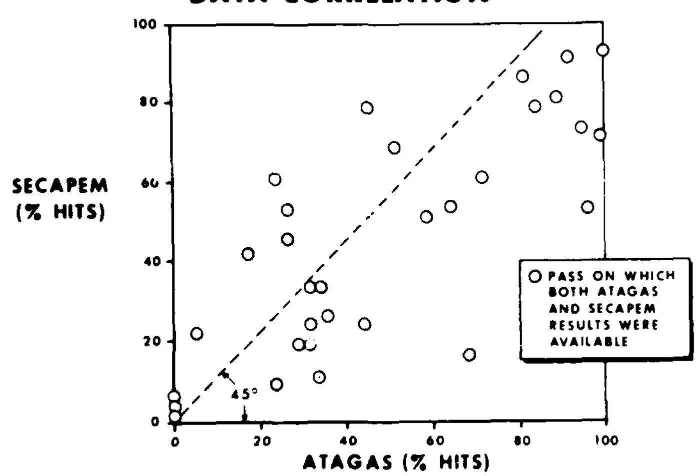


Figure 9-7

DUSK TRACER

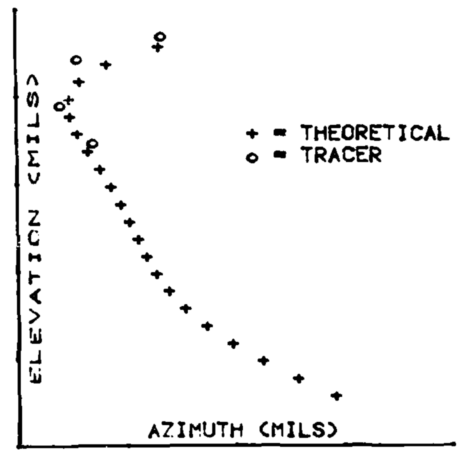


Figure 9-8

FIRING PASS RESULTS

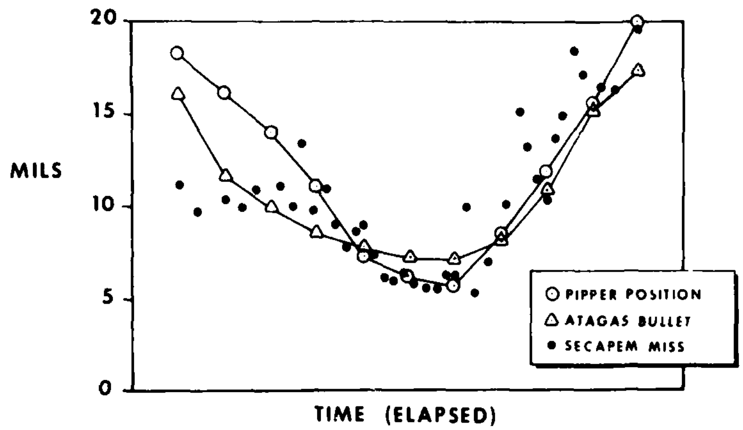


Figure 9-9

20MM FRENCH FLIGHT DATA

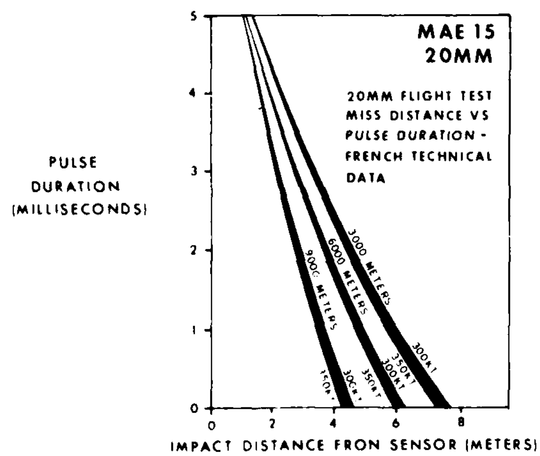


Figure 9-A1

GUN HARMONISATION USING THE SECTOR ACOUSTIC MISS DISTANCE INDICATOR

T. W. Chubb

Instrumentation and Trials Department,
Royal Aircraft Establishment,
Farnborough, Hampshire, England

SUMMARY

The Sector Acoustic Miss Distance Indicator fitted to the Rushton towed target was developed to meet the requirements of air-to-air gun harmonisation trials. This paper discribes the operating principles and indicates the performance limitations which determine the accuracy in both range measurement and quadrant determination. The application to air-to-air gun harmonisation trials using this target system is described.

1 INTRODUCTION

Any new combination of aircraft, gun and gun-sight must be tested to prove that the shells from the gun do indeed hit a target indicated by the gun-sight. This test must be performed under all the various conditions of aircraft manoeuvre that will arise during practical attacks. A system is required therefore where the indication of the gun-sight may be compared with the trajectory of the shells and corrective action taken to harmonise the gun-sight to the gun.

The Sector Acoustic Miss Distance Indicator, fitted to the Rushton towed target was developed to meet this requirement and this paper describes that system and its application to gun-harmonisation.

2 ACOUSTIC MISS DISTANCE SYSTEM

Miss distance or bullet scoring systems based on the measurement of the acoustic shock wave from a passing supersonic projectile have been used for gunnery practice for a number of years. In the UK the Rushton towed target has had an acoustic scorer for many years and is used for ground-to-air and ship-to-air gunnery practice. A ground based version has recently been introduced for air-to-ground gunnery practice. Many other countries have similar systems.

2.1 Principals of operation

Any supersonic object produces a shock wave. If the projectile is reasonably small and suitably shaped then within a short distance of the trajectory the shock wave forms the characteristic 'N-wave'. The formation and characteristics of this wave are given in Refs 1 and 2. The initial pressure rise of the shock wave is related to the distance from the trajectory (miss-distance) by:

$$\Delta P = \frac{CP(M^2 - 1)^{\frac{1}{8}}}{Y^{\frac{3}{4}}} \qquad (1)$$

where ΔP is the pressure rise across the front of the shock wave;
C is a constant and depends on shell dimensions and shape[1,2];
P is the ambient atmospheric pressure;
M is the Mach number of the shell, which is the ratio of shell velocity to the local velocity of sound, and
Y is the perpendicular distance from the trajectory, the miss distance.

The shock wave forms an expanding cone with the trajectory along the cone axis and the apex of the cone just ahead of the projectile. The semi-angle α of the cone is the Mach angle given by:

$$\alpha = \sin^{-1}\left(\frac{1}{M}\right) . \qquad (2)$$

2.2 Accuracy

The accuracy of the theoretical formula of Eq.(1) has been investigated for a number of applications including the original Ref 1. Our own investigations were performed many years ago, and showed that with 30mm calibre ammunition the accuracy was about 4% (standard deviation) for miss distances between about 3 and 8 m rising to about 7½% for miss distances of 2 and 15 m. The deviation from the ¾ power law of Eq.(1) is negligible over most of the useful range of miss distances. At extremely short ranges this law does not apply and individual differences in shell trajectories are responsible for the deteriorating accuracy. At long ranges the deterioration in accuracy is due to the small signal level on a constant noise level. The principal source of noise is the air flow over the microphone and is dependent on target velocity and microphone design. (The accuracy of the system is not so good, by a factor of about 2, when used with large calibre guns, eg 115mm Naval gun.)

These accuracy trials demonstrate that, with care, the acoustic miss distance system may be used for accurate gunnery scoring. In practice, the system is used to count rounds into scoring zones, usually three, and these zones are adjusted for the calibre of gun, mean range, target altitude and velocity. The mean range is used to determine the average Mach number for the shells and no correction is made for change of Mach number with actual range. This factor is usually the largest source of error in systems in use today.

2.3 Dead-time errors

If one round passes the target at a large miss distance and is followed by another round at a small miss distance it is possible that the second shock wave will arrive at the microphone while the system is still processing the first event. In this case the second shock wave would be missed. Detailed calculations[3] have shown that with the rate of fire of the 30mm Aden gun, and under certain attack conditions, this may occasionally occur. The system used during development would for this reason be unsuitable for guns with a high rate of fire, such as the 20mm Vulcan, fitted to F4s etc. The production version of this equipment will use a different miss distance coding technique which will reduce the total instrumental dead time to little more than the duration of the shock wave itself. Calculations[4] then show that even with the 6000 rpm of the Vulcan gun the maximum probability of one of two consecutive rounds going undetected is about 0.03.

It will be apparent from the preceding discussion that measurements using two or more guns simultaneously would be considerably degraded so that trials would normally be restricted to the firing of a single gun.

3 SECTOR INFORMATION

An omnidirectional microphone will be uniformly sensitive to shells in all directions and will not provide the gunner with any indication of the direction of the miss relative to the target. The direction of arrival of the shock wave may be accurately determined by using an array of microphones. There are a number of alternative arrangements of microphone arrays and processing systems, but a factor which must always be considered is cost. Targets are frequently lost and so it is desirable that equipment which is lost with them should be low in cost. It was for this reason that a relatively simple system was chosen which only defines the sector or quadrant of the passing trajectory.

3.1 Principle of operation

Four microphones are mounted on the target in the form of an upright cross. These microphones are shown on the nose of the target in Fig 1. A section through the plane of the four microphones is shown in Fig 2. A supersonic projectile passing the target at right angles to this plane produces a shock wave which is an expanding circle in this plane. Such a shock wave, shown in Fig 2, will reach the microphones in the order 2, 1, 3, 4. This simple 'order of arrival' is detected and used to denote the sector of the miss.

3.2 Accuracy and errors

The microphone array is a square of side 0.11 m and the shock wave travels across the plane at a velocity v given by:

$$v = \frac{v_c}{\cos \alpha}$$

where v_c is the velocity of sound (ie approximately 340 m s^{-1} at sea level). The speed of modern electronics is more than adequate to determine the sector of the miss with extreme accuracy. A calculation, including the measured aerodynamic noise from the microphones indicates an uncertainty of the sector boundary of about 0.3°.

If the trajectory of the shell is not normal to the plane of the microphones then the conic section of the expanding shock wave in that plane is not a circle. If the angle between the trajectory (ϕ) and the plane of the microphone array lies between 90° and the Mach angle α, the expanding shock wave is an ellipse. If $\phi = \alpha$, the shock wave is a parabola and if $\phi < \alpha$ the wave is a hyperbola. In any of these situations there are cases where the order of arrival of the shock wave at the microphones is different to the simple case. In the cases of the hyperbola or the parabola totally different orders of arrival are predicted and the technique fails. In the case of the ellipse detailed calculations have been made to define the magnitude of the errors. Figs 3 and 4 show the results of two such calculations. In both examples the azimuth angle between the relative trajectories of shell and target is 10.7° (shell travels left to right across the target trajectory) and the elevation angle is 10.5°. These give an angle between the shell trajectory and the normal to the microphone plane of 15°. A typical target velocity of M = 0.5 is used. The shaded areas of the diagrams are those regions where a passing shell would be erroneously scored into the adjacent sector. In Fig 3 a fairly typical shell velocity of M = 2.0 produces four error zones of 2-3° wide. In Fig 4 a low shell velocity of M = 1.2 produces two error zones of about 16^{o} width and two of about 7^{o}. Such a low shell velocity would only occur at extreme firing ranges (approximately 1000 m for the 30mm Aden).

The analysis indicates that the system will give acceptable results (error zones <5% of total) for all normal attack conditions provided the trajectory of the shell lies within $\pm 20^{o}$ of the target's trajectory. With increased angles of interception, the above error zones increase to approximately 30% when the intercept angle is 45^{o}.

Flight tests have shown that on rare occasions a microphone will appear to have reduced sensitivity. This effect usually lasts a short time and then corrects itself. It is thought to be due to water collecting on the protective gauze of the microphone and subsequently being blown or evaporating away. Many of the possible errors in sector information caused by this effect are suppressed by some additional electronic logic which implements the rule that the first two events in any sequence must come from adjacent microphones.

4 RUSHTON TOWED TARGET

The Rushton towed target is a cylinder about 2.5 m long by about 0.2 m diameter. It has stabilising fins at the rear incorporating various combinations of flares which are fired by means of a radio command link with the towing aircraft. The target also carries radar enhancing devices. The towing point is just above and slightly forward of the centre of gravity and a typical tow line for air-to-air gunnery is 3000 m long. One of the targets used for the development of this scoring system is shown in Fig 5.

The towing aircraft is equipped with a receiving system for receiving and recording data transmitted by the target. Even if the more elaborate facilities of a ground receiving station are desired the facility for recording data in the towing aircraft is a useful reserve in case of disruption of the target to ground station radio link.

The Canberra is the main target towing aircraft used in the UK. Recently, ex-Royal Navy Sea Vixens have been converted to the towing role and provide a higher performance than the Canberras.

3.3 Target system

Fig 6 shows a schematic diagram of the target-borne part of the system. The lowest of the four microphones (No.3) is used for the miss distance function. The signal from this microphone is fed to the miss distance code generator where a pulse whose duration is directly proportional to the peak amplitude of the incoming signal is generated. There is a maximum duration allowed for this pulse. All four microphones are connected to the sector code generator where the order of arrival of the shock wave at the microphones is recognised and an appropriate digitally coded signal is generated. The sector coded signal is delayed by the maximum permitted duration of the miss distance signal and then the two signals are added together. A modulator and a small radio transmitter complete the target equipment and the coded signal is transmitted to the receiver.

3.4 Receiver system

The receiving system may be located either in the towing aircraft or in a ground station. A schematic diagram is shown in Fig 7. The signal is received and demodulated to reconstruct the combined coded signal. At this point it may be recorded for later detailed analysis. The miss distance and sector signals are separated and decoded. In the miss distance decoder a pulse whose amplitude is directly proportional to the original microphone pulse is produced and presented to three adjustable comparators. The settings on the comparators are pre-determined for required zone radius, target altitude and speed, gun calibre and mean shell Mach number. The mean shell Mach number is determined from the ballistic range table for the mean firing range and the attacking aircraft's velocity. Each round passing the target is simply counted into the appropriate zone and sector. The production version of this equipment will have a combined two zone and four sector display as shown in Fig 8. For gunnery practise this is adequate and the later analysis of tape recordings is unnecessary. For gun harmonisation this is not adequate and the tape recordings are analysed.

A simple, if laborious, technique of analysing the tape recordings is to replay the tape, at reduced speed through a chart recorder, measure the length of each miss distance pulse and identify the sector from the digital code by inspection. This may, of course, be automated giving a print-out of the miss distance and sector of each round.

5 TECHNIQUES USED FOR GUN HARMONISATION

The individual signals generated from every round that passes the target are recorded and individually decoded to give miss distance and sector. The Mach number for the correct application of Eq.(1) depends on the attacking aircraft's speed, target speed, and firing range. The aircraft's and target's speeds are recorded from normal aircraft instruments and the firing range is determined by one of several techniques. For example, if the attacking aircraft is fitted with radar the range may be determined from that. A second technique is derived from the transmission from the attacking aircraft of a tone, generated when the pilot depresses the firing button. This tone is received at the ground station and recorded on the same tape recorder as the signals from the target. From this record the time between the gun firing and the shells arrival may be accurately determined.

The estimation of firing range and Mach number is then made from interpolation from the gun's range table with a correction for attacking aircraft speed and target speed.

The indication of the gun-sight during an attack is recorded on photographic film and so the performance of the gun and gun-sight may be compared.

Visual acquisition of such a small target is difficult. The procedure adopted requires the Range controller, using surveillance radar, to position the attacking aircraft relative to the target. From the position and range at which most attacks commence the pilot should be able to acquire the target visually. The pilot then calls for a target flare to be fired either to aid him in acquiring the target or as a visual enhancement during his attacking manoeuvre. Two types of flare are available; these are smoke flares and light emitting flares. The former are excellent for visual ranging but the smoke tends to be invisible close to the target. Consequently these flares are poor in providing an aiming point. The latter produce an excellent aiming point but are poor for visual ranging. The flares are fired sequentially and so it is possible to have two flares, one smoke and one light emitting burning together. This combination provides the best target but suffers from the limited number of flares (at present, six) which may be carried on any one target. Due to the reasonable cost of the system it is both highly satisfying to all concerned and a definitive proof of gun harmonisation if the later attacks in a trial series actually destroy the target.

6 CONCLUSIONS

The principles of operation and the performance of the sector acoustic miss distance measuring system fitted to the Rushton towed target have been described. The system has been shown to have adequate capability to meet the requirements of a target for air-to-air gun harmonisation, and the techniques used in harmonisation trials have been outlined.

REFERENCES

1. J.W.M. Du Mond, E.R. Cohen, W.K.H. Panofsky and E. Deeds. A determination of the wave forms and laws of propagation and dissipation of ballistic shock waves. *J. Acoust. Soc. Am.*, 18, 97-118, July 1946

2. G.B. Whitham. The flow pattern of a supersonic projectile. *Comms. of Pure and App. Maths.*, V, 301-348 (1952)

3. J.B. Abbiss. On the probability of projectiles going undetected by the acoustic miss distance indicator in stern attacks. Unpublished MOD(PE) material.

4. T.W. Chubb and J. Sheppard. An assessment of an acoustic system for scoring air-to-ground gunnery practice. Unpublished MOD(PE) material.

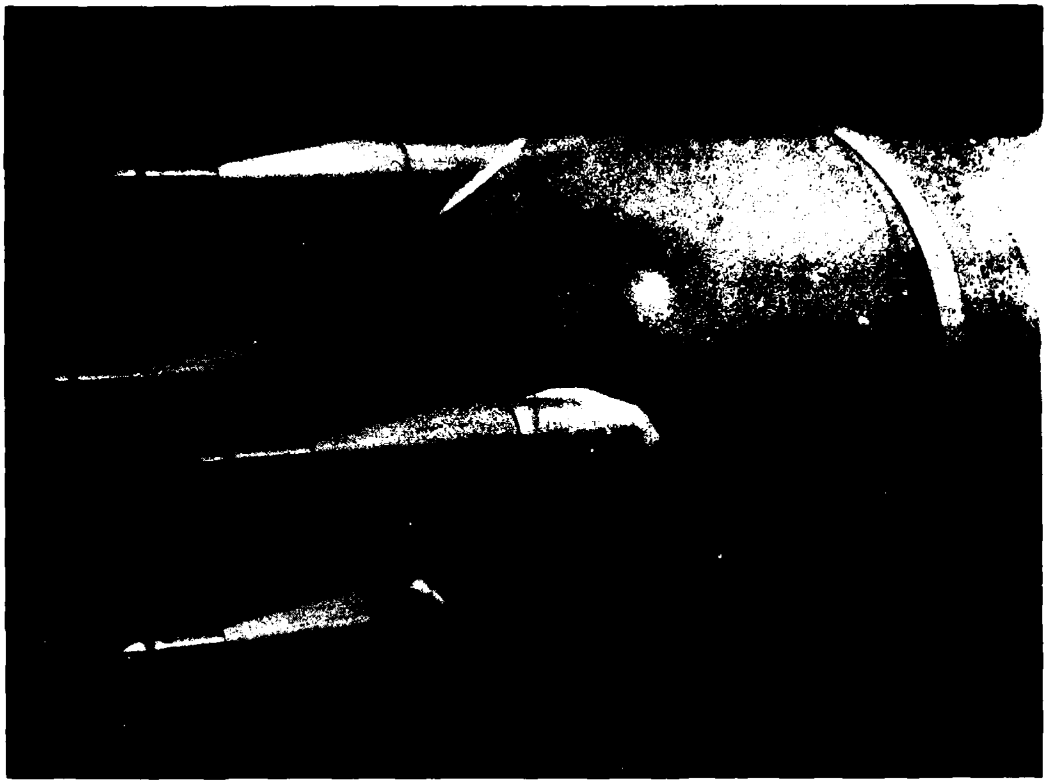


Fig 1 Microphone array on nose of target

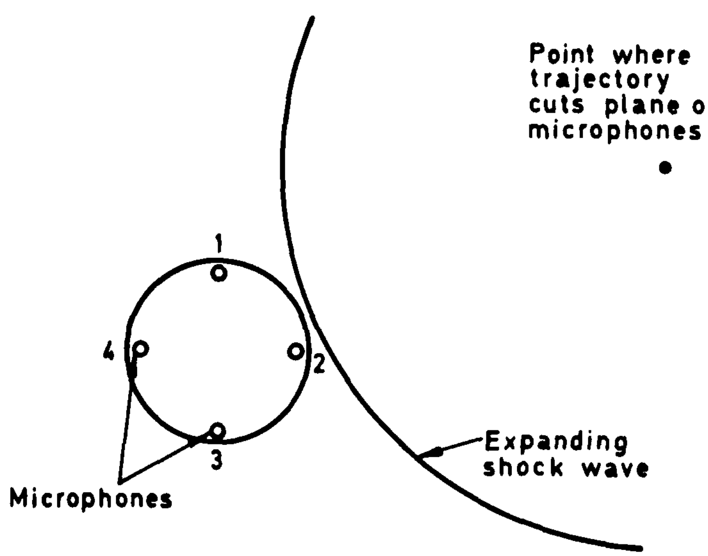


Fig 2 Diagram of expanding shock wave in plane of microphones

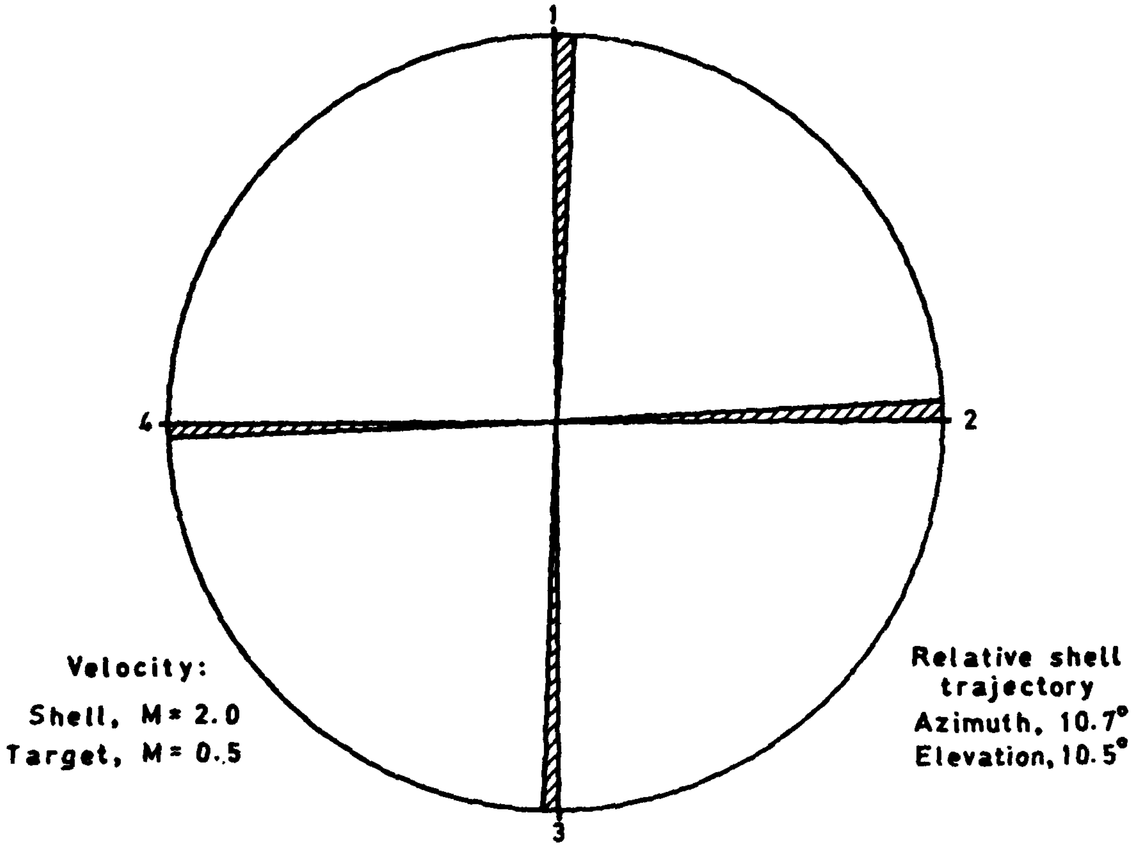


Fig 3 Sector error zones for angled attacks (1)

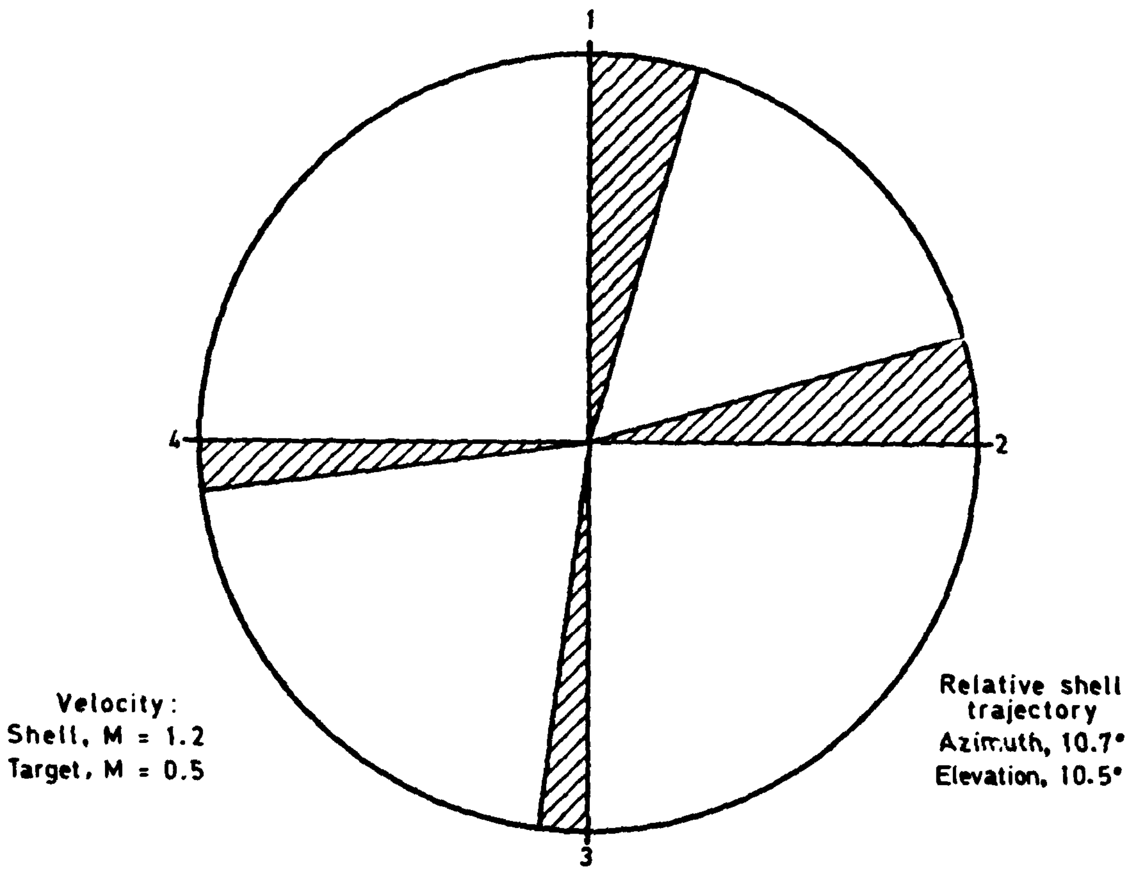


Fig 4 Sector error zones for angled attacks (2)

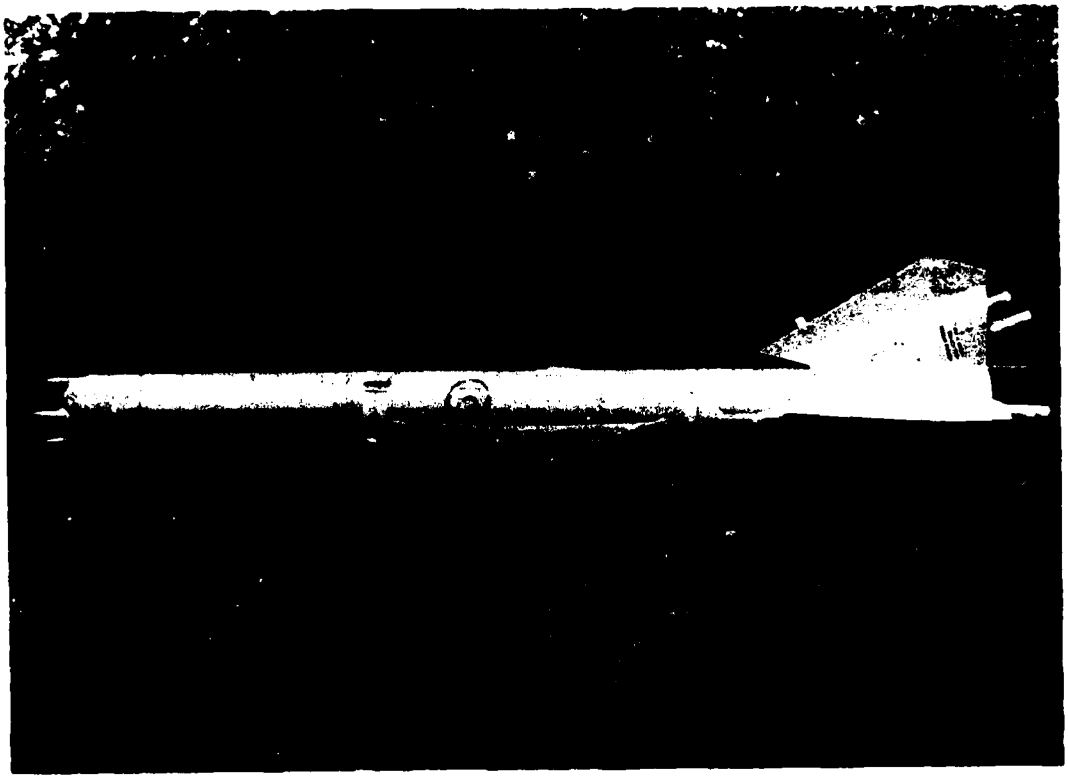


Fig 5   Rushton Mk 2 target with SAMDI

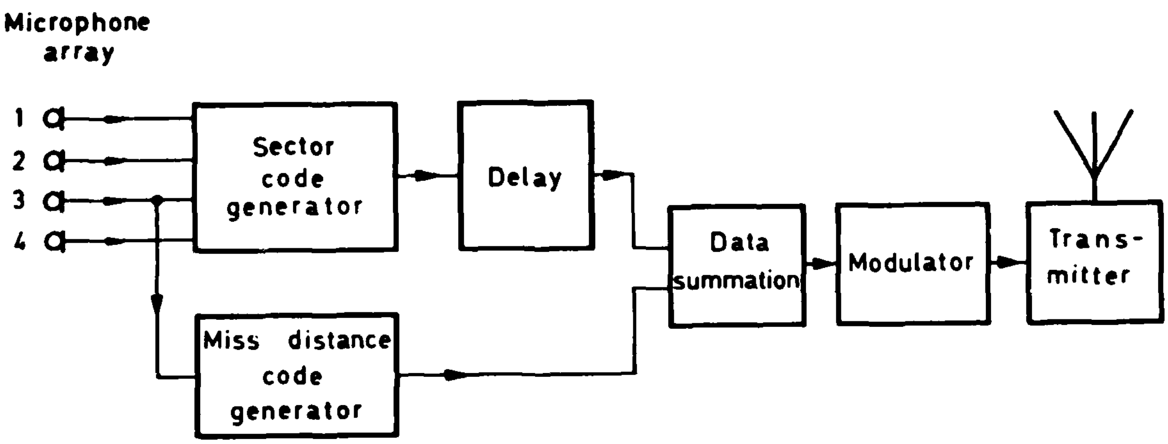


Fig 6   Diagram of target system

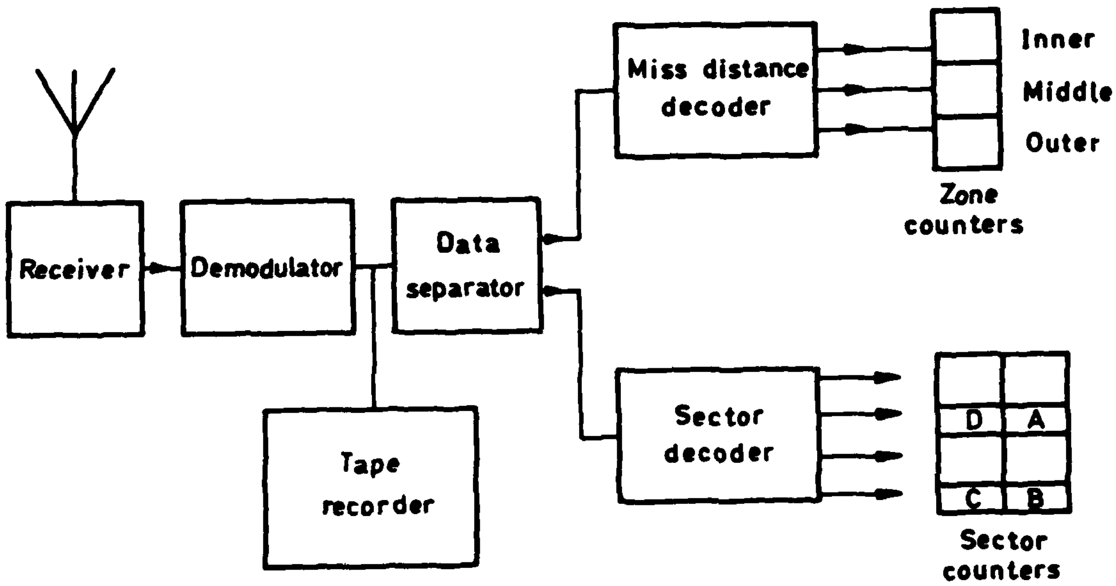


Fig 7 Diagram of receiver system

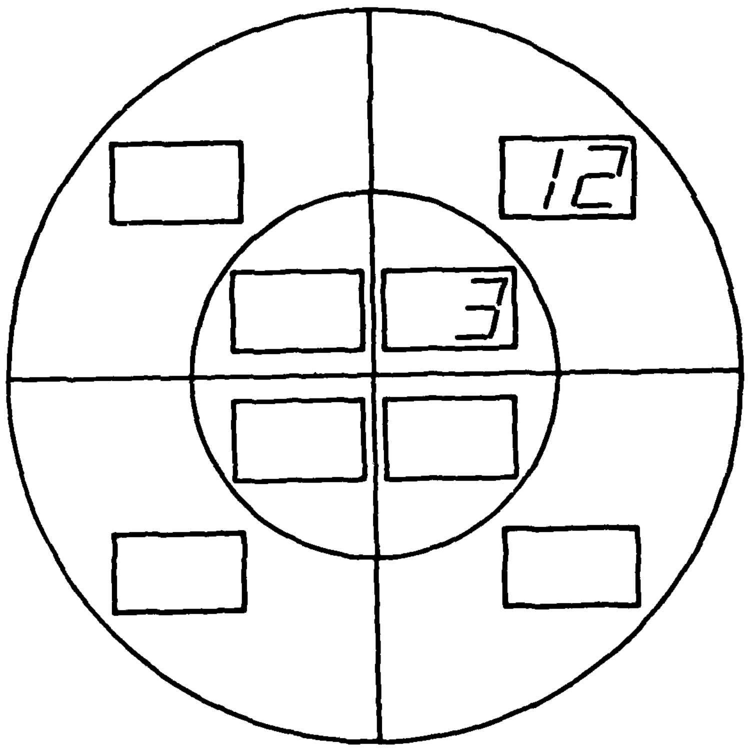


Fig 8 Diagram of display of production version

COMPATIBILITE DU MOTEUR AVEC LES AUTRES SYSTEMES DE L'AVION

par

Jean CONCHE
Chef du Service Essais
SNECMA - ESSAIS EN VOL
B.P. N° 8
13800 - ISTRES
FRANCE

RESUME

La complexité associée aux performances de plus en plus poussées des avions rend nécessaire, à tous les stades, une coopération très étroite entre l'avionneur et le motoriste. Dès la conception d'un avion de combat moderne, les bureaux d'étude respectifs ont à prendre en compte l'étude approfondie de l'adaptation et de la compatibilité de l'ensemble propulsif et de la cellule. Cette coopération étroite se maintient tout au long de la phase de développement.
Une fois arrivé le stade de la réalisation, les essais en vol du matériel font apparaître de nouvelles nécessités d'évolution de la configuration. Parmi celles-ci, beaucoup concernent la compatibilité du moteur avec les autres systèmes de l'avion. L'anticipation et la résolution de ces problèmes doivent absolument être réalisées de façon satisfaisante.
La qualité du produit final en dépend étroitement.

PREFACE

Une adaptation bien étudiée et une compatibilité satisfaisante du réacteur et de la cellule d'un avion de combat moderne constituent des qualités essentielles pour la réussite technique de la réalisation.
Cette observation est valable aussi bien dans le cas du montage d'un réacteur de type nouveau que dans le cas d'une remotorisation.
Dès le stade de la conception, chez l'avionneur comme chez le motoriste, tous les efforts sont mis en oeuvre pour que l'adaptation du moteur à l'avion soit réussie.

En plus du travail soutenu des bureaux d'études, les différents éléments du réacteur et le réacteur complet subissent toute une série d'essais, au sol (essais partiels, banc d'essais, banc d'altitude), puis en vol réel sur banc volant avant le montage final sur l'avion d'armes.

De son côté, l'avionneur développe la cellule et ses différents systèmes, au prix d'une non moins longue série d'essais. Une coopération étroite entre l'avionneur et le motoriste est maintenue pendant toute cette phase préparatoire afin de traiter les différents aspects d'adaptation qui se présentent.

L'expérience a montré que, malgré toutes les précautions prises aux différents stades, une mise au point complémentaire du matériel s'avère toujours nécessaire, une fois le moteur installé sur l'avion d'armes.

Ce sont ces problèmes d'adaptation et de compatibilité que nous nous proposons d'étudier en examinant les différents secteurs dans lesquels ils sont habituellement rencontrés. Il ne saurait s'agir d'une étude exhaustive, compte tenu que chaque cas d'avionnage présente ses caractéristiques spécifiques. Nous illustrerons les cas exposés par des exemples vécus au cours des essais en vol de l'avion ALPHAJET qui est équipé de moteurs LARZAC développés par SNECMA et TURBOMECA, réunis dans le groupement GRTS (planches 1 à 3).

Nous allons considérer successivement les différents aspects suivants :

- Compatibilité aérodynamique
- Pilotabilité du moteur
- Prélèvements { d'air / de puissance
- Démarrage du moteur en conditions extrêmes
- Alimentations { en carburant / électrique
- Conduite et surveillance des moteurs
- Frontières diverses
- Armement
- Aptitude à la maintenance.

1. COMPATIBILITE AERODYNAMIQUE

Le fonctionnement d'un moteur dépend étroitement des conditions d'alimentation en air. Une bonne alimentation en air nécessite :

- une bonne intégration du moteur dans la cellule,
- une entrée d'air bien étudiée.

1.1. Implantation du moteur dans la cellule

L'installation propulsive d'un avion est toujours soumise à l'influence du champ aérodynamique intéressant la cellule, qu'à son tour elle peut affecter. Il importe donc, dès le stade de la conception, d'intégrer convenablement les entrées d'air et les tuyères d'éjection à la cellule, de façon à maîtriser les interférences aérodynamiques.

. Les formes d'un avion dans la partie amont des entrées d'air sont dessinées de façon à assurer une bonne continuité d'écoulement, une couche limite réduite et l'absence de sillage affectant les entrées d'air. Des modifications éventuelles de forme (par exemple, forme du nez d'un avion avec moteurs dans le fuselage) ou des rajouts éventuels de sondes en saillie peuvent avoir des conséquences néfastes sur le fonctionnement du moteur.

Les deux versions, école et appui tactique de l'avion ALPHAJET ont des formes de nez différentes (planche 1). Grâce à une étude soignée des formes dans les deux cas, le fonctionnement des moteurs n'est pas affecté par la configuration du nez.

. L'étude des formes arrières les mieux adaptées est menée conjointement par les bureaux d'étude avionneur et motoriste et vérifiée en soufflerie sur maquette.
Les objectifs suivants sont poursuivis :

- axe de poussée conforme à l'objectif,
- traînée de culot réduite,
- non interférence accidentelle des jets des réacteurs.

Les essais en vol permettent de vérifier ces différents aspects. Il est assez rare que des remises en cause profondes de la conception résultent des essais. En revanche, il faut parfois retoucher les zones environnant les tuyères afin d'améliorer la performance globale (optimisation de la traînée de culot).
L'avion ALPHAJET, dont l'étude aérodynamique a été poussée, n'a présenté aucun problème de développement de cette nature (planche 1).

1.2. Compatibilité entrée d'air moteur

1.2.1. Une bonne qualité de l'écoulement de l'air à l'entrée du moteur est nécessaire à son bon fonctionnement. Tous les réacteurs sont, en effet, sensibles à la distorsion d'entrée d'air qui se traduit par :

- un abaissement des limites de pompage,
- une augmentation de l'excitation vibratoire des aubages de compresseur,
- une perte de performance dans les cas extrêmes.

Un maximum de précautions est donc pris pour que les entrées d'air et les moteurs soient correctement adaptés.

Les moteurs sont conçus avec des marges vibratoires et des marges au décrochage, en principe, suffisantes. L'expérimentation au banc partiel, puis au banc d'essais comporte systématiquement des essais avec des grilles de distorsion. Parfois, également, l'expérimentation sur banc volant est réalisée avec un ensemble complet moteur/entrée d'air représentatif de la configuration finale.

Les entrées d'air sont conçues de façon à assurer l'absorption correcte du débit d'air du moteur en toute circonstance et une distorsion aussi réduite que possible (pièges à couche limite, entrées d'air auxiliaires). Des essais en soufflerie permettent de vérifier les prévisions et d'établir les cartes de distorsion attendues.
Il faut souligner que des considérations de traînée globale de poids, de simplicité et de coût conduisent le plus souvent à ne pas réaliser les conditions optimales pour le moteur, mais à trouver un compromis entre ces exigences contradictoires.

1.2.2. Problèmes de compatibilité entrée d'air/moteur habituellement rencontrés

Certains cas de fonctionnement sont connus pour être critiques pour les réacteurs :

- Fonctionnement au point fixe - Tourbillons liés à la présence du sol - Influence du vent travers ou arrière,

- Fonctionnement à haute altitude et faible vitesse où les marges du moteur sont les plus faibles,

- Fonctionnement sous incidence élevée associée, éventuellement, à du dérapage. Sur un avion de hautes performances, les variations rapides d'incidence sont également à prendre en considération,

- Fonctionnement en supersonique élevé pour les avions équipés d'une entrée d'air sophistiquée.

On met en évidence, par ailleurs au cours des essais en vol, les manoeuvres du moteur qui s'avèrent les plus sévères dans les différents cas de fonctionnement considérés.
Certains aspects spécifiques de la configuration d'un moteur, liés à des caractéristiques de cycle, de conception et surtout de régulation peuvent conduire à des comportements particuliers. Cependant, certaines manoeuvres sont critiques pour la presque totalité des réacteurs, telles que :

- mouvements de manette alternés (agaceries),
- Affichage direct du plein gaz avec post combustion à partir du plein gaz sec ou d'un régime intermédiaire,
- L'état thermique du moteur joue souvent un rôle important. Un moteur "chaud" décroche, en général, plus facilement.

1.2.3. Méthodes d'expérimentation en vol utilisées en FRANCE

Généralement, les essais moteur sur l'avion d'armes comportent une phase initiale de mise au point. L'exploration systématique de tous les cas de fonctionnement du moteur dans le domaine de l'avion fait apparaître la nécessité d'un certain nombre de modifications.
Une fois atteinte une définition donnée du moteur et de l'entrée d'air, considérée comme correcte, commence la phase de validation proprement dite.
En FRANCE, cette épreuve appelée "Bon de Vol" consiste à expérimenter des moteurs rendus intentionnellement plus susceptibles au "décrochage" ou au "blocage" * par des modifications de géométrie (distributeur de turbine, section de tuyère) et de réglage (butée d'accélération). Ces moteurs extrêmes sont destinés à couvrir les dispersions susceptibles d'être rencontrées dans une production de série.
Cette méthode très sévère pour le matériel permet une grande sûreté dans les prévisions de comportement des moteurs de série.

Le bon de vol comporte une phase préparatoire sous responsabilité du motoriste, puis une phase "officielle" réalisée par le Centre d'Essais en Vol.
Le domaine de vol et les consignes d'utilisation du matériel résultent de ces essais. Les critères recherchés concernent le respect des clauses techniques, mais également l'obtention d'un bon fonctionnement du moteur sur le plan opérationnel.

Les marges au décrochage du moteur installé sont, en effet, difficilement prévisibles avec précision, même en fonctionnement stabilisé, compte tenu de la difficulté d'appréhender les phénomènes instationnaires qui se produisent dans une entrée d'air. Le problème se complique encore pour le fonctionnement transitoire que seuls les essais en vol permettent de quantifier.
Pour résoudre les problèmes de décrochage éventuellement rencontrés, on est parfois amené à régler le moteur d'une façon qui le rapproche du blocage, de telle sorte qu'il faut également regarder de près les caractéristiques du moteur vis à vis du blocage.

1.2.4. Dans le cas de l'avionnage du moteur LARZAC sur l'avion ALPHAJET, les difficultés de mise au point suivantes sont apparues :

- Persistance de décollement tournant à bas régime sur certains moteurs, avec passage en décrochage partiel sur mise de gaz. Le phénomène se manifestait au sol où il était aggravé par le vent travers ou arrière et à basse altitude.
  - Le problème a été résolu en montant des vannes de décharge du compresseur HP au 2e étage au lieu du 4e étage comme dans la configuration initiale.
- Sensibilité du moteur aux agaceries à régime intermédiaire à basse altitude, surtout sous incidence, se traduisant par un décrochage momentané ou par un décrochage partiel du moteur.
  - Le décalage du régime N2 de fermeture des vannes de décharge de 70 à 80 % a éliminé ce risque.
- Décrochage du moteur sur panne du calculateur à nombre de mach élevé dû à l'étranglement du flux secondaire associé à l'ouverture automatique des vannes de décharge dans ces conditions.
  - Une consigne d'utilisation attire l'attention sur ce risque qui reste assez improbable et confiné à une partie du domaine de vol peu utilisée.

Le domaine d'utilisation du moteur sur l'avion de série est donné sur la planche 4.

* NOTA : Le terme "blocage" peut s'appliquer à un blocage effectif du régime sur mise de gaz ou bien à une lenteur excessive d'accélération du moteur.

2. PILOTABILITE DU MOTEUR

La marge au blocage d'un moteur ne devient, en général, critique qu'à haute altitude. Toutefois, l'aspect plus général de la pilotabilité d'un moteur sur un avion donné doit faire l'objet d'une étude particulière.
Au cours des essais au banc du moteur, les caractéristiques suivantes sont déterminées et font, éventuellement, l'objet d'une adaptation.

- Caractéristique statique
  Loi poussée/angle manette des gaz
  L'objectif consiste à obtenir une évolution continue, en général linéaire à fort régime, sans plage morte ni discontinuité importante (liée, par exemple, à un régime d'ouverture de vannes de décharge).

- Caractéristiques dynamiques
  - Temps d'accélération depuis le ralenti et depuis des régimes intermédiaires.
  - Absence de déphasage important sur sollicitations alternées,
  - Gradients de poussée - Même en l'absence de spécifications techniques formelles, on considère, en FRANCE, qu'il est nécessaire d'évaluer avec précision la réponse du moteur et de la rendre conforme aux souhaits des utilisateurs. En ce sens, on fait intervenir la notion de gradient de poussée à partir de différents régimes, ce qui permet de caractériser la réponse initiale du moteur. On définit ainsi, par exemple, G1 qui représente l'augmentation de poussée en 1 seconde (planche 5).

Une fois le moteur avionné sur l'avion d'armes, ces caractéristiques sont de nouveau évaluées, au point fixe et en vol, ainsi que des aspects complémentaires, tels que la symétrie des manettes et les problèmes de synchronisation, dans le cas d'un multimoteur. Enfin et surtout, il est nécessaire d'effectuer l'évaluation qualitative globale de la pilotabilité dans des cas de vol particulièrement importants :

- phase d'approche,
- percée IFR,
- vol en patrouille,
- combat,
- ravitaillement en vol (s'il y a lieu).

Cette évaluation est, en particulier, effectuée, en FRANCE, sur des moteurs en configuration extrême blocage.

Sur l'avion ALPHAJET, les moteurs n'ont pas eu à être modifiés pour des raisons de pilotabilité.

3. PRELEVEMENTS D'AIR

Les prélèvements d'air sont utilisés principalement pour pressuriser la cabine et les réservoirs, parfois aussi pour pressuriser ou conditionner des équipements, ou pour des besoins de dégivrage.
L'avionneur demande un niveau suffisant de pression qui impose une source au niveau des étages haute pression des compresseurs.
Les taux de prélèvement peuvent être assez élevés.
Les conséquences d'un prélèvement d'air sont :

- perte de performance installée ou diminution de la marge de température turbine (suivant le type de régulation),
- relèvement de la ligne de fonctionnement associé à une augmentation des temps d'accélération du moteur,
- amélioration de la marge au décrochage du fait du désétranglement du compresseur.

Ces différents effets ont été mis en évidence dans le cas du moteur LARZAC. L'illustration de l'influence sur la ligne de fonctionnement est donnée sur la planche 6. La détermination des limites de décrochage a été effectuée en coupant le prélèvement de climatisation cabine du moteur en essais, afin d'éliminer cet effet favorable. L'évaluation du "blocage" a, au contraire, été effectuée avec le prélèvement cabine ouvert.

4 - PRELEVEMENT DE PUISSANCE

L'entraînement des pompes hydrauliques et des générateurs électriques d'un avion est effectué à partir de boîtes relais entraînées par le compresseur HP. Le prélèvement de puissance correspondant est très variable d'un avion à l'autre. Sur les avions d'armes modernes, la puissance prélevée est assez importante, compte tenu des exigences hydrauliques imposées par les commandes de vol et des besoins électriques pour alimenter, en particulier, le système d'armes.
Le fait de prélever une charge mécanique a les conséquences suivantes sur le fonctionnement d'un réacteur :

- perte de performances ou diminution des marges de température turbine suivant le mode de régulation,
  . Cet effet est généralement assez faible.

- pas d'effet sensible sur les limites de décrochage.

- diminution des marges au blocage. Cet effet résulte du relèvement de la ligne de fonctionnement du compresseur HP (planche 7).

  La détermination des limites du moteur LARZAC en matière de pilotabilité et de blocage a été faite en réalisant la charge prélevée maximale possible en utilisation (montage de l'alternateur de dégivrage associé à un banc de charge électrique).

5 - ALIMENTATION EN CARBURANT

Le système de carburant d'un avion et, en particulier, les pompes de gavage, est conçu de façon à assurer une alimentation correcte des moteurs en carburant.

. Il est exceptionnel que des difficultés soient rencontrées au niveau des moteurs pour des cas normaux de fonctionnement du circuit carburant avion.
Des essais, par temps chaud et froid, sont généralement inclus dans tout programme de développement (Kiruna (Finlande) et Marrakech (Maroc) pour l'ALPHAJET). Ces essais peuvent faire apparaître des caractéristiques inacceptables ou gênantes de fonctionnement, nécessitant des modifications. Parfois, il suffit d'adapter les consignes d'utilisation.

. L'étude du fonctionnement des moteurs en cas de panne simulée des pompes de gavage avion implique des essais spécifiques.

Ces essais ont fait apparaître, sur ALPHAJET, des problèmes de fonctionnement des moteurs à haute altitude avec carburant TR4. Une modification des mises à l'air libre des réservoirs et des modifications des tuyauteries de retour carburant moteur sont intervenues pour améliorer le fonctionnement des moteurs dans ces conditions.

6 - DEMARRAGE DU MOTEUR DANS DES CONDITIONS EXTREMES

Des essais de démarrage dans des conditions thermiques extrêmes et en altitude simulée sont généralement effectués au banc d'altitude afin de couvrir les conditions extrêmes de démarrage des moteurs, en adaptant éventuellement les réglages pour améliorer le comportement. Les essais sur avion, par ambiance très chaude et très froide, ainsi que sur des terrains en altitude permettent ensuite de contrôler la validité des consignes de démarrage proposées.

7 - ALIMENTATION ELECTRIQUE

Sur les moteurs modernes dont la régulation fait de plus en plus appel à l'électronique, la qualité de l'alimentation électrique des calculateurs revêt un caractère important. Il faut, en particulier, que :

- Les fourchettes contractuelles de tension et de fréquence d'alimentation soient respectées,
- Les circuits d'alimentation soient correctement protégés contre les parasites,
- Les blindages aient une bonne continuité,

afin d'éviter des mises en panne intempestives par les circuits d'autosurveillance des calculateurs.

Le fonctionnement du moteur LARZAC sur l'avion ALPHAJET a toujours été satisfaisant de ce point de vue, mais il a été nécessaire, sur d'autres matériels, d'installer des alimentations électriques spécifiques pour l'alimentation du calculateur du moteur.

8 - CONDUITE ET SURVEILLANCE DES MOTEURS

L'intégration des moyens de conduite et de contrôle des moteurs dans des tableaux de bord toujours trop exigus résulte de compromis entre des exigences contradictoires. Les qualités suivantes sont requises :

- accessibilité des commandes,
- bon emplacement des instruments et alarmes,
- qualités correctes de visualisation des paramètres de conduite.

Une attention particulière doit être accordée aux deux aspects suivants :

Contrôle de la poussée au décollage

Il importe que le pilote puisse facilement contrôler que la poussée demandée est effectivement disponible. En ce sens, l'avionneur et le motoriste sont amenés à travailler de concert pour définir le(s) paramètre(s) de conduite le(s) mieux adapté(s) pour cette fonction. Parfois, cette surveillance est synthétisée sous la forme d'un voyant unique.

Manette de commande de puissance

Son positionnement doit être correct et sa cinématique doit présenter des caractéristiques agréables de déplacement et d'effort.

9 - FRONTIERES DIVERSES

Le réacteur est intégré dans la cellule et relié aux différents systèmes de l'avion par un certain nombre de plans frontières tels que :

- frontières mécaniques - suspension, accessoires, raccordement de l'entrée d'air, tuyère froide, prélèvements d'air - inverseur (éventuellement) - commande de puissance.
- alimentation en carburant,
- frontières électriques,
- raccordements hydrauliques et électriques des générateurs,
- système de démarrage,
- détection incendie - circuit d'extinction (éventuellement),
- raccordement des purges, drains et mises à l'air libre.

Tous ces aspects, mineurs de prime abord, donnent généralement lieu à une multitude de petits problèmes de compatibilité nécessitant des adaptations au cours de la mise au point. En effet, il est bien rare que les prévisions soient entièrement confirmées en matière :

- de déformations mécaniques,
- d'environnement thermique,
- de niveau de pression,

à l'intérieur des compartiments dans lesquels sont installés les réacteurs.

10 - ARMEMENT

Il s'agit là d'un vaste sujet que nous ne ferons qu'évoquer puisque les essais de compatibilité moteur-système d'armes ne sont pas réalisés par le motoriste. Les gaz chauds émis au moment d'un tir (tir canon ou tir d'un engin) peuvent entraîner le décrochage du moteur si le sillage est absorbé par l'entrée d'air.
En fonction des problèmes rencontrés, on est donc conduit à prévoir des dispositifs de détarage automatique des moteurs pendant la phase critique.
Un moyen simple souvent utilisé consiste à ouvrir temporairement un (ou plusieurs) électrorobinet(s) appauvrisseur(s) monté(s) sur le circuit d'injection de carburant. Des modifications intéressant l'avion sont également possibles (emplacement des engins, optimisation des séquences de tir, mise en place de freins de bouche...). Dans ce domaine, l'empirisme constitue la règle et chaque cas se traite de façon particulière.

11 - APTITUDE A LA MAINTENANCE

Nous ne voudrions pas clore l'analyse des cas habituels de conflits de compatibilité sans parler de l'aspect de l'accessibilité du moteur pour les opérations de contrôle routinières, qui rentrent dans le cadre de "l'aptitude à la maintenance".
Lorsque pour des raisons de simplicité et de légèreté de structure, les moteurs sont intégrés dans le fuselage, une attention particulière doit être accordée à l'accessibilité. En ce sens, il y a lieu de prévoir, et éventuellement d'adapter, des trappes de visite permettant d'accéder commodément aux diverses frontières, aux régulateurs, aux prises test des calculateurs, au réservoir d'huile ou à son remplissage à distance, aux filtres et bouchons magnétiques et à des accès de contrôle endoscopique du moteur.
Tout l'habillage des moteurs et des nacelles doit faire l'objet d'un souci constant en ce sens, pendant le développement du matériel.
L'implantation des moteurs retenue sur l'ALPHAJET (en nacelles accolées au fuselage) permet une très bonne accessibilité et des temps de dépose et de montage des moteurs remarquablement courts.

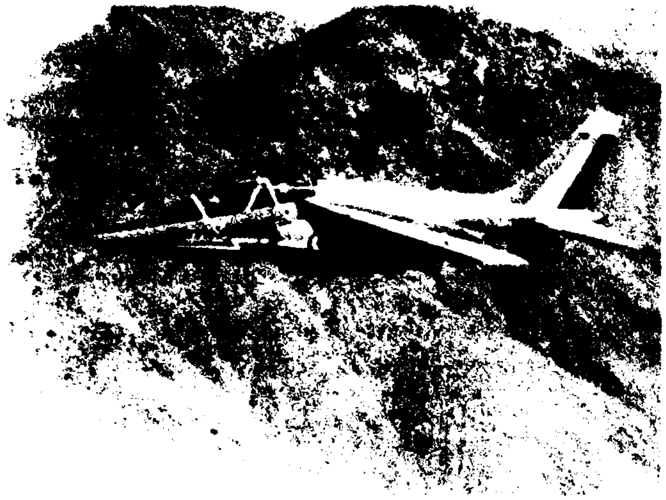
CONCLUSIONS

Au terme de cet exposé, nous voulons essentiellement revenir sur la nécessité de prendre en compte les aspects d'adaptation et de compatibilité du moteur et de la cellule à tous les stades de l'étude et des essais.

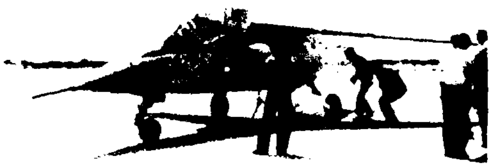
Du succès de ce travail de coopération étroite entre l'avionneur et le motoriste dépend, en grande partie, le succès final de la réalisation.

SOMMAIRE DES PLANCHES

PLANCHE N°1

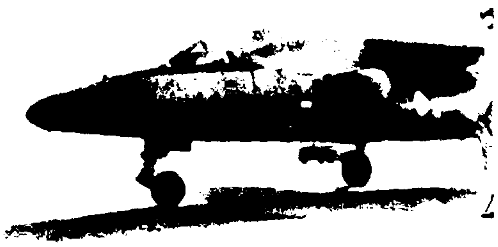


L ' AVION
ALPHA-JET



VERSION APPUI TACTIQUE

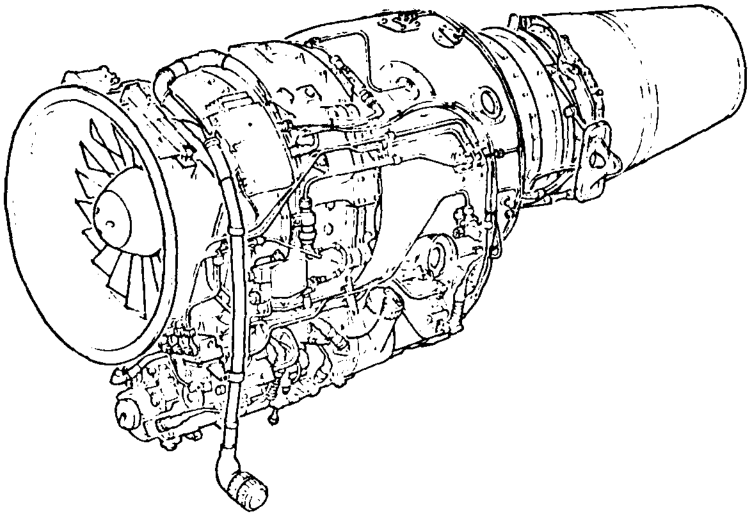
(nez pointu)



VERSION ECOLE

(nez rond)

TURBOREACTEUR DOUBLE CORPS DOUBLE FLUX LARZAC 04



CARACTERISTIQUES PHYSIQUES

Diamètre à la bride d'entrée		451,6 mm
Tuyère d'éjection		
Flux chaud :	- Section de sortie	556 cm2
Flux froid :	- Section de sortie	365 cm2
Longueur totale sans tuyère	- 1320 mm	
Longueur hors tout	- 2165 mm	
Masse GTR avec équipement	- 317 kg	

La masse s'entend sans huile ni carburant dans les circuits

Compresseur BP	axial 2 étages
Compresseur HP	axial 4 étages
Chambre de combustion	annulaire
Turbine HP (aubes refroidies)	axiale 1 étage
Turbine BP (aubes à talon)	axiale 1 étage
Régulation de carburant	hydromécanique et électronique
Démarrage	électrique
Lubrification	autonome sous pression
Allumage	2 bougies HT

Les accessoires du moteur sont installés sur une boîte située à la partie inférieure avant du moteur et sont entraînés par un ensemble mécanique lié au rotor HP.

Conteneur : Longeur - - Hauteur - - Masse -

PLANCHE N° 2

- **DATES CLES DU DEVELOPPEMENT DU MOTEUR LARZAC**

Premier fonctionnement au banc d'essais	20/05/69
Premier fonctionnement au banc d'altitude	22/04/71
1er vol sur banc volant (Constellation puis FALCON 10)	27/04/71
1er vol avion ALPHAJET 01	26/10/73
1er vol d'essai spécifiquement moteur	02/07/75

- **HEURES DE FONCTIONNEMENT AU COURS DES DIFFERENTES PHASES A LA FIN 79**

		Nombre de vols
Banc d'essais sol	11 200 h	
Banc d'altitude	1 400 h	
Constellation	80 h	29
Falcon 10	300 h	197
Essais moteurs sur ALPHAJET	537 h	312

PLANCHE N° 3

DOMAINE D'UTILISATION DU MOTEUR

LARZAC SUR L'AVION DE SERIE

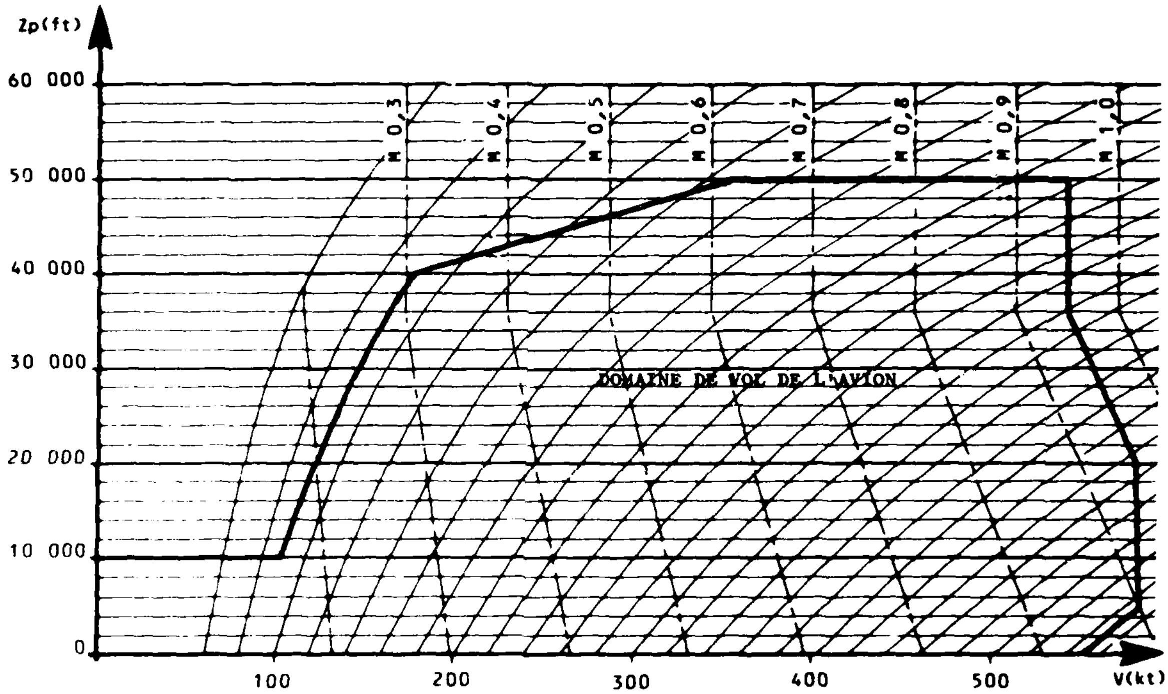
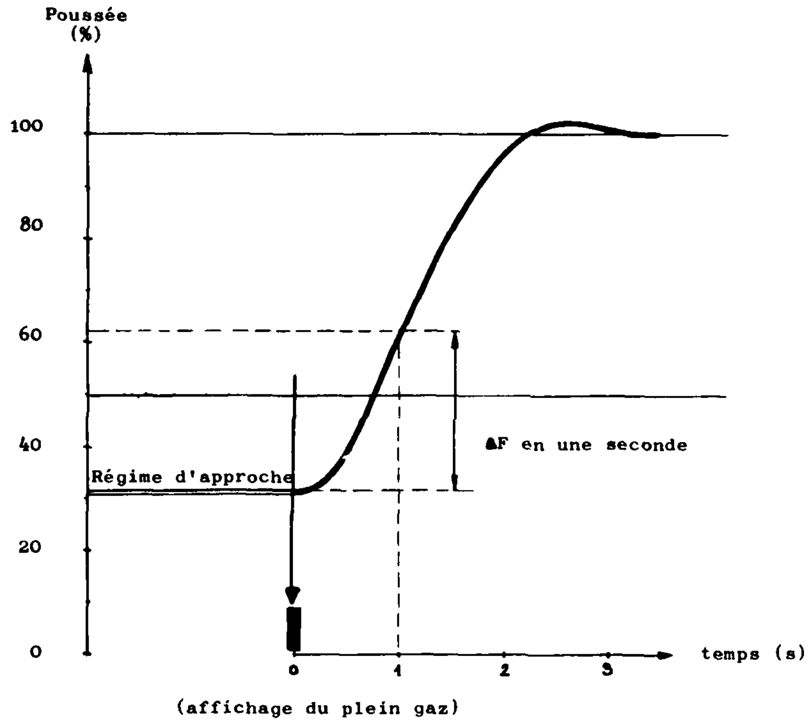


PLANCHE N° 4

GRADIENTS DE POUSSEE



Dans le cas illustré ci-dessus, G1 à partir du régime d'approche = 30 % poussée PG/s

PLANCHE N° 5

INFLUENCE DU PRELEVEMENT D'AIR SUR LA LIGNE DE FONCTIONNEMENT (MOTEUR LARZAC)

Conditions de vol	Régime du moteur	Prélèvement d'air (% débit total)	Relèvement de la ligne de fonctionnement
Sol	Ralenti	0,65	4 %
	Plein Gaz	0,65	2,5 %
25000 ft/110 kt	Ralenti	0,95	2 %
	Plein Gaz	0,95	3,4 % *
36000 ft/120 kt	Ralenti	1,25	2 %
	Plein Gaz	1,25	1,5 % *

(débit de carburant réduit)

* **REMARQUE**

L'influence au plein gaz est plus marquée à 25000 ft qu'à 30000 ft, compte tenu que l'autre moteur, qui est maintenu au ralenti dans les deux cas, est à un régime plus faible à 25000 ft qu'à 36000 ft (72% au lieu de 78 %) et donc, participe moins au prélèvement global demandé par le système de climatisation cabine.

PLANCHE N° 6

EFFET D'UN PRELEVEMENT DE PUISSANCE MECANIQUE
SUR LA LIGNE DE FONCTIONNEMENT DU MOTEUR LARZAC
(ESSAIS AVEC BANC DE CHARGE ELECTRIQUE)

Conditions de vol	Régime du moteur	Charge électrique prélevée	Relèvement de la ligne de fonctionnement
Sol	Ralenti	6 KW	7,3 % *
	Plein Gaz	"	2 %
30000 ft/160 kt	Ralenti	"	5,3 % *
	Plein Gaz	"	2 %

(débit de carburant réduit)

REMARQUE

L'effet constaté au ralenti est moins important à 30000 ft qu'au sol, à cause de l'augmentation du régime N2 de ralenti (75% au lieu de 53 %)

PLANCHE N° 7

FUEL SYSTEM TESTING AND TEST INSTRUMENTATION

by Renato AIMO
New Project System Design
AERITALIA - Combat Aircraft Group
Corso Marche 41, 10146 Torino
Italy

1. FUEL SYSTEM TESTING OF HIGH PERFORMANCE AIRCRAFT

- Fuel system design requirements for high performance aircraft involve critical items such as time and cost, due to the necessity that complete assurance of satisfactory operation is a must long before the prototype is ready to fly.
- Should the high safety or the high performance factors be the requirements, the quality is a fact proven by actual test of the system and all its components under simulated conditions tougher than anything likely to be experienced even during the subsequent flight test program.
- There was time not long ago, when fuel system design involved engines of low specific fuel consumption, moderate rate of climb, and no aerodynamic fuel heating problems.
- A well tried analytic approach could be applied to such a design with considerable confidence. Test stands were then mostly limited to static ground tests, with often only a portion of the system being evaluated. Such stands together with ground and flight testing of the completed prototype were adequate to demonstrate the proposed system capability to fulfil the requirements.
- The past years have changed the picture quite drastically. The use of the turbine type engine has brought greatly increased fuel flows, rates of climb, and pre-heating of the fuel both through aerodynamic heating and by the use of the fuel as a heat sink. In addition, the use of different fuels has made the prediction of system and components functioning more difficult. In an accelerated program, production aircraft are often built concurrent with prototype testing. In this case, the penalty for wrong estimates in basic system design can involve expensive production retrofit of grounding of aircraft if flight safety is involved.
- Since the beginning of the TORNADO Project, AERITALIA has been tasked to study and develop a full scale Fuel System Test Rig.
- The necessity for a large project as full scale fuel system testing, can be summed up as reliability assurance and safety of the aircraft.
- As far as possible, all aspects of fuel system management on board the aeroplane have been proved on the ground and broad tested to give reasonable assurance of satisfactory design, as early as possible before first flight.
- The fuel test rig results have been compared with both theoretical calculations, and flight test results.
- It has been proved that a full scale fuel rig has reduced to a minimum the time required to clear the aircraft first flight.
- All modifications introduced on preproduction as well as on production aircraft have been fully tested and the results were as predicted. Furthermore tests were performed to back up flight test results.
- The work carried out on one of the most sophisticated fuel system has fully demonstrated the necessity and importance to manufacture a full scale fuel system rig and the need for a complete integration of the work of Technical Department, Test House Dept., and Flight Test Dept.
- Particular attention has been given to the test data acquisition system with the purpose to display visually, to record and to log fuel test results such as pressures, flows temperatures, angular movements, currents and voltages under static and transient conditions. The above is in addition to the standard aircraft fuel pressure fuel flow and fuel quantity gauging system.
- On studying and finalyzing this test data acquisition line, efforts have been made to adopt latest technologies with capable and high value test equipment in order to obtain reliable fuel rig results.
- Specific flight tests have been carried out on the Tornado aircraft during hot weather trials to evaluate the heat rejection from the engine into the fuel used as a heat sink.
- Parameters such as flows, pressures and temperatures of lube oil and fuel were measured at the fuel cooled oil cooler inlets and outlets as well as in the fuel recirculation line, and compared with both theoretical calculation and rig testing.

The analysis has demonstrated a good correlation between the results obtained in flight and the results obtained on the fuel test rig.

- Furthermore, the fuel test rig has been used to back-up and solve malfunctions encountered in flight and to explore the real limits of the fuel system outside the flight envelope.

It is AIT opinion that the full scale fuel test rig used, has satisfactorily fulfilled its task.

- In terms of time and experience gained checking out a new system can mean real money saving.
- Initial flight safety is ensured.
- Component failure may be checked by simulation in a manner not possible in the actual aircraft.
- Complete system reliability is verified.
- A basis is established for future design changes.
- System "Margin of Safety" is finalised by establishing maximum limits of performance which would not be feasible in flight.
- Preliminary calculations could be verified.

Whilst the fuel test stand is not the answer to all the problems of fuel system design, it is certainly a good instrument for finding those answers.

- In the next pages the following will be briefly examined:
  - the fuel system under investigation
  - the test house capabilities
  - the slave system
  - the test house instrumentation
  - the flight test department instrumentation
  - brief analysis of tests carried out.

2. FUEL SYSTEM SCHEME

2.1 Fuel System Description

The fuel system as shown in (fig. 1) has been designed to meet the performance and requirements of a Stol aircraft designated TORNADO.

The system consists basically of a number of fuel tanks, (fig. 2) an engine feed system, a transfer system, a refuelling/defuelling and dumping system, a vent and pressurization system, a cooling system, gauging and flow metering system.

Under normal operating conditions no action is needed by the pilot to sequence the fuel usage out of the various internal or external fuel tanks. Once the engines are started and the aircraft is ready for take-off the internal and external fuel of the aircraft is transferred and pumped to the engines automatically.

Normal center of gravity control is maintained by feeding the engines separately from the two main fuselage tank groups. Should there be a need, it is possible to adjust the centre of gravity by manual operation.

2.2 Engines

Each engine is normally fed from its own main tank group by double ended AC motor driven boost pumps. These pumps are installed in collector boxes.

Two parallel switched boost pumps are provided to supply one engine in normal cases. In the event of one boost pump failing the remaining pump is capable of delivering the maximum reheat flow at sea level standard conditions (i.e. Take-off).

Therefore one boostpump or two boost pumps inoperative will not affect the aircraft safety.

A cross-feed valve between the engine feed lines will be opened automatically when the fuel low level signal is activated.

The pilot has the capability to override the low level signal to the cross-feed valve. With the cross-feed valve open, both engines will be supplied from both tanks simultaneously, or, in the case of complete fuel loss of one main tank, both engines will be fed from the remaining tank or, in the case of single engine failure, one engine from both main tanks.

Check valves in the engine feed lines prevent reverse flow from the cross-feed manifold and possible loss of fuel through a severed engine feed line in a battle damage tank.

The engine feed lines are isolated from the engine bay by electrically operated fire wall shut-off valves.

2.3 System cooling

The peculiarity of the system described is that the fuel is used as heat sink to cool the aircraft system and this was one of the major tasks we had to face.

The cooling system is required to cool the aircraft system, i.e. hydraulic oil, the gear box, CSD (Constant Speed Drive) and APU (Auxiliary Power Unit) oil and engine oils during flight and ground operation of the aircraft.

There is one cooling system per engine feed system. Fuel has been used to maintain the oils at acceptable temperatures.

The aircraft system is cooled by an integrated heat exchanger. The heat exchanger is designed such that the hydraulic oil and gearbox oil are completely separated.

Under condition of low engine fuel consumption the fuel flow is insufficient to dissipate the heat input of the system without exceeding the acceptable engine inlet temperature.

Fuel flow through the engine feed systems is increased by opening a recirculation valve which allows fuel to return to the tanks from a tapping downstream of the engine oil cooler. This recirculated fuel is cooled before returning to tanks by a ram air heat exchanger.

For cases with still relative hot fuel temperature of the heat exhanger outlet a fuel jet pump is provided to mix the recirculating fuel with tank fuel before the fuel enters the tank.

The system is shown diagrammatically on fig. 3.

2.4 System Function

The maximum acceptable temperature of the fuel at the engine HP fuel pump outlet is 150°C. This temperature has been taken as the design criteria for the fuel cooling system.

To maintain the fuel temperature at the HP fuel pump outlet a recirculation system is required. This recirculation is taken from the HP pump spill flow and controlled by a temperature controlled fuel recirculation valve. In order to prevent the allowable fuel tank temperature to be exceeded an air cooled fuel cooler and a fuel jet pump are installed in the recirculation line.

The fuel outlet temperature of the tank has been controlled by telemetry during flight testing.

A second recirculation line comes from the reheat control system back into the main feed line upstream of backing pump.

This flow is a reheat servo flow, using the boost pump delivery pressure as back pressure for the control unit and is primary a cooling flow of the vapour core pump sealing and actuators.

2.5 Ground Operation

During stand-by the APU, the starboard gearbox, part loaded generator and unloaded hydraulic pump are running. The APU fuel consumption is too low to cool the aircraft systems by rejecting the heat into the consumed fuel and a recirculation system is therefore employed.

Tank fuel will therefore be heated and the stand-by time limited by the maximum tank temperature.

The recirculation change over valve is in this case in ground operation position. The required cooling fuel flow will flow through the open connection between main and recirculation line. An air ejector driven from APU bleed air is provided to suck cooling air through the air cooled fuel cooler to cool the recirculation fuel and therefore increase stand-by time.

2.6 Tank Temperature Increase During Stand-by and Flight

An example of tank temperature increase during stand-by and flights is shown in fig. 4. For a stand-by case of 4 hours the ambient fuel temperature at the beginning shall not exceed 29° C so that the fuel temperature inside the tanks after alert time will stay below 50°C.

The flight case shown is one of the design case for the air cooled fuel cooler. For take-off and climb it is assumed that there is no fuel recirculation to the tanks, due to the higher fuel consumption of the engines.

Figure 5 show the temperature distribution of the fuel system during stand-by.

3. FUEL TEST RIG FACILITIES DESCRIPTION

3.1 Fuel Test Rig Building

An adequate Test House has been provided to contain the fuel test rig, its instrumentation, control and power supplies and to accomodate its attendant staff.

The test room 30 x 30 mt. with a useful height of 18 mt. is equipped with the movable platform carrying the fuel rig dummy structure. The test room is provided with a 5 tons crane, driven from the ground, capable of servicing all the 900 SQ. MT. of the testing area.

The floor centre part of the test room is provided with rails buried into reinforced concrete, which allow the fixing of the two stout stanchions of the rig.

The test area is provided with adequate network of underground passages to allow collection and conveying by gravity liquids to a suitable external tank (leakage or pooring of fuel, etc.).

With the purpose to avoid fuel vapours remaining inside the test house, a thermoventilation plant is provided, this installation allows 4 complete air changes per hour and maintains a constant small overpressure towards the building external pressure.

A fire extinguishing system is provided in the test room which could be manually operated or automatically ignitiated through vapour concentration sensors placed in differents appropriate positions.

In the main building under-ground rooms (30 x 10 mt) are installed the central heating and control-room-air-conditioning systems the electrical main power transformer; the electric motors and generators driving the first stage pumps; the hydraulic power generation for the rig movements; and the emergency electric power generator capable to ensure essential services in emergency or black-out case.

The control room is placed at a higher level than the test room with the purpose to have a complete view of the test rig and its associated equipments through large window panels.

The control-room is air conditioned and noise protected, it contains the control panel, the failure simulation panel, the data processing and recording devices with the relevant interface components.

Figure 6 gives an indication of the test house dimension.

3.2 Test Rig Structure

It can be seen from fig. 7 that the test rig structure consists of two stout stanchions where, by means of two oscillating roller bearings, rests an horizontal beam having two cranks at its ends.

In the middle of the horizontal beam is positioned the rotating platform. The movement in the horizontal and vertical plane or any combination of both, allows the positioning of the aircraft dummy structure in any required position in pitch and roll, including inverted flight.

The peed rotation in the roll axis permits to reach the inverted flight attitude from level attitude in 30 seconds.

The test rig movements are obtained by means of hydraulic system and reduction gear.

3.3 Aircraft Dummy Structure

Every effort has been made to simulate the shape, the relative position of each accessory, the pipe routing and the aircraft fuel tanks internal bays.

All components and pipes are the same as the ones of actual aircraft. The complete fuel system with both wings external tanks (under wing and under fuselage), vent and dump line routing, engine feed and in flight refuelling lines are provided.

The dummy structure is composed of a welded steel supporting structure reproducing the centre fuselage and the fin; the wings are made of light alloys components machined and assembled in the same way as for prototypes.

The wings are connected to the wing centre-box through a pivot allowing the wing positioning at any required sweep angle.

The connection of the dummy structure to the movable platform is obtained through an adapter. Fig. 8 shows the dummy installed on the adapter.

4. SLAVE SYSTEM DESCRIPTION

4.1 Fuel System

Low and medium pressure fuel subsystem of the complete fuel system of the aircraft up-to engine inlets are practically identical to the aircraft actual ones.

All components, pipes routing and different equipments are the original ones and are installed as they are on the aircraft.

The simulation of the engine fuel consumption is obtained on the fuel test rig through a slave system which will be described here after.

4.2 Engine fuel consumption simulation

The engines fuel consumption modulation is obtained by means of special servo-valves installed along the engines feeding lines.

A servo closed loop in each servo-valve incorporating a flowmeter gives automatically the required connection to maintain the fuel flow at the desired value.

Monitoring and flow control is done in the Control-Room.

4.3 First stage pump driving system

Three machines axially coupled on a unic steel supporting frame (one squirrel cage three phase asincron electrical motor and two 3 phase syncron brushless generators) supply the power for the d.c. motors of two conversion groups.

Each of these groups, made of two axially coupled machines comprises a d.c. motor with separated excitation, complete of counter dynamo, and a three phase syncron generator to supply the First Stage Pump driving motors.

Each of these motors absorbs a maximum power of 15 Kw, with a peak torque moment of 6,5 Nm and are able to accelerate from 6000 RPM to 14000 RPM in 5 seconds.

The variation of the angular speed of the driving motor is obtained by means of a lever placed on a panel in the Control Room close to the Engine RPM indicator.

4.4 Heating & cooling simulation on the dummy

The simulation of the heat load on the engines fuel feeding lines is obtained feeding the actual aircraft fuel cooled oil cooler with hot oil, which will transfer into the fuel the required heat loads.

The simulated engines heat loads in the recirculation and servo-reheat lines is obtained via two special designed heat exchangers.

The temperature of the oil feeding the above heat exchangers is automatically regulated as a function of the fuel temperature at the heat exchanger outlets, in order to guarantee within narrow tolerances the preselected fuel outlet temperatures.

The block diagrams of the above systems are shown in fig. 9.

The air cooled fuel cooler is activated by an ejector which simulates the cooling effect on the recirculation lines both on ground and during flight.

The air flow is controlled from the control room.

4.5 Refuelling and Defuelling

The system has been realized with the purpose to simulate refuelling and defuelling of the aircraft as well as fuel return to the external store tanks.

The refuelling system is capable to refuel the aircraft at any desired flow and pressure up to 380 KPa (55 PSIG) at the airplane refuelling connection. Furthermore one of the external store tank is capable to heat the fuel up to 60°C.

In a fenced protected area outside the test house two cylindrical fuel tanks are placed each with a fuel capacity of 22000 liters. One of the coibented tanks is used to heat the fuel up to 60°C, via an internal coil heated with hot water. The heating system is provided with relevant thermostats hot water shut-off electrovalves, temperature probes-temperature indicators - etc.

The system is capable to heat the fuel by means of a double heat exchanger, between over-heated steam and water in a first stage and between water and fuel at the second stage.

The driving and regulating panels of the heating system complete with all relevant control and safety instrumentation, is located in the suitable concrete building in the external store area.

4.6 Refuelling line

This supply line is provided with a centrifugal pump (driven by an asyncron three phase motor) having a max. outlet pressure of 920 KPa, (133 psi), in order to ensure the fuel delivery to the dummy; on the same line a basket filter and a microfilter separator foreseen for a maximum flow of 2300 lt/min. (500 G.P.M.) are installed in series; furthermore a flow regulator and a pressure regulator are installed which allow at the single point pressure refuelling connection, values between 70 KPa and 380 KPa (15 ÷ 55 P.S.I.).

Return Line

Two interconnected return lines are provided for fuel collection connected to the delivery pipes, by means of gate valves, which allow the fuel circulation without involving the aircraft system. On the return line a suction pump is installed,driven by an asyncron motor usually controlled by suited level sensor duly located in the consumption collection tank inside the Test House. In emergency case it is possible to control such a motor manually.

4.7 Engine Consumption Collection Tank

At the inside of the Test House the engine simulated fuel consumed is collected in a suitable 3000 liters capacity tank positioned on a precision balance, thus allowing fuel weighing operations each time this is requested for test purposes.

In this tank, level sensors are positioned controlling automatically the engine driving the suction pump.

4.8 Measuring Group

Inside the Test House a junction box is provided between the mock-up and the external area, where, near the deaerator filter group, a volumetric flow meter and a totalizer of the fuel flow are positioned. In the same zone a control panel is installed which is comprehensive of the various indicators and push buttons for pumps starting and representing a partial repetition of a similar panel placed outside building.

5. TEST DATA MEASUREMENT CONCEPT

The Test Data Measurement Concept has been developed on the basis of "Test Information Sheets" for ground testing, System Specifications and direct information given by the system engineers.

The types and number of parameters to be measured are dictated mainly by problem areas to be investigated and by the requirements and results of the individual tests.

5.1 Description of the Test Instrumentation

The most important requirements of the test instrumentation are the following:

- all the functions of the system describing signals must be measured and recorded.
- The instrumentation equipment must not influence the system under test.
- The reliability must be as high as possible.
- Influence of the environment (temperature, shock, vibration, noise, etc.) on the signals must be negligible.
- An easy maintenance, calibration and handling of the whole test instrumentation is required.

5.2 Parameters to be measured

The following parameters are to be measured during each test:

- Pressures
- Flows
- Temperatures.

Their positions, the maximum value to be measured or their ranges have been carefully evaluated and positioned according to the fuel system scheme and are the same that have been used during flight testing.

5.3 Transducers Specification

5.3.1 Pressures

For measuring purposes of slowly variable pressures, linear variable differential transducers have been used, providing an electrical linear output, corresponding to the applied fluid pressure.

A suitable transformer is built-in into the housing of the transducer with the purpose to allow transmission of the output signal in D.C. form, which is indipendent of capacitance, inductance and cable lenght.

The transducers operating characteristics are as follows:

Static error band	0,2% F.S.
Excitation supply	8 v. RMS; 8,5 KHz
Analog output	0-5 V D.C.
Ambient temperature	10° - to 40° C
Temperature range of fuel	up - to 200° C
Temperature coefficient	0.003% F.S./° C

For the pressure transient measurement requirements, strain-gauge transducers are used, allowing up-to 1000 Hz frequency response.

5.3.2 Flow

Flowmeters having the following characteristics have been used:

Repeatability	Better than ± 0,05%
Accuracy	Better than ± 0,50% reading
Minimum operating pressure	10 ÷ 30 P.S.I. (69 ÷ 207 KN/m^2)
Operating temperature range	0°C ÷ +200°C

Pressure Drop	4 P.S.I. (27. 6KN/m^2)
Response time	Better than 30 ms.
Max operating pressure	3500 P.S.I.

5.3.3 Temperatures

Temperatures have been measured by means of copper-constantan thermo-couples, having an error ± 0,4% reading.

5.4 Interface concept and data processing

The data acquisition and interface concept, which is used on the rig consists of an analog circuit which, through a programming board, is connected to analog recording equipment and to a digital circuit for data processing.

All the pressure and flow transducers have their individual meter display. The system is designed in such a way that a rapid exchange of channels by means of programming board is possible.

5.4.1 Measurements and recordings

The output signals from the single transducer are read in engineering units (pressure in P.S.I.G., flows in LBS/HR, temperature in °C) on pointer instruments and are recorded both by means of ink or U.V. recorders and by means of a data logger after their conversion into digital form.

The use of the ink-recorder or U.V. is not extended to all parameters or all interested channels in a single test.

However, a programming board has been foreseen, to allow a complete flexibility both on the channel choice and parameters to be recorded and on the choice of the type of analogic recorder (ink or U.V. recorder).

5.4.2 Pressure and flow channels

All the fuel test rig measurements, carried out by transducers, are arranged in such a way to measure accurately high pressure transient, low working pressure and flow.

Each transducer is separately connected to its own supply, calibration and display unit.

5.4.3 Signal conditioners and indicating instruments

They are adequate to satisfy the requirements of compatibility, sensibility, stability, and response of the whole channel of data acquisition.

Indicating instruments are adjusted into engineering units. Their reading scale is circular within an arc of 250°.

5.4.4 Analogic recorder

The multichannel recording for pressure and flow measurements is guaranteed by the following instruments:

- Ink-Recorder type, actual time, for frequency response up-to 50 Hz
- U.V. Recorder, immediate self developing type, for frequency response up-to 1000 Hz.
- The multichannel point recording, which is limited to the temperature measurement , writes directly on paper.

5.4.5 Data Logger

The data logger has been foreseen for digital monitoring and recording of pressure, flow and temperature signals.

The data logger includes:

- Digital indicator to visualize all the test parameters in engineering units.
- Scanner to preselct the measuring points with a split possibility.
- Ink printing on paper.
- The possibility has also been foreseen of printing the data, running time, type of test and all other elements necessary to a better clarification, by means of a key board.

5.4.6 Results presentation

Upon completion of each test a report including the description of the test carried-out, of the rig and of the instruments used is issued for system engineering analysis and evaluation.

The data acquisition system block diagram is shown in fig. 10. The parameters measured number is indicated in fig. 11.

6. FLIGHT TEST INSTRUMENTATION

6.1 Foreword

The evolution of instrumentation system from the 1950's to the present, reflects quite a big amount of radical changes. Instrumentation system designers are utilizing tools and capabilities not imagined only few years ago.

Simultaneously with the evolution of instrumentation systems a computer technology evolved. Analog tape recording replaced oscillographic and graphic recording using the pulse duration modulation (PDM) encoding technique for time division multiplexing of quasi-static data and narrow band frequency modulation (NB-FM) for multiplexing of higher frequency data.

Telemetry supplemented these basic recording system by transmitting data via FM radio links modulated with the NB-FM carrier with these advances, data was recorded on tape which could be edited, converted to digital form, and recorded in a computer.

Within the last years hybric systems and derivatives of frequency multiplex systems have become available; however the primary means of airborne data acquisition continues to be Pulse Code Modulation (P.C.M.) and FM systems for telemetry and tape recording.

Increasing use of airborne mini-computers now permits the reduction of data on board during testing, lessing the amount of data recorded or telemetered, and making possible the selection data for istantaneous display to the test crew.

Data processing ground stations have become almost fully automated and offer great flexibility in adapting rapidly to changing requirements.

The evolution of the philosophy of testing reflects the increasing complexity of aircraft and the stringent requirements of flight testing.

6.2 Airborne Flight Test Instrumentation System

With the purpose to use a centralized data acquisition system reducing to a minimum the cable numbers coming from the transducers, it has been adopted a project which uses standard wiring interface box (S.W.I.) in different aircraft zones.

From each standard interface (S.W.I.) box a cable loom connects the transducers to the signal conditioner switching and amplification boxes (S.C.S.A.B.) of the data acquisition system.

The total number of cables connecting S.W.I. to the data acquisition system is equivalent to the channels total capacity of the system. The S.W.I. are installed in the front fuselage, in the cockpit, in the centre fuselage, in both wings right and left and in the rear fuselage.

All transducers are wired from the installation point to the nearest S.W.I. - being the S.W.I. provided with 5 poles connectors, the transducers required for a specific flight test phase could be connected into one S.W.I. channel thus allowing the possibility of pre-flight programming.

The connecting cable from transducer to S.W.I. box and to the data acquisition system is a 4 poles shielded and twisted cable.

In all cases in which signal conditioning is required adequate signal conditioners are installed near the transducers.

The sampling rates of the individual data channels can be programmed in binary relations as follows: 1 - 2 - 4 - 8 - 16 - 32 - 64 - 128 - 256 sampling per second (S.P.S.) the word lenght is 10 bit plus parity and the system capacity is normally 4096 words/second.

For pulse code modulation (P.C.M.) and to provide an output for analog recording n° 6 S.C.S.A.B. are provided on the aircraft. Each S.C.S.A.B. contains a maximum of 64 signal conditioning modules, necessary amplifiers and analog multiplexing.

Each S.C.S.A.B. is connected to the central control Unit. Furthermore from each S.C.S.A.B. up to 16 channels could be derived which could be connected through a programming panel to the analogic recorder system.

Each S.C.S.A.B. is connected to n° 3 different feeders, one feeds the conditionner, one sends the registration signal to the transducer connected to the signal conditioner and one feeds the recording voltage for special transducers.

N° 3 different types of signal conditioners are provided with inlet voltage from 2 mV full scale up to 40 V.

It is possible to connect internally to the signal conditioners some components with the purpose to complete the transducer circuit (i.e. inlet voltage regulator) furthermore the possibility to connect resistances for zero as well as sensibility calibration is provided.

Excitation of calibration circuits is obtained by means of a S.C.S.A.B. calibration box, which could be installed on board the aircraft or used as ground check unit.

The C.C.U. of the P.C.M. system uses a memory unit to store the programme and supply the maximum flexibility for data "formatting". It checks the gain, the off-set, and sampling of the S.C.S.A.B. output data, conditions and samples 24 x 10 bit parallel inlet, up-to 16 inlet frequencies and 4 inlet period measures.

A remote control unit is connected to the central control unit with the purpose to transmit special data such a flight number, date, pilot number and to select 3 different speed of the central control unit and start to discharge the memory content of the C.C.U. on magnetic tape.

The measure unit is normally programmed by means of a punched paper tape or by a type writer. The same type writer is used to write the memory content having the purpose of check; with the memory unit the sampling speeds are programmed, off-set and signal conditioner gain, sampling speed at the period inlet, parallel digital input and P.C.M. format.

Analogic signals recording with frequency up-to 200Hz is obtained by a constant band with multiplex F.M. system. The system capacity is limited to 21 channels, 20 of which are used for data and one has a check function.

The inlet signal conditionning to the multiplex F.M. signal is obtained via the P.C.M. system signal conditionner.

The parameter connection to the multiplex F.M. system is obtained via the S.C.S.A.B. output analogic signals and the programming panel.

An analogic signal selector permit programming up-to 6 groups of 21 analogic signals to be sent to the multiplex F.M. system. Group selection could be done by the pilot during flight tests.

All measures obtained on board of the aircraft are recorded on a magnetic tape through an analogic magnetic tape recorder.

The recorder is a standard IRIG 6 speed, 14 channels able to record the following information on 5 separated path.

- 1 - P.C.M. system output in serial form
- 2 - Multiplex F.M. system output
- 3 - IRIG A & B coded timing
- 4 - Crew voice
- 5 Control signal

The remaining 9 path are utilized to record whatever high frequency measure is needed. The signal conditioning is obtained similarly to the multiplex F.M. system signals. A codified time generator used to syncronize different data flow is a part of the data acquisition system. The generator sends an IRIG A or B coded time signal for magnetic tape recording and photographic machines syncronization.

Pilot signals are totalized in the codified time information to be recorded on the P.C.M. system.

In addition to the data acquisition system photographic machines are used to film external carriage release, cockpit indicators, etc.

6.3 Ground Station

The Aeritalia telemetry ground station has been set up for the Tornado in flight test programme according to the latest in flight test instrumentation technology.

Basically it consists of:

- real time on line data system
  Real time on line telemetry data acquisition and processing systems which reduces by approximately one third flight test time
- Data systems
  The airborne data acquisition system already described in details in the preceding paragraph comprises
  - time and frequency multiplexing technique
  - timing system
  - data recording
  - telemetry data transmission
- Ground station
  The ground telemetry data processing station comprises
  - telemetry tracking and receiving
  - data recording
  - data demodulation and decommutation
  - calibration and data processing
  - control room display

The block diagram presented in the fig. 12 indicates the flow information realized in AIT.

7. TORNADO TESTING AND TESTING PHILOSOPHY

One of the Tornado preproduction aircrafts carried out hot weather trials in summer time in Sardinia in 1978.

Main purposes besides others, were the environmental system testing and the heat rejection measurements.

A satisfactory evaluation of the flight test data with respect to the heat loads is only possible if there are stabilized conditions over a time range of several minutes during a certain flight phase. Taking this point in consideration some flights have been selected for the analysis among others because of high ambient temperature (take off-landing).

In addition to flight tests an alert stand-by has also been examined to show correlation between calculation, Rig test and Flight Test.

The purpose of this comparison is to update the heat ejection mathematical model in order to be able to extrapolate the flight and ground test results obtained in peculiar environment conditions, to the full range of the flight envelope and therefore to evaluate the capability of the fuel system to fulfill the fuel temperature limitation.

The components of the airframe and of the engine which have an influence on the performance of the fuel oil cooling system are listed below

- Boost pump
- First stage pump
- Air cooled fuel cooler (A.C.F.C.)
- Fuel cooled oil cooler (F.C.O.C.)
- Engine
- Engine gear pump
- Engine oil cooler
- Recirculation control valve (R.C.V.)

For evaluation purposes the herein after flight test instrumentation parameters were used

- Hyd. oil temperature, F.C.O.C. inlet
- Hyd. oil temperature, F.C.O.C. outlet
- S.P.S. oil temperature; F.C.O.C. inlet
- S.P.S. oil temperature, F.C.O.C. outlet

- Engine oil temperature, E.O.C. inlet
- Engine oil temperature, E.O.C. outlet
- Air temperature in S.P.S. bay
- Outside air temperature
- Fuel temp. B.P. outlet
- Fuel temperature F.S.P. outlet F.C.O.C. inlet
- Fuel temperature F.C.O.C. outlet
- Fuel temperature E.O.C. outlet
- Fuel temperature (Fuel Control Unit) outlet
- Fuel temperature A.C.F.C. inlet
- Fuel temperature A.C.F.C. outlet
- Fuel temperature, reheat servo return line
- Fuel Mass flow engine consumption
- Fuel Mass flow recirculation line A.C.F.C. inlet
- Fuel Mass flow reheat servo return line
- Engine rotor speed in percent.

To perform the calculations it was necessary to make some assumptions concerning data not yet measured on the aircraft i.e.:

- the hydraulic oil flow has been assumed to be average leakage flow
- the S.P.S. lubrication oil flow has been assumed to be the supplier suggested flow.

Some doubts about the reliability of hydraulic and S.P.S. oil flow figures appeared and it was difficult to measure the hydraulic oil flow on the aircraft, and worse were the possibilities to measure the S.P.S. lubrication oil flow because of aeration of the oil.

The influence of these errors on the calculated heat loads, for the hydraulic system is small specially during stabilized flight conditions, but revealed some importance for the secondary power system.

Reasons to mention the above are to indicate the difficulties we had to correlate calculation with testing.

To calculate the fuel side heat rejection of the E.O.C. and the heat load of the gear pump engine, manufacturers reports were used.

A constant lube oil flow was assumed to evaluate the oil side E.O.C. heat load. The specific heat of the engine lube oil was assumed to be equal to that of the S.P.S. lube oil.

7.1 Inaccuracy of the Evaluation

The inaccuracy of measurements with thermocouples is about ± 2% of full scale deflection, that of flow meters is about ± 1%.

The flight test data, used for evaluation, are taken from the telemetry traces. The reading of these traces is inaccurate by less than ± 1%.

The measured values have been taken from points of traces near the end of stabilized flight phase, where they can be well read.

Upon completion of the examination of the results of the calculation with the data obtained from testing we are in a position to give a summary of the results of the above analysis considering step by step each point of the circuit.

F.S.P. The heat load is significant below the expected values, based on test rig measurement. The reason for this is unknown, but the inaccuracy of temperature measurements may explain the difference.

F.C.O.C. The heat load measurements on the fuel/oil side shows that the S.P.S. oil flow must be larger than the one assumed. Moreover also the S.P.S. heat load is larger than predicted, whereas the hydraulic heat loads are less because of the stabilized flight phases analized. The total heat load, given via F.C.O.C. into fuel, is in most cases larger than calculated data (fig. 13-14-15-16).

E.O.C. A comparison of the engine heat load measured on the fuel side with those on the oil side demonstrates that the oil flow must be smaller than assumed below 95% engine R.P.M. and larger beyond 95%.
Comparison between aircraft flight test and calculation shows that actual engine heat rejection is larger up to 10 KW at engine RPM 95% and in agreement at engine speed 95% RPM

GEAR PUMP The heat load of the gear pump is less in general by about 30% than predicted, which is in fact confirmed by rig test results of two pumps of same standard

A.C.F.C. The cooler performance data during flight tests are better up to 3 Kw for aircraft speed M 0,5 and worse up to 2 Kw M 0,5.

REHEAT SERVO RETURN FLOW This flow was measured the first time on the aircraft. The values are slightly below the predicted values

S.P.S. AIR TEMPERATURE It was found to be a linear relationship of the S.P.S. by air temperature and the flight altitude and engine speed: the higher the speed and the lower the altitude, the higher the temperature.

In conclusion, the tests carried out both on the fuel test rig and on the aircraft during hot weather conditions have given the possibility to introduce the necessary modification on the fuel system well before the first production aircraft flight. Back-up tests on the rig, following aircraft fuel system malfunction in flight have allowed the engineers to introduce immediately corrective actions. In more details:

- Knowledges concerning engine, gear pump and reheat servo return flow gained by aircraft test flights have shown to be valid for production standard aircraft.
- Engine heat loads data have been changed by engine manufacturer based on aircraft flight experience.
- Analysis have shown a good agreement between aircraft data and calculation with new data.
- Aircraft flights have shown that the inputs for performance calculations have to be changed, to obtain a more accurate evaluation.
- Oil mass flow in S.P.S. and engine, used for calculation, have proven to be unreliable, as comparison of oil/fuel side measurements have shown.
- The oil flow data must be corrected and rig work concerning oil flows and heat rejection must be investigated.
- The above statements are confirmed by computer analysis, the computer programme is able to simulate flight mission realistically. The assumed time constant, to calculate oil temperature changes must be improved to fit the programme inertia to reality.
- From the preceeding documents it appears that the philosophy followed by Panavia in conducting the fuel system project, calculation and testing has given a valuable methodology which could be used in future programs.
- The iterative processus followed i.e. calculation, ground testing and flight testing has demonstrated that the correlation between inputs and outputs is good and the confidence level gained shows that the mathematical model is representative of the actual system behaviour.
- The instrumentation used both for ground and in flight testing being at the latest state of the art, has remarkably decreased flight test costs to investigate the specific system behaviour (same flight, different system data transmission and recording).
- The effort produced by Panavia to shorten time for flight tests and to introduce immediately mods required by systems to obtain the best from them has proved to be satisfactory having followed the above procedure.

ACKNOWLEDGMENT

The information presented are the results of a Panavia team work. The author wants to acknowledge the contribution of MBB and BAe colleagues who have collaborated to realize the Tornado fuel system testing programme.

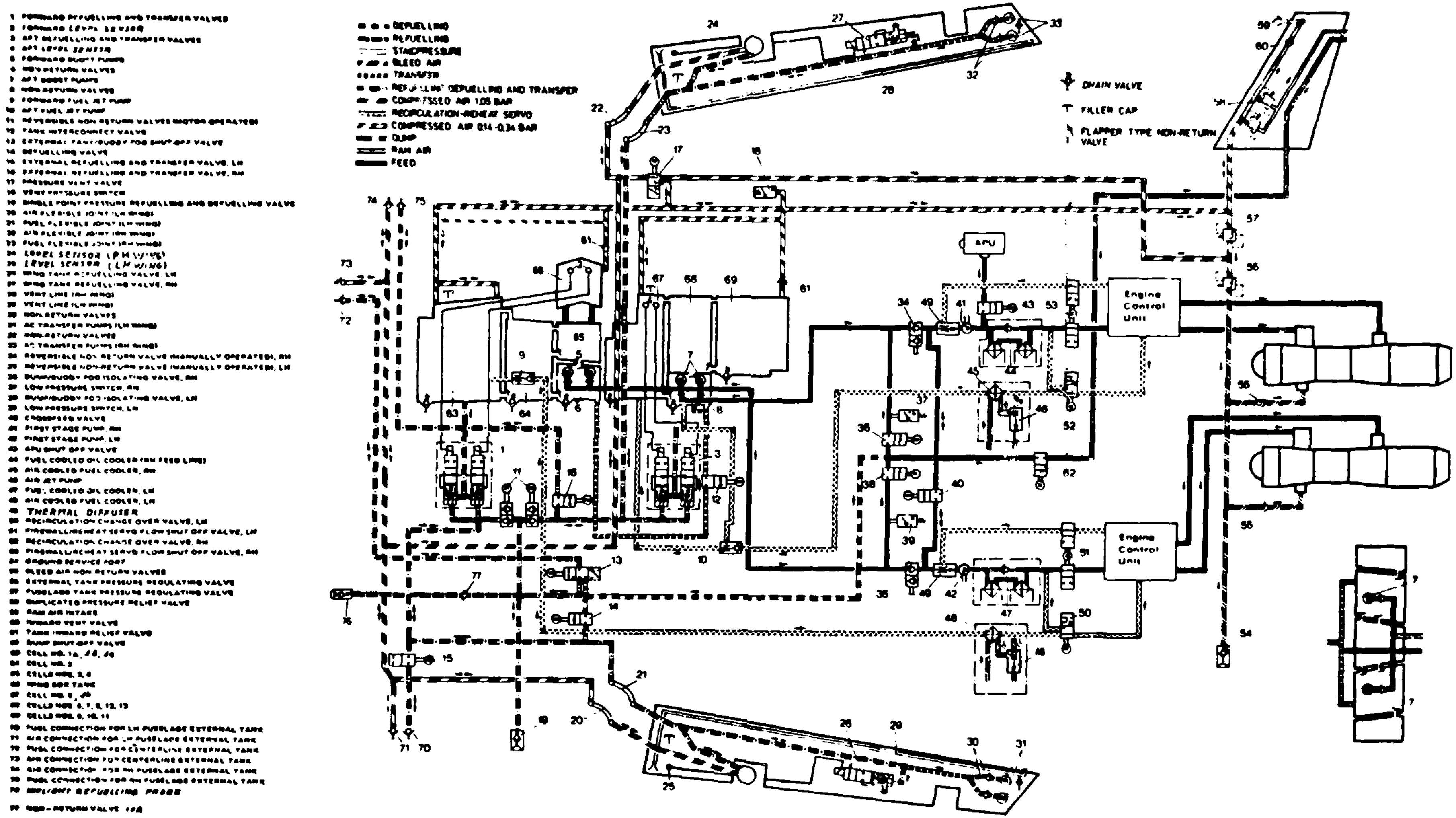


Fig.1 Fuel system diagrammatic

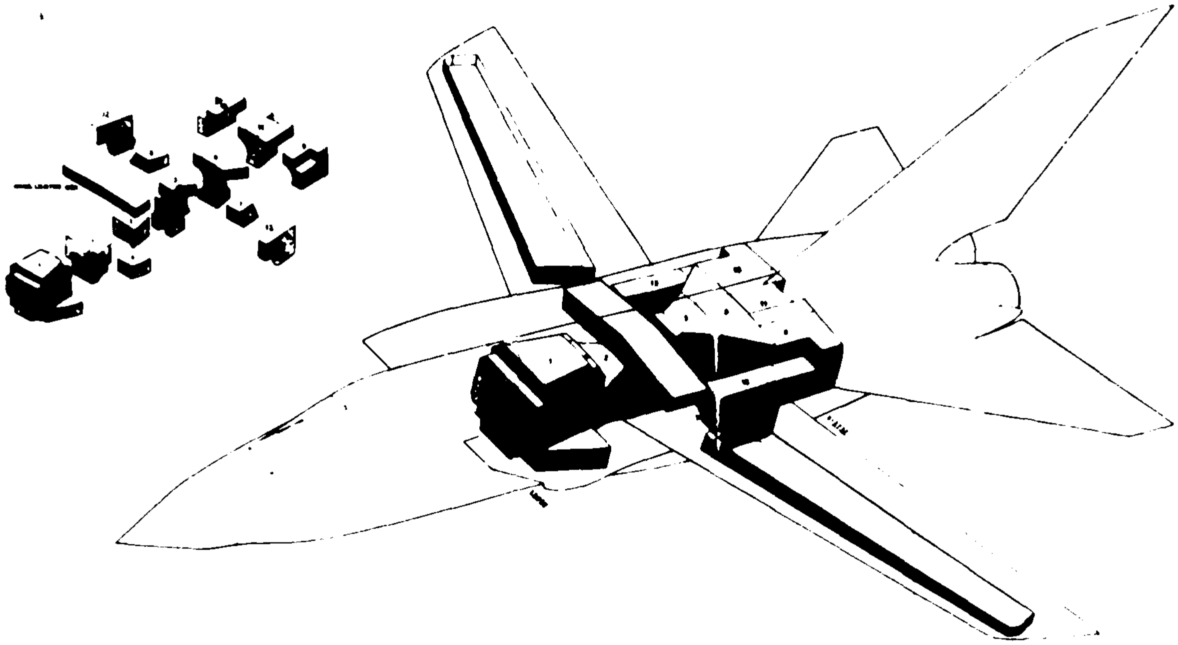
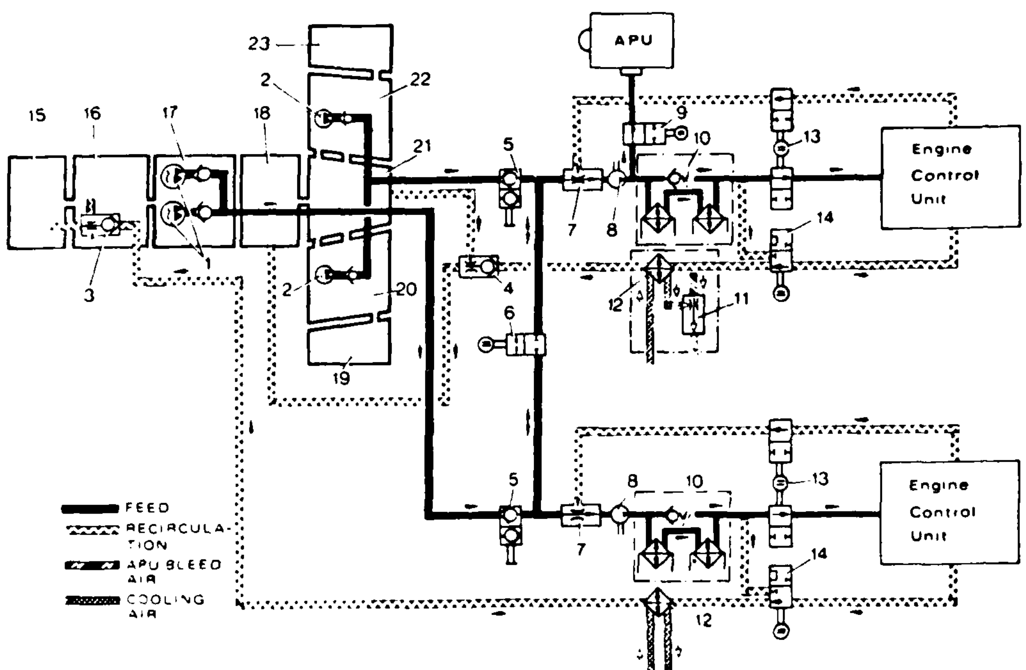


Fig.2 Fuel tank arrangement



1 FORWARD BOOST PUMPS
2 AFT BOOST PUMPS
3 FORWARD FUEL JET PUMP
4 AFT FUEL JET PUMP
5 REVERSIBLE NON RETURN VALVES (MANUALLY OPERATED)
6 CROSSFEED VALVE
7 FUEL MIXING PUMPS
8 FIRST STAGE PUMPS
9 APU SHUT OFF VALVE
10 FUEL COOLED OIL COOLERS
11 AIR JET PUMP
12 AIR COOLED FUEL COOLERS
13 FIREWALL/REHEAT SERVO FLOW SHUT OFF VLAVES
14 RECIRCULATION CHANGE OVER VALVES
15 CELL NO. 1
16 CELL NO. 2
17 CELL NO. 4
18 CELL NO. 5
19 CELL NO. 13
20 CELL NO. 7
21 CELL NO. 6
22 CELL NO. 8
23 CELL NO. 12

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Fig.3 Engine feed system diagrammatic

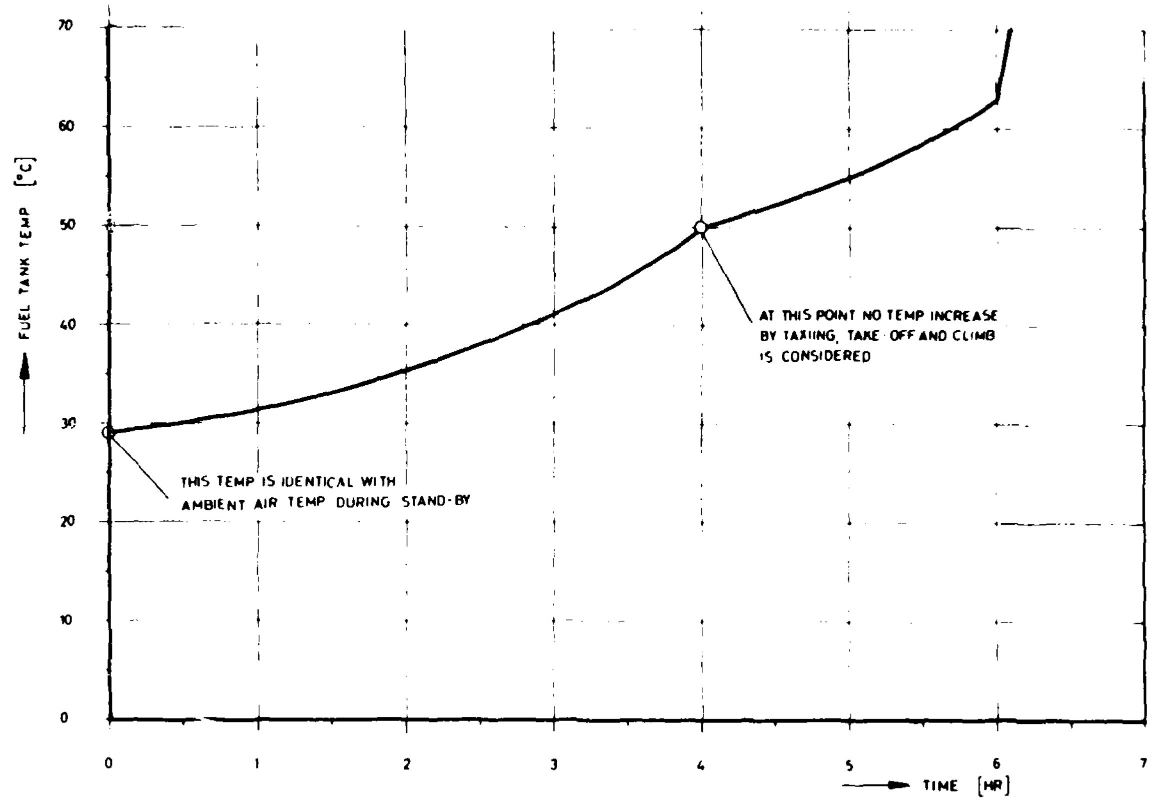


Fig.4 Tank temp. increase during stand-by and flight

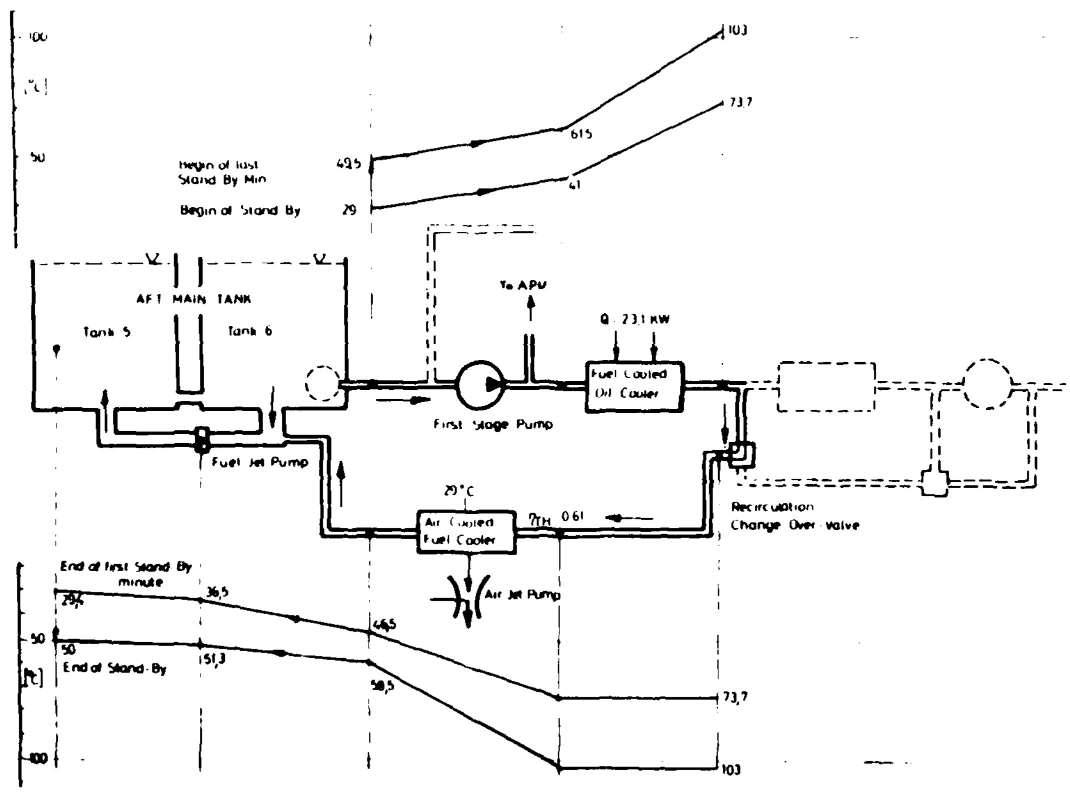


Fig.5 Fuel temperature during stand-by

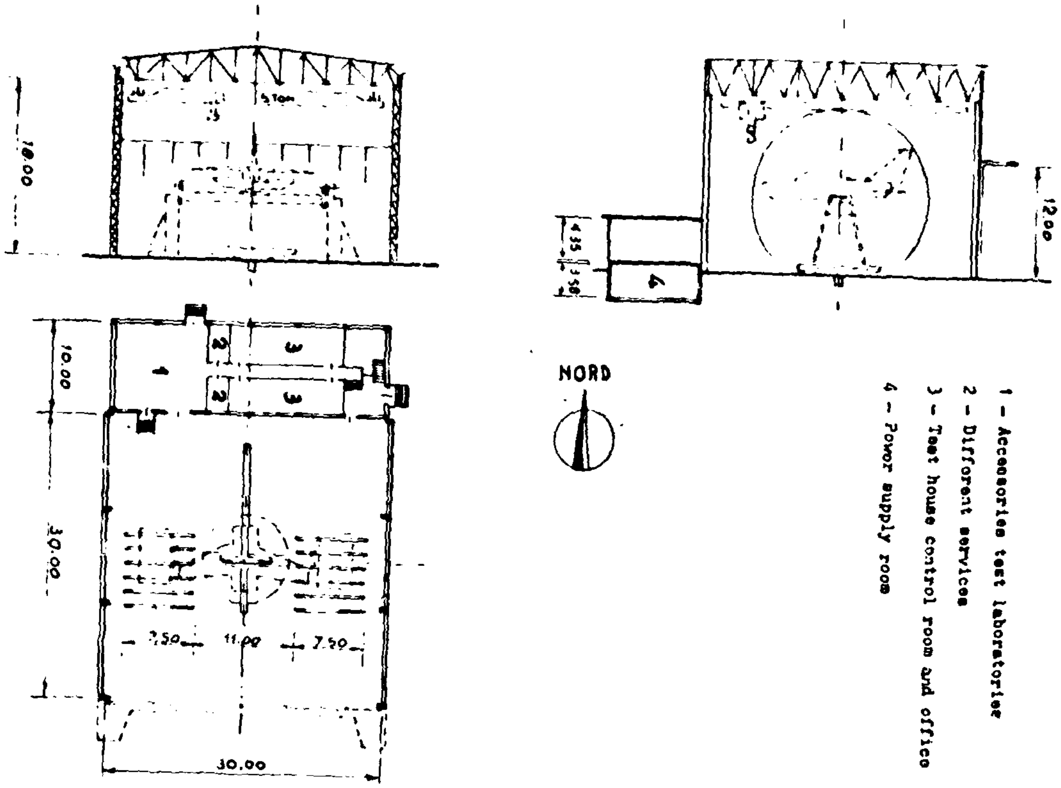


Fig.6 Test rig building

Figure 7

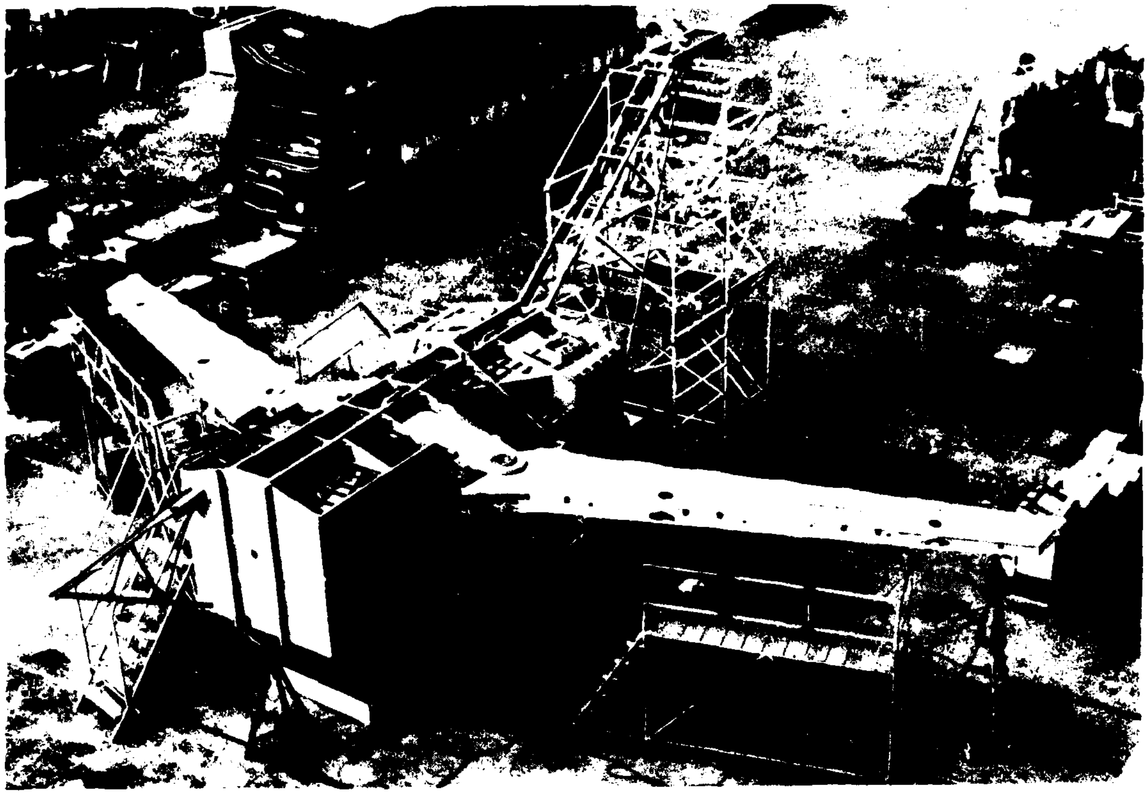


Figure 8

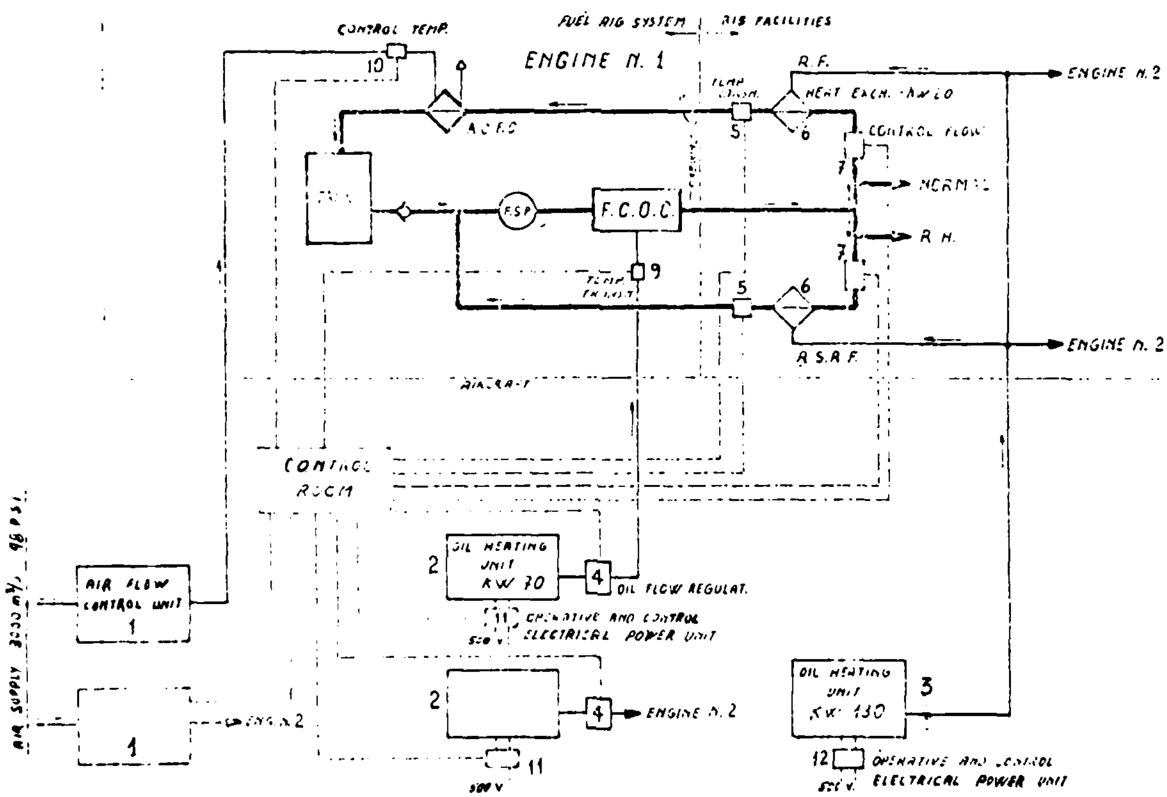


Fig.9 General plant for heating and cooling systems simulation

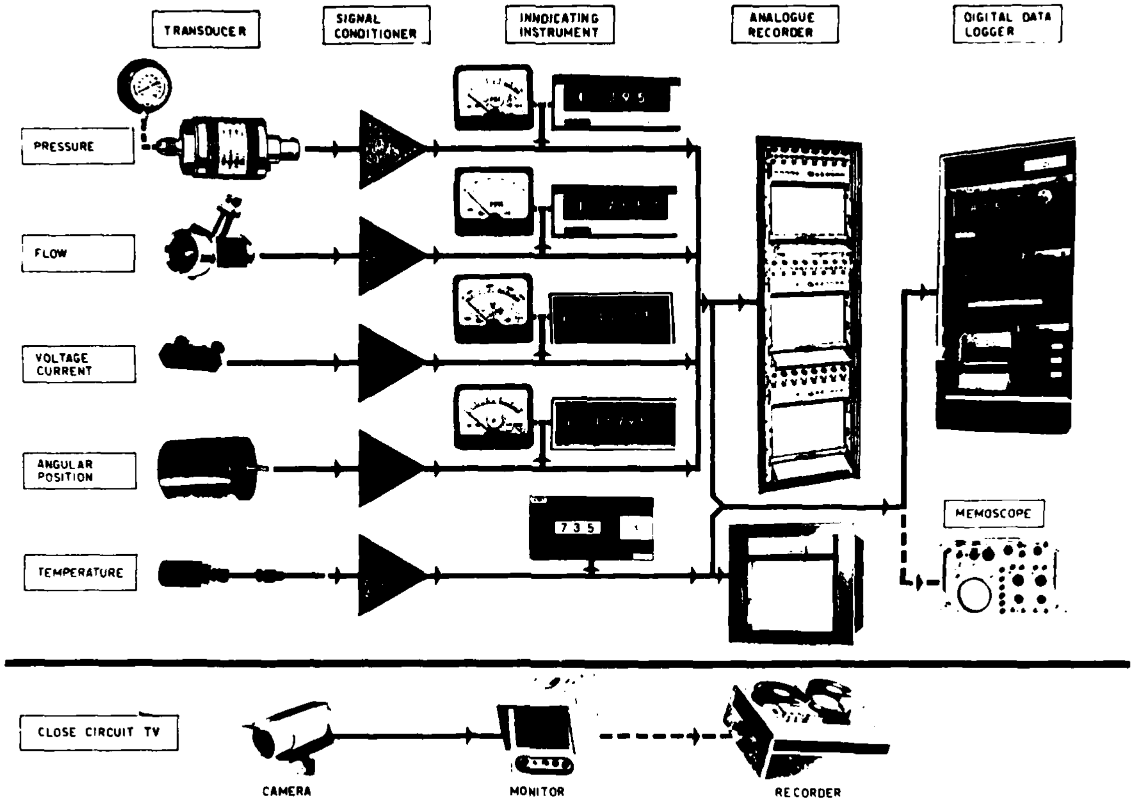


Fig.10 Fuel rig - Test data acquisition system

SENSORS		GAUGING POINTS	INDICATING INSTRUMENTS	ANALOGUE RECORDERS		DATA LOGGER	MAX FREQUENCY RESPONSE
				INK	U.V.		
[illegible]	LVDT	55	55	10		55	
	[illegible]G	[illegible]	/	/	19		[illegible]Hz
FLOW		9	9	9	/	9	
TEMPERATURE		45	/	24	/	30	

MAX ERROR ON THE WHOLE SYSTEM

UP TO ANALOGUE INDICATORS	± 1%
UP TO INK RECORDER	± 1.5%
UP TO U.V. RECORDER	± 1.5%
UP TO INK RECORDER FOR TEMPERATURE	± 2%
UP TO DATA LOGGER	± 1%

Fig.11 Sensors grounding table

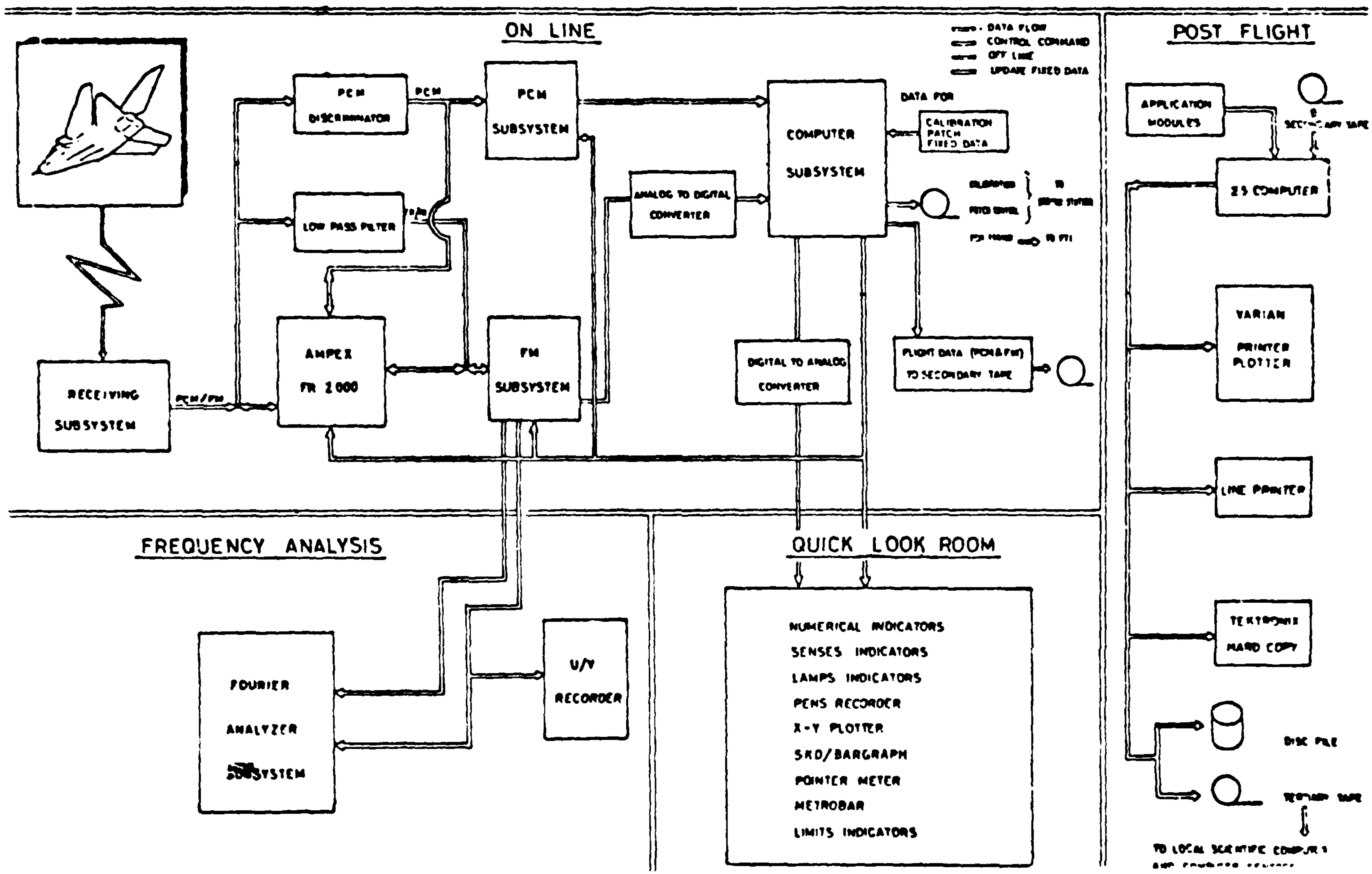


Fig.12 Flight data analysis: Simplified block diagram

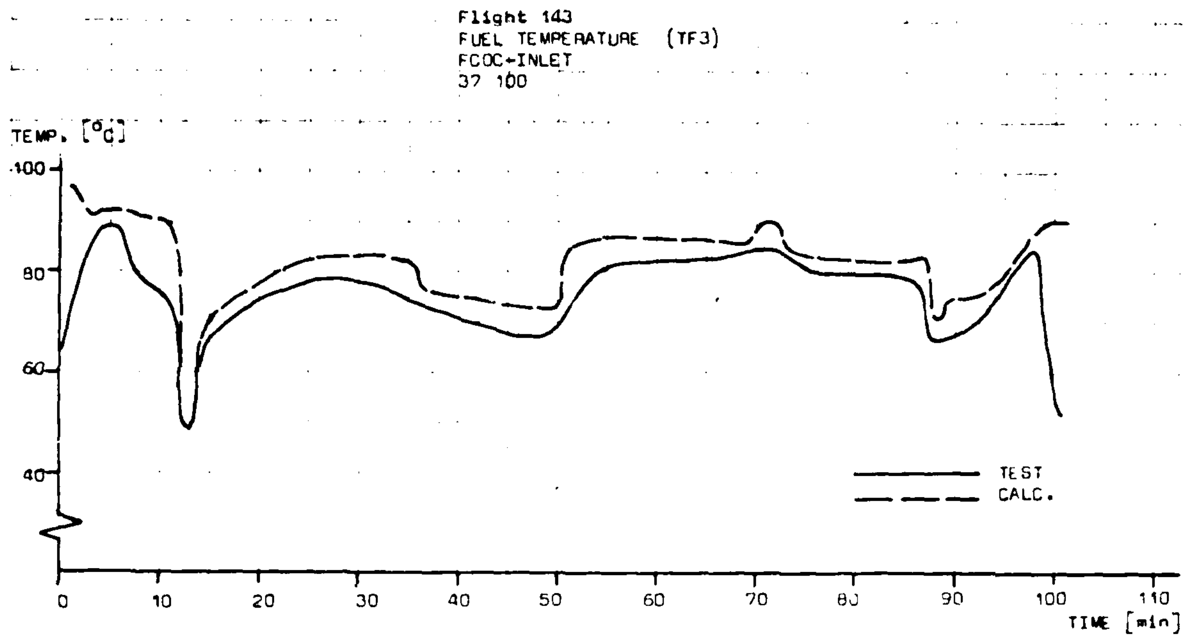


Figure 13

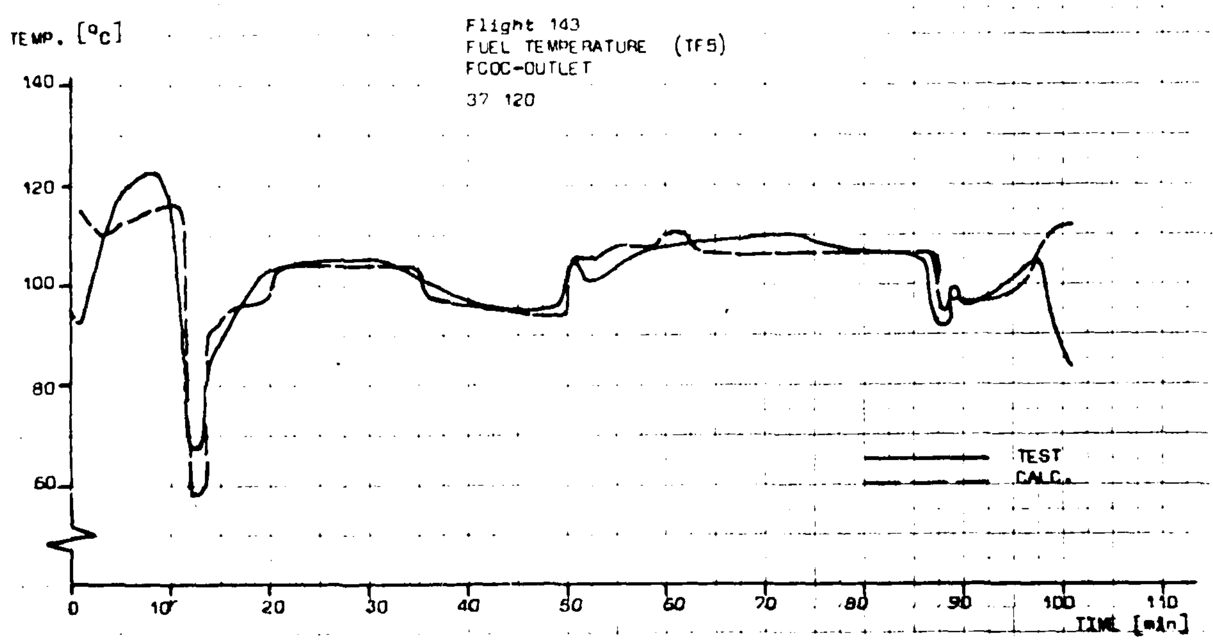


Figure 14

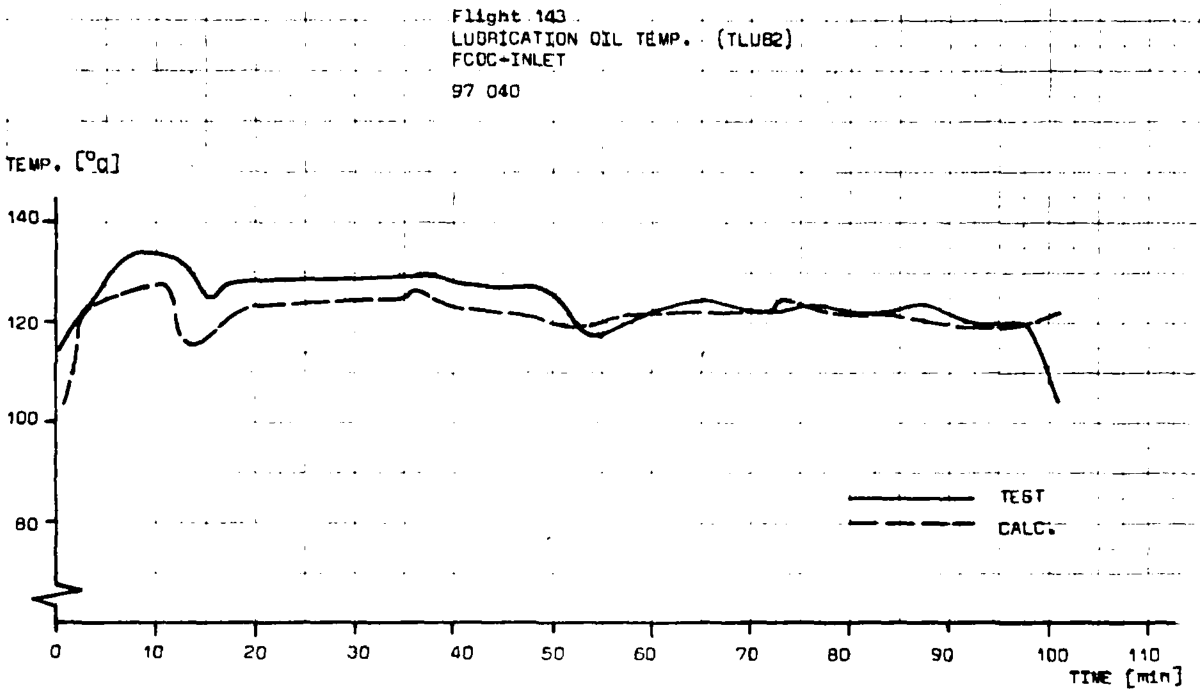


Figure 15

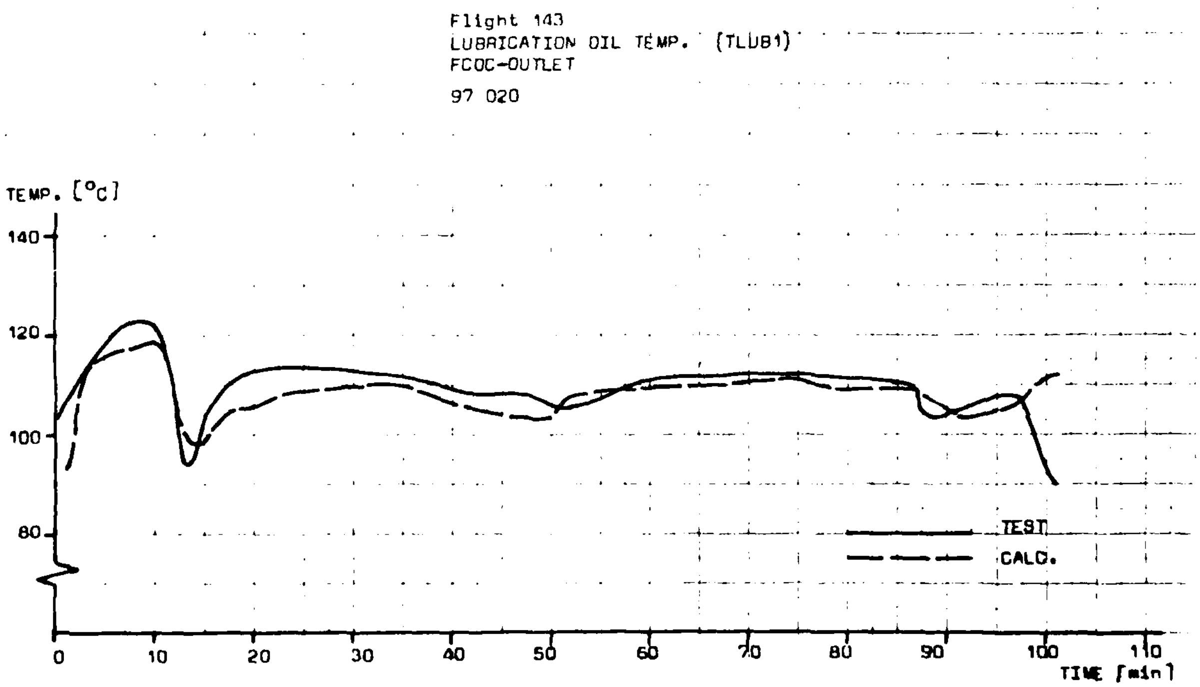


Figure 16

ADVANCES IN LANDING GEAR SYSTEMS

N. S Attri
Airplane Technology Manager
Advanced Airplane Branch
Boeing Military Airplane Company
Seattle, Washington

R. L. Amberg
Specialist Engineer
Boeing Commercial Airplane Company
Seattle, Washington

It may appear that landing gear has changed very little since the early days of aviation. Other than a similarity in appearance, much has however, changed. This paper provides a fond look at these changes and provides a status of landing gear technology.

To develop perspective, perhaps one can begin by a ground air ground cycle concept. Airplanes undergo taxi (ground roll at slow, medium to high speeds), take off and after the mission return to land.

During taxi, tires play a vital role in supporting aircraft weight, undergo tremendous dynamic load, provide steering at low speeds, assist steering at medium to high speeds, transmit loads to runway without sinking or tearing runways and without getting destroyed in the process, and provide economic life. If one should need asymmetrical control, brakes come in handy.

During the high speed taxi or before take off, should it become necessary to refuse take off, the brakes play a vital role in stopping the aircraft. Upon return to ground the brakes again play a major role in stoping the airplane safely in desired distance.

During landing again tires play a role as a part of shock absorption system and brakes assist braking and starting. The process of braking involves interaction of more than tires, wheels, brakes and the landing gear structure. The tires would normally lock up as brakes are applied. A device called antiskid was developed to prevent wheel lockups.

As the size of airplanes grew, the number of tires, brakes and the design of bogies grew more and more complex. Airport runways had finite length 6000 - 12000 feet, depending on traffic, size of aircraft etc.

The large number of people travelling during rain, sunshine, cloudy/windy, snow and other weather conditions expected a safe travel so the braking system had to become much more efficient.

The shock absorption during landing had to be smooth, the taxi ride also had to be good during ground roll.

All these features had to be provided at minimal cost, weight and complexity. The latter aspect is important so that the system could be maintained through years of service.

As the system started to become sophisticated, some problems arose, such as brake chatter/squeal, landing gear walk, truck pitching and landing gear shimmy.

This paper provides a status of development for some of the hardware components and describes the system evalution and addresses the problem of validating expected performance. The matrix chart (Figure 1) shows a relationship between functional landing gear sub-system and the hardware components The interaction between components and functional subsystem will be discussed during subsystem discussion.

Other subsystems and problems discussed include:
- o Antiskid system/automatic brake system
- o Aircraft steering and ground handling
- o Shock absorption, ride comfort and rough runway operation
- o Brake squeal, truck pitch, shimmy

WHEELS & TIRES

No attempt is made to provide detailed discussion on wheels or tires as these components find considerable discussion in popular literature and supplier proposals. The following comments are in order for wheels.

What is on the Horizon for Aircraft Wheels

Continued heavy dependence on forged aluminum wheels is foreseen. Steel wheels are no longer given serious consideration, and titanium wheels, while practicable, are still quite expensive. A graph of cost per roll-mile for aluminum and titanium wheels illustrates the advantage of aluminum wheels (Figure 2). The industry can, however, furnish wheels fabricated of titanium if the operator will accept the higher initial cost. Most of the premium for titanium wheels results from the expense for the forging itself. Current quotations for titanium, closed-die forgings are 10 to 11 times those of forged aluminum material. Another problem is the present state-of-the art of titanium forging tolerances. No one has been successful in producing a titanium aircraft wheel forging with the precision obtainable from an aluminum forging. Machining of all surfaces is thus required to control weight and obtain the desired form.

The aircraft tires carry out many functions as will be discussed in the ensuing sections of this paper. The tire static and dynamic properties play a significant role in optimizing landing gear system performance. Table 1 shows some of the tire properties. No suitable source exists for dynamic properties of tires. NASA-TR-64 is perhaps the best available source for static properties and is widely used by landing gear designers.

BRAKES-CURRENT STATE-OF-THE-ART

Brake Description

The conventional high performance aircraft brake consists of multiple friction discs of alternate stationary and rotating members. The rotating elements (rotors) are keyed loosely to the wheel to allow axial movement. The stationary elements (stators) are keyed loosely on the inside diameter to a fixed stationary member and also are free to move axially. Brake lining is applied to either the rotating or non-rotating members as a given design dictates to improve the frictional characteristics. Upon brake application, the rotors and stators are forced together, causing a frictional drag that provides a retarding force to wheel rotation.

Figure 3a shows an exploded view of a typical fighter brake assembly. Item 1 illustrates a rotor of the segmented type. Figure 3b illustrates several types of rotor configurations used in current aircraft. The rotors constitute a large percentage of the reservoir capacity for the storage of heat generated during braking. The heat is generated at a very rapid rate, is stored in the heat sink, and then is slowly dissipated to the atmosphere and surrounding hardware.

Progressive shrinkage, induced by alternate cycles of high heat and subsequent cooling, result in high residual hoop stresses in a nonsegmented rotor. These stresses, depending on sign, result in one of two types of distortion. One type distorts the disc into a conical washer, and the other type produces a wave washer effect. Both types of distortion are disastrous to the brake, since they eliminate running clearance. This culminates in a dragging brake which rapidly becomes overheated and could result in loss of brake capability or a fire.

To minimize this distortion radial slots are incorporated, resulting in partial segmentation in some designs. Other designs are completely segmented and the detrimental effects of shrinkage are eliminated.

For most design applications the stator plate is the lining carrying member. The lining is attached by either riveting cups to the plate or by bonding the lining directly to the plate. The torque resulting from stator-rotor friction is then structurally reacted by keys attached to a torque tube (Item 9) which in turn introduces the torque into the axle flange of the airplane. The pressure plate is the same general configuration as the stators, but differs in plate thickness. Increased thickness is desirable to insure even distribution of the clamping force, produced by the actuating pistons, over the entire rubbing surface of the friction discs. Lining is attached to the one rubbing face of the pressure plate. This plate also accommodates the return spring pins which, when assembled in the brake with the return springs, pull the plate back to the carrier when pressure is released from the pistons. The resultant movement forces the pistons back into their bores and provides a gap between the lining and the adjacent rotor. This gap assures running clearance for the balance of the friction stack and eliminates any generation of torque when pressure is not applied.

The backing plate (Item 3) is an important structural member of any disc brake. It provides the reaction to the actuating piston or pistons. Lining is attached to the surface of the backing plate, adjacent to the last rotor; thus providing a portion of the overall brake torque. Due to its mass for structural rigidity purposes, the temperature of the backing plate always remains much lower than stator or rotor temperatures during any brake application.

The prime function of the carrier (item 5) is to provide the actuating force for clamping the friction stack for torque generation. The actuating force is created by hydraulic pressure applied to a single ring-type piston (see items 6, 7, and 8) which is accommodated by an annular groove machined in the carrier. The carrier must be pressure-tight and rigid. Deflections decrease available piston travel. Item 7 is a rubber seal to prevent hydraulic fluid leakage past the piston (item 6) while item 8 is an insulator to prevent heat from the pressure plate (item 4) from deteriorating the rubber seal. Other carrier designs utilize multiple pistons spaced around the circumference of the brake friction radius.

Four automatic adjusters (item 10) are mounted to the carrier of this brake and attached to the pressure plate by means of the adjuster pins. The adjuster provides a constant running clearance when the brake is released.

In recent years composite carbon brakes have made considerable strides. Several military aircraft such as F-15, F-16, F-18 and B-1 prototypes now have carbon brakes. These brakes are light and offer long life potential. Initial problems included oxidation causing sudden loss of strength, moisture causing temporary loss of braking and high initial cost. The oxidation problem has been minimized, but torque variations due to moisture and other caused still are a source of problems particularly to system integration. The rapidly rising cost of fuel, along with rapid strides in composite carbon brake technology assure wide spread of this type of brakes.

SHOCK STRUT

The primary purpose of the landing gear structure is to support the airplane on the ground. In addition, it is called upon to absorb and dissipate the landing impact energy and provide ride comfort during ground operations. The shock absorption is accomplished by an air over oil shock strut.

The basic weight support function is provided by a compressed cylinder of air. The volume, cross-sectional area and extended gear pressure determine the static spring rate of the shock strut. These parameters have a direct bearing on the ride comfort during taxi. In addition, they determine the stroke during landing.

The energy dissipation is accomplished by forcing the oil through an orifice, whose area, is made variable through the use of a variable diameter metering pin. The damping is therefore, a function of shock strut stroke.

Attempts have been made to provide optimum energy dissipation and ride comfort through the use of dual air chamber struts. Development work is also being done on active shock strut designs which allow variation of characteristics as a function of its operating point. The challenge is to develop a system that is as simple and reliable as existing designs.

STEERING SYSTEM

The steering system consists basically of a hydraulic metering valve, an actuator to rotate the nose wheels, positional feedback to valve and an input mechanism from the pilot. The system operates as a position servo, causing the nose wheels to be positioned according to the pilots input.

The overall steering problem, though, is much more encompassing. The actual turning of the aircraft is dependent on the weight distribution between main and nose gear tires, tire characteristics, aerodynamic characteristics and airplane dynamics.

BRAKING SYSTEM DEVELOPMENT

Antiskid systems were originally intended to prevent prolonged skids and subsequent tire blowouts. This basic role has been expanded to include optimization of stopping performance under a variety of runway conditions e.g., dry, wet, icy, etc. The development cycle of a brake system consists of many actions which occur at different times. The earliest item which gets firmed up is the location of nose and main gears. The typical range of the center of gravity during operational regime is also fixed. These items determine the main and nose gear landing gear location which have a major influence on braking and directional control. Other information available at this stage includes the expected tail size and basic aerodynamic characteristics necessary for calculation of aerodynamic performance and airplane stability during flight. Adequate heading control during takeoff in the presence of an engine failure, is an important consideration and impacts tail size.

The early landing gear design efforts are mainly aimed at configuring a convenient way to stow the gear without excessive complexity of linkage, failure modes and incurring any drag penalties. The system performance can be impacted by early landing gear design efforts. The gear attachment structure and gear height can impact gear fore and aft stiffness and weight transfer during braking. The basic configuration and torque reaction method can influence the truck pitch dynamics. About this time the shock strut characteristics also become matters of concern. The available stroke is nearly firm and all that needs to be done is to utilize it to minimize loads under extreme design conditions. The design sink rate is typically 10 ft/sec for commercial transports while operational landings seldom exceed 1-2 ft/ sec. The braking system could greatly benefit if special attention is paid to the operational regime during the early trades used to configure metering pins. Special emphasis should be placed on minimizing rebound.

Once the landing gear configuration takes firm shape other gear components also begin to be considered. The earliest item besides landing gear structure which receives attention is the brake since it is a long-lead item. Once developed the brake does not lend itself to adjustment of its basic characteristics. The primary consideration in brake development is the heat sink required to meet the kinetic energy requirements during stopping, both landing and refused takeoff. This aspect deserves careful evaluation to assure adequate brake life in service.

The other aspects which have received little or no attention pertain to the brake torque requirements and other associated dynamic characteristics. The consequence of insufficient torque are obvious, so the designer may be tempted to specify an excessively high torque capability to make sure he avoids the problem. Unfortunately, the approach can induce a difficult skid control problem on slippery runways (icy, wet, etc.) The skid control system will be forced to control at extremely low pressures which may be in an area of the brake pressure/volume curve that is highly nonlinear, resulting in reduced efficiency. Care must also be taken to minimize the nonlinear area of the pressure/volume curve to insure satisfactory response in the low pressure region. The type of brake lining selected can also impact brake dynamics, gear dynamics and brake control system response. The noise characteristics of the lining selected should be carefully assessed to assure a squeal and chatter free brake.

The tire is another crucial element which is selected early in design cycle. The primary consideration is its load-carrying capability during the speed regime normally applicable for landing or takeoff cycles. The tire specifications provide roll tests simulating landing, taxi and takeoff loads and speeds. The tests are largely tire structural integrity tests and very little is known about tire wear, cut resistance, or its traction characteristics after the qualification tests are complete.

Other considerations in the selection of tire include flotation and stowage requirements. The traction characteristics are seldom considered. This aspect is important and requires a careful trade study. A high pressure tire results in favorable wet runway traction and helps the designer in reducing his stowage requirements, but it increases tire wear on dry runways and aggravates flotation. A tire custom tailored for each aircraft provides the best solution. The tendency to select an off-the-shelf tire is motivated by cost savings in procuring original equipment and perhaps assuring commonality with other fleet. The problems arising from such trades seldom justify this penny-wise approach.

With the release of major landing gear design drawings and specifications related to tire, wheel and brake, attention now turns to other aspects of the system. The brake hydraulic system is configured along with the steering hydraulics system. The line lengths and sizes are firmed up to assure adequate power for various flight controls and auxiliary functions. This is the time for landing gear designer to make his requirements known and to receive special consideration consistent with his needs. The proper sizing of the hydraulic system at this time will assure adequate system response and avoid costly and difficult changes later in the development of the system. The system redundancy, power supplies, indication system and other interface requirements are also firmed up.

It is now time for release of skid control and steering component specifications. The discussion of this specification and the role it plays in successful design will be relegated to a later section so as to provide some discussion of the many variables which impact skid control performance. Proper understanding of these variables and their role is an important prerequisite to writing a good specification.

The environment in which the aircraft operates places further constraint on system design. So far we have discussed constraints imposed on the system during early aircraft design. These are necessary as the aircraft is mainly designed to be an efficient flying vehicle.

Among the elements that require consideration are:

1. Runway traction
2. Pilot technique

Runway Traction

The various aircraft may operate into a variety of airfields with commercial operations involving primarily asphalt and concrete runways. Some commercial operators do operate into grass and gravel fields. Military operations often necessitate use of quickly prepared rough-dirt fields. The diversity in runway materials provide a wide variety of traction characteristics.

Concrete and asphalt runways provide good flotation, runway life and traction when surfaces are dry. In the presence of ice, snow, slush, and rain the available friction capability degrades. In addition, on wet runways a phenomena call "hydroplaning" can occur. The incidence of hydroplaning can be reduced by selection of suitable tire design, improvement of runway micro and macro texture and a good runway maintenance program.

Poor runway design can result in an almost complete absence of friction at high speeds on extremely wet runways. This situation may cause a loss of the locked wheel protection devices which normally assure safe operation. The various studies conducted by NASA, FAA and USAF have verified this and runways with good traction have been stressed.

Several new runway concepts are being actively studied by USAF such as porous surface runways. FAA favors the use of grooved runways. These runways provide a positive path for water to drain away and thus offer higher traction under similar rain condition. A good maintenance program is considered integral to any of these concepts. In the absence of such a program, rubber deposits and other contaminates can build up. In the presence of moisture this results in slippery conditions.

Several excellent studies have been undertaken to acquire a better understanding of traction and the landing gear designers will greatly benefit from acquiring better understanding of this problem.

Unprepared Fields

Generally speaking, the unprepared runway traction is not too well understood. The available data is scant and more test data is needed to develop efficient systems for operation of such airfields. The available cornering data is sketchy under all conditions of operation. The available data indicates that traction on unprepared fields is marginal. The presence of excessive moisture causes degradation in both flotation and traction capabilities of the soil. Grass cover also makes runway very slick in the presence of water. Good drainage is necessaray to provide improved traction. This can be accomplished with a surface layer of gravel provided no ruts or low spots are allowed to develop. Soft fields add another dimension to the problem.

Impact of Pilot Technique

Besides elements such as runway traction or crosswinds, the pilot is perhaps the most significant external influence. The pilot influences the landing and takeoff field length requirements in several ways.

One of the elements of concern is late touchdown which uses up much of the available runway thus reducing the margin of safety. This problem is merely the inverse of another problem in which he may land short. Thus, precise landing aids permitting an all-weather landing capability are needed.

For purposes of this discussion it is assumed that such a problem is recognized and will be handled before too long to the satisfaction of all concerned. Even if a precise touchdown is assumed the pilot still can land hard and at excessive speeds. The landing technique, besides losing precious available distance, may cause undesirable rebound and make early braking and cornering difficult. The landing gear, thus, has to be designed for some abuse margin and metering pin analysis should carefully assess operational performance along with critical design condition where peak load values are the primary concern.

Whereas the pilot exercises braking by merely depressing the brake pedals the timing and manner in which he uses these is critical. On dry and good traction runways he can depress pedals firmly to help achieve optimum stopping. On wet runways, where excessive water is present or on very slippery runways (icy) he should apply brakes gradually. By partial, but early brake application, he can get some braking while preventing wheel lockups. As the skid control system learns, the pilot can increase brake pressure by depressing pedals further.

Even though good runways may be available at sometime in the future and pilot advisory information and training may be improved, the skid control systems must be designed to cope with varying conditions of available traction. Presence of paint markings, tar strips and puddles along with differences in micro-texture/macro-texture and drainage will necessitate continuous updating in technology.

BRAKING/STEERING

It was mentioned earlier that braking and cornering are related. This arises from the simple premise that same vector must be shared for traction and cornering, Figure 4. As demand in braking mode increases, the vector available for cornering decreases. Fortunately, as demand is placed in the cornering sense the skid control system will reduce braking effort due to its inability to assess excessive slip in the directional sense. The directional control of aircraft may be divided into high, intermediate and low speed regime. At higher speeds considerable aerodynamic control is available, but as speed decreases to the intermediate range, nose gear steering control becomes the only effective means of controlling aircraft heading. At low speeds all turning is accomplished using the steering system.

The steering system requirements have not been developed to a level of sophistication seen in skid control system design. The low-signal response (critical for high-speed regime) and large-signal response requirements (critical to low-speed maneuvers) are not well established. Due to the nature of directional control which requires several inputs from the pilot as against a simple brake pedal input for braking, the response cannot be easily assessed.

Recently, ground simulators have been developed which permit pilot evaluation of steering system adequacy and workload. A schematic of a ground handling simulator is shown in Figure 5. These simulators are still not in wide-spread use. The comments such as "All simulators are alike" or "A moving base simulator is better than a fixed base simulator" clearly evade the basic issue: The need for extensive and accurate math models such as in use on skid control simulators, and the need for adequate verification by pilot demonstration and flight test.

At this time it ... haps timely to re-emphasize tire data needs for such work. The tire cornering and combined cornering and braking data is nearly non-existent. Data needs for shimmy analysis also deserve mention. Nose gears have been frequently found to shimmy and steering valves have been often used to provide shimmy damping. The availability of tire data and new innovative approaches are needed to resolve the shimmy problem. Presently shimmy damping requirements often conflict with good high-speed directional control.

With the foregoing background the development of efficient skid control systems can now be discussed.

The procurement of the skid control system begins with the release of the specification, which outlines the typical system interfaces - hydraulic, electrical and structural. Besides this, several system configuration requirements are imposed on the antiskid system manufacture. Included in these may be requirements for:

- failure indication
- touchdown and locked wheel protection
- locked wheel protection - turning interface
- parking brake system
- automatic spoiler deployment

Although these features are not a direct part of the braking force optimization system, careful consideration of these requirements is extremely important. The influence of some of these requirements can have a major effect on stopping performance, particularly under adverse runway conditions.

The skid control specification may also impose direct constraints on the type of system used.

Recently, emphasis has been placed on providing more detailed accounting of tire, brake and other associated component performance in the specification to assure harmonious operation of the skid control system.

Since the development of an antiskid system from scratch is both time consuming and costly, the selection of a system is usually limited to an existing system or a modified version of an existing system. Currently, operational aircraft use systems which cover three generations of antiskid development. Figure 6 shows the improvement in system efficiency gained through system development as a function of available ground friction coefficient. Current third-generation systems provide high efficiency over a wide mu range, under ideal conditions.

One of the recent advances in brake control systems which deserves mention is the advent of the automatic braking system. This Boeing developed system concept provides:

- Automatic application of brakes upon touchdown
- A controlled deceleration to stop

The following represent some of the payoffs of this concept:

- Reduction of pilot work load
- Consistent early brake application
- Consistent operational stopping performance
- Smooth, comfortable operation
- Improved cornering capability
- Reduced tire wear

This type of a system is now available on advanced B-737, B-747, B-727 and is currently being developed for use on B-757 and B-767.

Additional performance gains can be obtained through both refinements to existing systems and the development of new concepts. In both areas, the airframe manufacturer is expected to play the key role since he is closely related to the system's operation, service problems and is ultimately responsible for the safety of the aircraft. The antiskid supplier on the other hand is more closely related to the hardware and, with proper guidance from the airframe manufacturer, is in a better position to facilitate component improvement.

Laboratory System Development

The overall system performance depends to a great degree on the work done during the laboratory development. With the airplane, landing gear and brake design essentially finalized, the skid control system components must be optimized to provide the desired performance.

The key element in optimizing performance of the skid control system is the simulator. The purpose of the simulator is to evaluate the system under all operating conditions. Figure 7 shows a block diagram of a typical simulator in use at Boeing. The simulation consists of a computer simulation of the airplane and a hydraulic mockup. As much actual hardware as possible is used to insure complete representation and to minimize computer requirements. Simulated hardware is used initially, being replaced by prototype hardware as it becomes available.

The computer modeling includes:

- 3 Degree of Freedom Airplane
- Engine Thrust Characteristics
- Aerodynamic Characteristics
- Landing Gear Dynamics
- Tire, Wheel and Brake Dynamics
- Ground Force Generation
- Truck Pitch Dynamics (if applicable)
- Brake Torque with Associated Energy and Velocity Influences

The brake and tire dynamics form a major part of the simulation because of their basic influence on stopping performance.

The simulator is used throughout the design process. It must be used early in the process to avoid being locked into long-lead time hardware. Thus, the simulator is used to screen system concepts as well as system components prior to vendor selection, thus assuring the capability to meet the design requirements.

Once the hardware has been procured, the system integration and optimization can begin. The test outline used during tuning and system development must be carefully developed to cover the entire anticipated regime of operation. This must include not only stabilized stops, but also step mu changes, wet runway operations, touchdown reactions and environmental extremes. By means of the simulation, the stopping system is given a thorough preflight check for both dynamic stability and stopping performance. The critical control parameters of the antiskid control box are adjusted to insure an adequate stability margin, assuring "gear walk" free braking operation. Other dynamic influences such as truck pitch tendencies have to be assessed carefully and compensation necessary is provided in the electronics circuit.

Any simulator test must go hand-in-hand with flight test. Unless the simulator reflects what is happening on the aircraft, the testing done on it is of little value. Likewise, the cost and difficulty in precisely controlling all variables during flight test make it virtually impossible to optimize a current skid control system on the aircraft. In addition to performance feed back, pilot opinion of system operation during flight test is very important.

LANDING GEAR SHIMMY

Self-excited vibration in a landing gear (shimmy) is initiated and sustained by the ground excitation forces. The problem is further complicated by increased emphasis on reducing the weight of the landing gear system in modern aircraft and the effects of this emphasis on gear structure. Because of its frequent occurrence and the seriousness (cost, safety, etc.) of the problem, analysis for the prediction of shimmy or landing gear instability is an important part of landing gear design. It also serves as a guide to shimmy testing programs. However, often only limited success is achieved in shimmy analysis, mainly because of over-simplification of many variables as well as lack of meaningful data on tires. At least two forms of shimmy have been discovered. These are:

- tire yaw shimmy
- structural torsional shimmy

The occurrence of both forms depends on the amount of overall system damping. For low damping, shimmy usually involves rigid-body torsional motion of the landing gear restrained by the yawing motion of the tire. That is, the model mass is essentially pivoting about the swivel axis, and the effective spring rate is the tire dynamic-torsional spring rate, accounting for the reciprocal distance of the footprint lateral load line of action from the swivel axis. This type of shimmy is referred to as tire yaw shimmy.

With high damping, tire yaw shimmy is stabilized but the structural modes of the gear can become unstable. One commonly occurring mode has essentially the same model mass as the first mode; however, the spring rate is the torsional-spring rate of the landing gear with the damper (if any) locked. This type of shimmy is often referred to as structural-torsion shimmy. Both types of shimmy are affected significantly by the tire parameters.

CURRENT TECHNICAL APPROACHES

Analysis of shimmy must consider the following:

- the nature of airframe flexibility
- the nature of the attachment of the gear to the airframe
- flexibilities of the attachment and gear structure
- the interaction or coupling between the airframe and the gear structure
- the nature of excitation forces and moments
- the presence of freeplay at all support points and joints
- the interaction between the structure and the damping device

Attachment Considerations

The inner cylinder is generally idealized as a beam, with the bearing plate connection consisting of a number of stringer elements (axial load only) per bearing plate. This provides the required bearing area, as well as preventing swivel moment transfer from the inner to the outer cylinder through anything other than the torque links. The landing gear structure is sometimes extended to include the landing gear support beam which is sometimes pinned at both ends. Other support points may include the rear spar end of the trunion (free to rotate about the trunion centerline), the side strut apex (free to rotate in the plane of the side strut), and the actuator support (completely fixed). Simplifications are also essential due to the presence of a steering system (on nose as well as steered main gears). The steering cylinders are generally idealized as stringers.

Boeing has used, with considerable success, the concept of influence coefficients in evaluating structural influences. The Boeing tests consisted of mounting the gear and support beam in a jig, applying loads and measuring deflections. The data was plotted and reduced to influence coefficients. Sufficient loading conditions were tested to provide data crosschecks, and these indicate that the data is quite accurate. The data obtained from plastic models provides the best comparison to the data obtain for this test.

In addition to their use in the shimmy analysis, these data can also provide a basis for developing a mathematical model to determine the influence coefficients of other landing gears. Moreland was the first to suggest the use of the transfer function approach to evaluation of the structural influences, although he never included it in any of his analyses. This approach still merits consideration.

Presence of Freeplay

Due to the presence of a large number of moving and restrained support points and a rather severe impact and vibration environment, the development of freeplay at some support points, such as the trunnion, is inevitable.

This freeplay has a significant effect upon the magnitude of the perturbations which an otherwise stable landing gear can tolerate.

Trail Stabilizing or Destabilizing

Several investigators have considered the contribution of mechanical trail to landing gear shimmy. The derivative of the critical damping ratio with respect to the trail generally can be either positive or negative. It thus follows that increasing the trail from zero may require either more or less damping which will depend upon the relative magnitudes of the influencing variables.

SHIMMY DAMPER

A shimmy damper is often an agency requirement for averting the occurrence of shimmy. Damping as such, however, is not a certain means for avoiding instability. The system stability or limit cycle can be assured, however, by use of an appropriate type and amount of damping. Analysis of the 737 main landing gear showed that both excess and deficit damping would result in an unstable gear. The choice of a V^2 damper is not recommended for all configurations.

TIRE

The importance of the tire to the system and the shimmy phenomenon is not equally appreciated by various investigators. In recent years, however, the importance of the tire in this self-excitation phenomenon has been more and more recognized. The difficulty in incorporating an appropriate tire model in the shimmy analysis lies in the lack of knowledge about the dynamic behavior response of a rolling tire to an oscillatory side and angular motion. In most theories the tire is represented as a linear a torsional and/or lateral spring-damper system for which static or low speed rolling tire data are used.

BRAKE SQUEAL/CHATTER

Brake induced vibration is caused directly by the normal operation of the brake, as opposed to vibration excited by other sources such as runway discontinuities, and antiskid operation.

Brake induced vibration can be divided into two frequency ranges:

a. Low frequency (less than 100 cps)
b. High frequency (more than 100 cps)

The low frequency vibration is called chatter and is generally related with the fore and aft movement of the landing gear strut. This strut movement is commonly referred to as "gearwalk" or strut chatter". Strut chatter is detrimental to the operation of the antiskid system and, if the amplitude becomes high enough, can by physically destructive to the airplane.

The high frequency vibration is termed squeal and involves the brake and associated structure. Squeal is generally annoying to the passengers and disruptive to brake operation.

The airplane self-induced brake vibration has ingredients discussed in the literature, but is not well understood. The literature does not contain much that relates directly to airplane brake squeal.

Figure 8 shows a typical recording of brake squeal. The analysis of these flight test records shows that vibration occurred in two modes, torsionally about the axle and linearly along the axle. The oscillation occurred at wheel speeds as high as 1000 rpm and as low as 80 rpm.

THEORETICAL STUDIES

A 727 brake was analyzed to gain further understanding of the dynamic characteristics of the brake. The analysis showed the presence of both torsional and axial vibration modes. The degrees of freedom in these directions were considered to be as follows:

1. TORSIONAL SYSTEM

Torsionally the 727-100 brake is a nine degree of freedom system. The inertia of the system are: the piston housing, the pressure plate, the six stators, and the backing plate. An undamped free vibration frequency analysis of the torsional mode was made to determine the natural frequencies and mode shapes of the torsional system. The mode shapes relate the position of the inertias for any instant of time. The lowest calculated frequency was 206 cps. This corresponds to an observed frequency of 190 cps. The vibration at this frequency consists of the entire brake oscillating as a unit on the stiffness of the axle. The calculated frequency of the second mode was 1620 cps.

In general, the vibration can be a combination of the nine modes. The flight test data generally limits observation to the two lowest frequencies. The lowest frequency has been predominent in flight test records.

2. AXIAL SYSTEM

The flight test records have shown vibration in the axial direction i.e., direction parallel to the axle. The axial vibration occurred at the same time and the same frequency as the torsional vibration. Since the instability occurs in the torsional system, the axial vibration must be dr ven by the torsional system.

The 727-100 axial system is essentially a two degree of freedom system consisting of the piston housing and the heat sink vibrating on the stiffness of the brake mounting flange and the torque tube hydraulic system, respectively. The natural frequencies of this system are 271 cps and 1765 cps. The shaker tests gave frequencies of 300 and 1500 cps.

A more exhaustive treatment must consider the presence of non-linear damping due to the sliding action of the rotor and stator keys in the keyways of the torque tube and the wheel. The dry friction between the sliding surfaces produces a force opposing the motion.

COUPLING

Since the primary source of the vibration occurs in the torsional mode, a transferred mechanism between torsional and axial systems is needed. Basically, two types of coupling are encountered. These are:

a. elastic coupling
b. inertial coupling

Elastic coupling is coupling in which the displacement of one mass causes a corresponding force on a second mass. Inertial coupling results from the interasction of the inertial forces of the masses.

Several theories have been put forth. Some of those which have been disproved are:

a. Torque tube shortening
b. Axle shortening
c. Axle Bending
e. Shear deflection of the lining

LANDING GEAR TRUCK PITCH

Bogie type landing gear are frequently incorporated on large commercial transport airplanes. These bogie or truck type gears have now been used on various airplanes for many years. The early landing gears were not fully equalized.* This resulted in severe truck angular movement in a plane parallel to the airplane fore and aft axis. This motion is often termed truck hop or truck pitching. In a fully equalized landing

*Essentially in an equalized truck the brake drag and the load in the compensating link (equalizing rod) produce no moment about the bogie and thus no load is transferred between truck (bogie) wheels due to brake drag.

gear truck, the braking torque is transmitted directly to the strut through the equalizing rods. Without these equalizing rods, the braking torque is applied directly to the truck beam.

Figure 9 shows the pitching moments along with the free body diagrams for nonequalized, front brake equalized, rear brake equalized and fully equalized trucks. The figure shows clearly that partial equalization does not eliminate the tendency to pitch the truck forward but does reduce it. Full equalization practically eliminates the tendency to pitch the truck forward. Experience has shown that a fully equalized truck is very stable and is the only type used on modern airplanes. A certain amount of truck pitching will always exist due primarily to the excitation of runway roughness.

MAJOR EFFECTS OF TRUCK PITCHING ON LANDING GEAR SYSTEMS

The high cyclic loading or the truck assembly resulting from pitching can result in component failures and costly unscheduled maintenance. The presence of violent truck pitching jeopardizes the brake system performance. During modulated braking, the antiskid wheel speed sensors monitor all braked wheel speeds. Whenever the system detects a braked wheel slowing below its synchronous speed, in a predetermined manner, it reduces the brake pressure on this wheel, permitting the wheel to speed up. A pitching truck introduces wheel speed perturbations. These perturbations can "fool" the antiskid into responding as though the wheel is going into a true skid. The unnecessary brake releasing that follows results in an increase of airplane stopping distance.

TRUCK PITCHING PROBLEM

Several things can be done in the initial design to avoid a truck pitching problem entirely or to reduce it to a tolerable oscillation level. The following are a few of the recommendations to achieve this:

- o If possible, design the landing gear truck having its mass centered on the truck pivot.
- o If this is not practical, an attempt should be made to separate the natural frequencies of vertical translation and pitching of the truck.
- o The design of the brake control should incorporate intelligence to prevent pressure modulations at the truck pitching frequency.

ADAPTIVE LANDING GEARS

The primary function of the landing gear oil/pneumatic shock-absorber is the efficient absorption of the airplane vertical kinetic energy during touchdown and the secondary function is to provide a suspension system for taxi. The oleo metering pin was introduced as a means to achieve a more optimum load stroke relationship. Present day digital simulation techniques have reduced touchdown design loads to a minimum. However, with the exception of the dual stage air spring, taxi performance has in the past been more or less ignored since it was generally not a problem. Little or no wing and body structure was being designed by runway roughness induced taxi loads and ride and ground performance was acceptable. Today the aircraft industry is presented with numerous taxi related design problems and the cause can be traced to the slowly changing aircraft designs.

There are basically four areas which have contributed to the slow degradation of taxi performance:

1) Gross Weight Typical airplane gross weight has increased 5 to 10 times since World War II which has aggravated suspension and flotation problems.

2) Airframe Flexibility Airframe structure has become considerably more flexible over the past few decades. This is especially true in the case of large supersonic aircraft. A more flexible structure adversely contributes to taxi dynamic response causing larger normal accelerations, strut loads, etc.

3) Ground Roll Speeds Taxi dynamic response is approximately proportional to ground roll speed. The trend in aircraft design is to higher speeds. Again this is particularly true for supersonic aircraft.

4) Operation on Substandard Fields Recent AF military operations have imposed the requirement that transports and ground support fighters operate from semi-prepared runways in forward areas. Under these conditions the dynamic performance of the landing gear during takeoff and landing roll is no longer of secondary importance.

The core of this discussion is that aircraft designs are dynamic; they are changing. Airplanes are becoming larger, heavier, and more flexible. However, landing gear designs have been static; they have not been advancing technologically to reduce, or eliminate penalties associated with taxi performance.

Taxi related penalties are a problem today and they will be worse 5 years from now and may be intolerable in 10 years. Landing gear technology must advance hand in nand with airframe technology.

Taxi incurred penalties impact airplane overall performance in four areas:

1) Survivability Survivability for military airplanes designed for front line field operation is of major concern. The ability to taxi on rough surfaces and over obstacles increases the airplane mission effectiveness and survivability. This permits operation into a significantly greater number of airfields which are presently inaccessible.

2) Design Loads Taxi induced loads can size wing and body structure as well as design the gear itself. Weight is directly related to such loads and therefore payload is reduced.

3) Fatigue Life Taxi loads reduce fatigue life by contributing to the ground air ground cycle. If taxi loads and/or mission profile are not properly assessed the useful life of an airplane can be seriously affected.

4) Ground Performance Landing gear design dictates ground handling performance and is coupled to the effectiveness of the braking system, pilot controllability, stability, and ride comfort.

Military airplanes are particularly susceptible to penalties incurred through severe dynamic taxi response. The C-130 is a good example of an airplane which has experienced taxi incurred damage. This airplane was used extensively in Southeast Asia for forward line support. The missions were numerous, short in length, and field-surface conditions were poor. This activity adversely affected the fatigue life of the airplane and resulted in early crack propagation in the wing structure.

Boeing has been actively involved in the development of advanced landing gear systems and as a result has studied a number of concepts. Dual mode (adaptive) landing gear concepts are particularly attractive because of their inherent simplicity and potential reliability.

As previously mentioned, the landing gear performs two functions which are: 1) to efficiently absorb the airplane kinetic energy during touchdown, and 2) to act as a suspension system during taxi.

Unfortunately, a conventional passive gear can only be optimized for maximum dynamic performance in one mode of operation and that has historically been the touchdown mode.

In response to this mechanical shortcoming the dual mode oleo was conceived.

The principle of the dual mode concept is to permit the landing gear oleo to function in one of two possible passive modes: one optimized for touchdown performance and the other optimized for taxi performance.

A typical operational sequence would occur as follows:

1) During flight the landing gear dual mode mechanism is positioned in the touchdown mode.

2) During touchdown (probably after the first or second rebound) a signal is received from some source (either external to the oleo or internal) causing the mechanism to switch to the taxi mode.

3) The oleo remains in the taxi mode until another signal is received immediately after liftoff. The cycle is then repeated.

Two hardware dual mode concepts developed by Boeing are designed to improve ground roll characteristics for aircraft with damping dependent suspensions and aircraft with stiffness dependent suspensions.

LANDING GEAR TESTS

Many testing tools are used both during and after landing gear design to substantiate strength, life, and performance.

A. Structural Tests

(1) Photostress
The use of photostress has become standard practice to determine stress levels and direction in the transition areas of all complex gear parts. Usually the complete gear is built out of plastic as soon as the drawings are available and checked by photostress, followed by strain gaging. This permits local changes to be made before the tooling is completed for the parts. The photostress process is repeated when the first gear is loaded prior to fatigue testing.

(2) Fatigue Test
Since the landing gear is usually a fail-safe structure, the FAA now requires substantiation of gear life by fatigue testing. All the various gear loading conditions are applied in blocks to simulate those loads expected to be typical for normal service. The gears are completely dismantled and inspected periodically to check for cracks, wear, galling, etc.

(3) Static Test
Static testing to ultimate load is not a requirement but is sometimes used to determine growth potential. Since most parts of the gear are designed by fatigue conditions, it is common to find generous margins of safety under ultimate loads; therefore, static testing probably does not justify the expense involved.

(4) Detail Tests
Detail parts or features are often fatigue tested or static tested separately from the above tests. Gear breakaway or fuse details are often testing in detail, if analysis is in doubt and where precise failure modes are required.

B. System Tests

(1) Drop Test
The drop testing of the gear is required to develop and substantiate the energy absorption characteristics of the shock strut. This test is conducted by the gear vendor under aircraft manufacturer's direction. The number of official drop test conditions can vary from as few as 20 to perhaps 50 or 60. The variables include:

- o Landing weight
- o Sink speed
- o Level attitude
- o Tail down attitude
- o Wheel speeds

Drop testing is accomplished by loading the gear in a jig mounted in a vertical tower. The jig can be loaded with varying weights. The jig is hoisted to appropriate heights, depending on the desired sink speed, and allowed to drop free, the wheels "landing" on a calibrated platform to determine vertical and drag loads. The effect of spinning up the wheels upon landing is simulated by spinning the wheels to the desired landing speed before the drop, whereupon the wheel rotation is abruptly stopped when contact is made with the ground platforms.
The drop test is also used to substantiate the capabiility of the gear to sustain over-load landings without structural failure and to obtain data points for fatigue analysis.

(2) Retraction Test
As rapidly as production parts can be acquired, both the main and nose gear retraction systems are laboratory tested. All production parts involved in the retraction are tested, including lock and door systems. The parts are mounted in a jig fixture so that the entire airplane system can be operated. Parts that are not designed or affected by retraction loads are usually dummies. The gear weights must be simulated, and compressed air cylinders are commonly usesd to simulate aerodynamic forces on the gear and doors. These tests serve three functions:

(a) The rig is used for as long as two or three months to tune and adjust the hydraulic and mechanical systems systems so the gear operates smoothly without large impact forces (thus noise). The manual extension system is proved out, and often changes are made to locking springs, snubbing orifices, etc. to ensure positive gear operation under all conditions. Needless to say, this phase of the testing should be completed as early as possible so that all important design changes can be implemented prior to first flight.

(b) The second use made of the retraction test is to provide service life of moving parts. A minimum life of 25,000 flight cycles has proved to be an adequate demonstration that the wearing parts such as seals, bearings, bushings, locks, etc. will servive one overhaul in airline service. Frequent teardown inspections are conducted in the early cycling for evidence of premature wear. Any part that is badly worn or destroyed at 25,000 cycles should be redesigned.

(3) Beginning with the 727, the retraction testing was continued far beyond the 25,000-cycle point. The continuing phase of testing, to as many as 150,000 cycles in the case of the 737, is required to prove adequate fatigue life of all the structural mechanism. Our 707 and 727 experience has taught us that the failure to test and prove fatigue life of many parts in the retraction mechanisms can have serious safety implications. Strain gaging is employed in many locations in the Phae I tests to provide the Stress Group with improved data for fatigue analysis. However, due to the complex shapes of many mechanism parts, the only final proof of fatigue life is actual cycling. The most important safety aspects are absolute assurance that the gears can be unlocked from UP, reliable sequencing of doors, and 100% reliability of snubbing in the extend cycle. A landing gear extending to DOWN without benefit of hydraulic snubbing will generally destroy the brace mechanism and allow the gear to dangle free. The subsequent landing will then be an

emergency type, with the danger of ground loops, wheels up landing, and possible fire. The extended fatigue test cycling will also prove useful to continue comparison tests of various different bearings and optimize the final production selection before too many airplanes have been produced.

This series of tests will usually continue for a year or more after the first flight.

C. Shimmy Testing
Landing gear shimmy is an unstable condition caused by the coupling of the torsional mode with side bending mode of the gear. The shimmy characteristics of a gear are analyzed by computer simulation as soon as reliable spring rate information is available. The exposure to shimmy is great on all nose gears and main gears having laterally disposed wheels. An additional tool for checking gear shimmy is the flywheel dynamometer. The entire landing gear must be used and mounted by actual or simulated support structure. Combinations of load, yaw, tire unbalance, gear slop, and speed must be run, with an abrupt torional excitation artifically applied by apply one brake or applying an abrupt punch to the end of the axle.

D. Wheel and Brake Testing
The flywheel dynamometer is the basic tool for testing wheels and brakes. The dynamometer consists of a large road wheel that is spun up to the desired speed by an electric motor. The mass of the wheel can be varied to obtain the desired kinetic energy. One wheel, brake, and tire is mounted on an axle, which is attached to a hinged arm. The dynamometeris brought up to speed with the tire off the wheel. When the desired wheel speed is established, the wheel is allowed to coast. The mandrel arm is then "landed" on the road wheel with sufficient force to obtain a predetermined rolling radius and the brake is applied. By this means all necessary data is obtained to assess brake performance. This information includes continuous plots of speed, torque, and distance, and a large number of brake temperature readings.

Wheels are static tested to ultimate and yield load with tires installed, and are usually roll tested several thousand miles to establish fatigue life.

INSTRUMENTATION

As indicated above, a considerable amount of testing is done in the development of the landing gear system. This testing involves a large range of variables and a variety of test facilities. Tables 2 through 6 indicate the variables and the measurement means for some of these tests.

The usefulness of test depend, to a large extent, on the ability to accurately measure the test data and to transform it into meaningful information. In some tests this is a simple task. For example, the test may be simply pass/fail, as in fatigue tests. Most often, detailed performance data is required. Measurement of the data is generally not a problem, but recording and reduction of the data can be difficult.

Three systems are commonly used for data recording:

1. Light beam oscillograph
2. Multiplexed tape
3. Wide bandwidth FM tape

The lightbeam oscillograph provides data with good bandwidth and allows quick determination of data trends. But detailed reduction must be done by hand. Multiplexed tape provides a recording medium for a large number of variables in a form easily processed by digital computers. Because it is a data sampling system, the available bandwidth is limited by the number of recorded variables. FM tape provides the widest bandwidth but only for a very limited number of variables. Each system has its place, and successful testing requires proper selection of the recording method.

A LOOK INTO THE FUTURE

The future of landing gear systems hold the promise of considerable advancement. The availability of digital electronic systems will provide increased sophistication in the antiskid system. In addition, systems which optimize both braking and directional control are in development. Active shimmy damping is a possibility.

Increased emphasis on ride quality will result in gears which produce lower fatigue loading. This will provide the military with improved surviablity and increased capacity to operate from unprepared and semi-prepared fields.

Carbon composite brakes, already in use in military aircraft, will become available on commercial transports. Tire design will improve both through construction technique, eg, radial cord, and advanced materials.

Inflation/deflation systems will also receive serious consideration for austere field operation to enhance aircraft flotation. Training simulators will be much more representative of ground roll, landing and take-off operation due to added landing gear models.

ACKNOWLEDGEMENTS

The authors wish to thank many collegues who have contributed to the development of landing gear systems technology. In particular we wish to express our appreciation to W. G. Nelson who supervised some of these efforts and B. C. Hainline who provided much needed encouragement and funds. In addition, contributions of B. Urbonis, J. Kilner and H. Domandl are gratefully acknowledged.

CATEGORIES AND FUNCTIONS \ COMPONENTS	TIRE	WHEEL	BRAKE	BRAKE CONTROL	SHOCK ABSORBER	LANDING GEAR STRUCTURE	STEERING	RUNWAY	ARRESTOR AND HOOK	AUXILIARY DEVICES	ENVIRONMENT	ACTUATOR	ENGINE	AGE
LANDING IMPACT AND SUPPORT														
1. LANDING IMPACT	●	●	●		●	●		●		●	●			
2. STATIC SUPPORT	●	●			●	●								●
DIRECTIONAL CONTROL														
3. STEERING	●	●	●	●	●	●	●	●		●	●	●		
4. TAXI	●	●	●		●	●		●			●		●	
5. TAKEOFF	●	●			●	●		●		●	●		●	
6. DAMPING	●		●	●	●	●	●		●			●		
VELOCITY CONTROL														
7. RETARDATION	●	●	●	●		●		●	●	●	●		●	
8. ARRESTMENT	●					●		●	●					
GEAR POSITIONING														
9. RETRACTION					●	●	●				●	●		
10. EXTENSION					●	●	●				●	●		
11. KNEELING					●	●					●	●		●
12. CROSSWIND					●	●					●	●		
FLOTATION														
13. FLOTATION	●	●						●			●			
14. SKIING	●	●			●	●		●		●	●	●		
GROUND MAINTENANCE														
15. TOWING	●	●				●		●			●			●
16. TIE-DOWN	●	●				●		●	●		●			●
17. JACKING					●	●		●			●			●
18. SERVICING	●	●	●		●							●		●

Figure 1. Landing Gear System Matrix

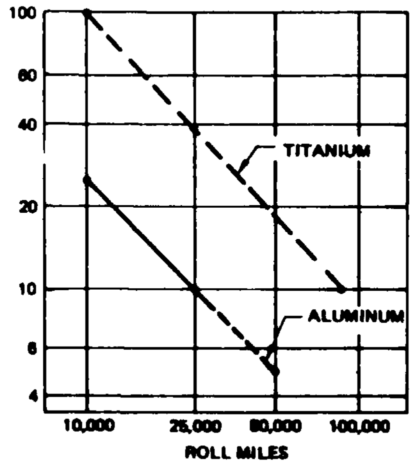


Figure 2. Cost Factor Vs. Roll Miles

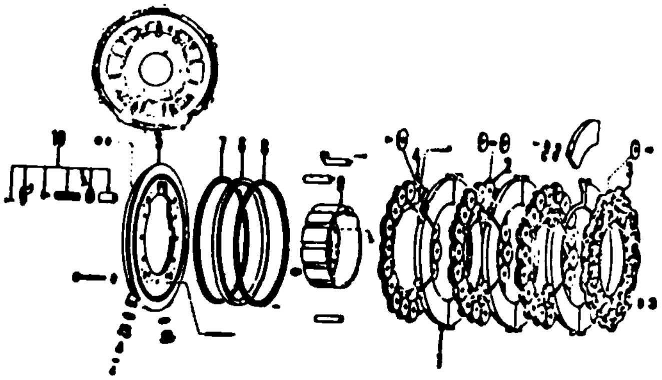


Figure 3a. Exploded View of Fighter Brake Assembly

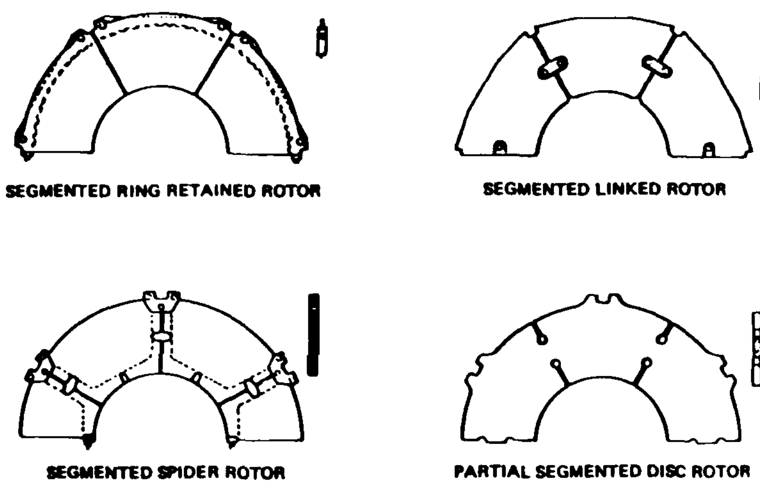


Figure 3b. Various Types of Rotor Designs Used in Aircraft Brakes

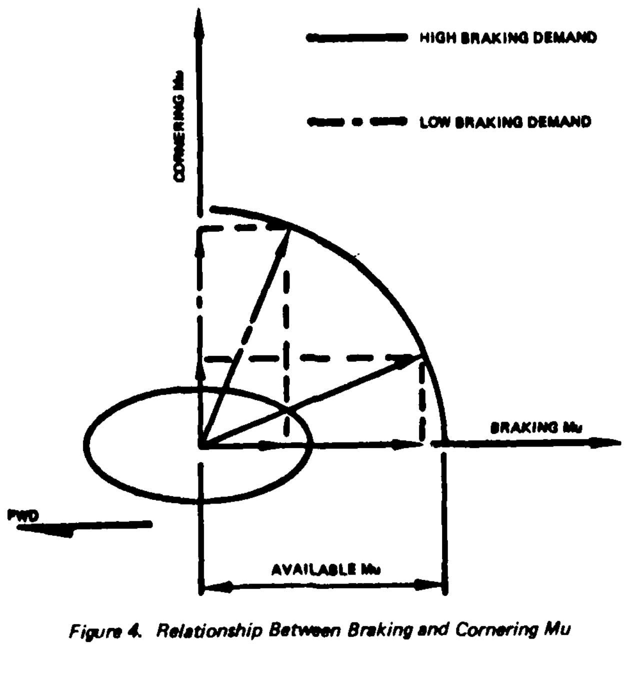


Figure 4. Relationship Between Braking and Cornering Mu

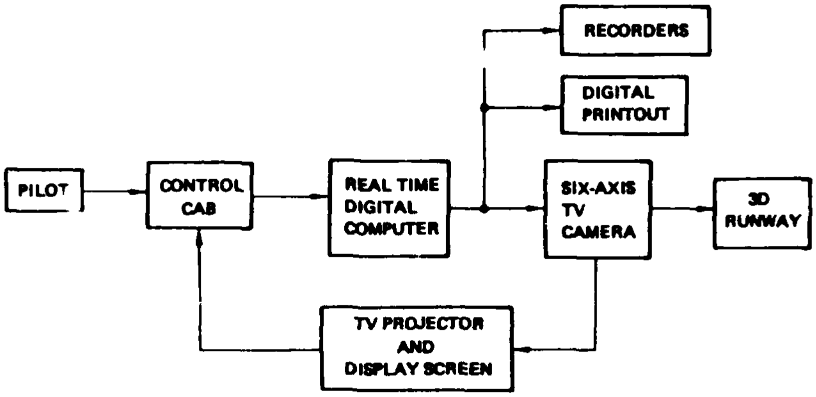


Figure 5. Ground Handling Simulation Schematic

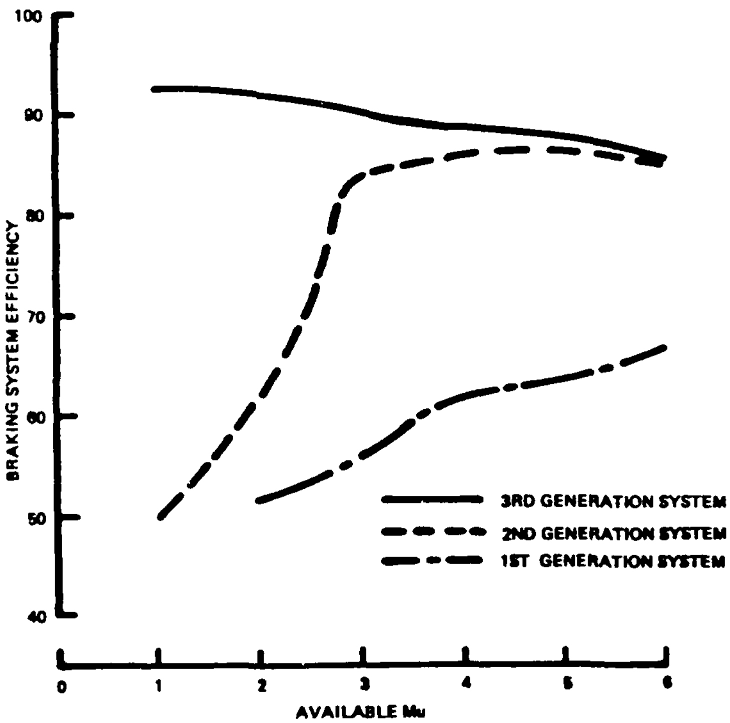


Figure 6. Antiskid System Efficiency Improvements

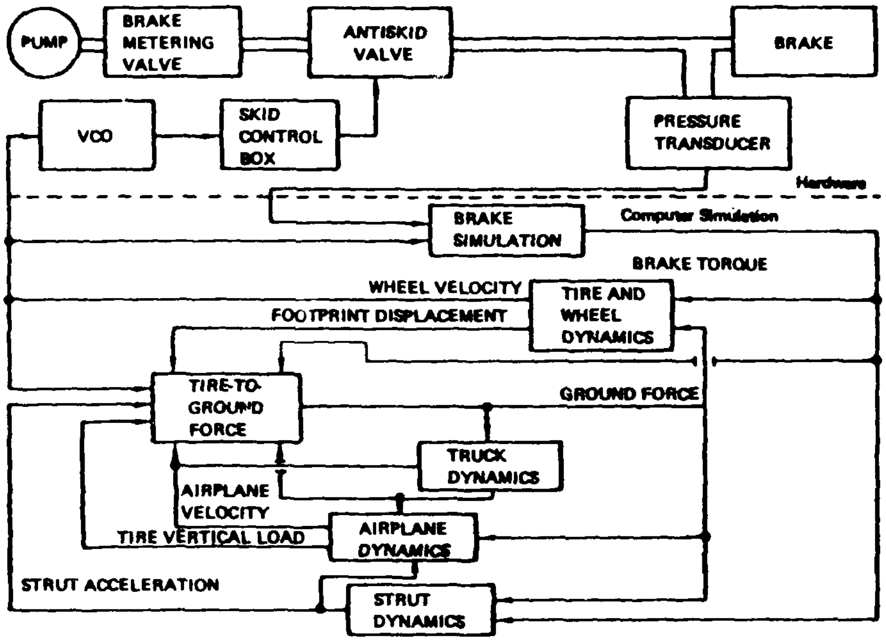


Figure 7. Simulator Block Diagram

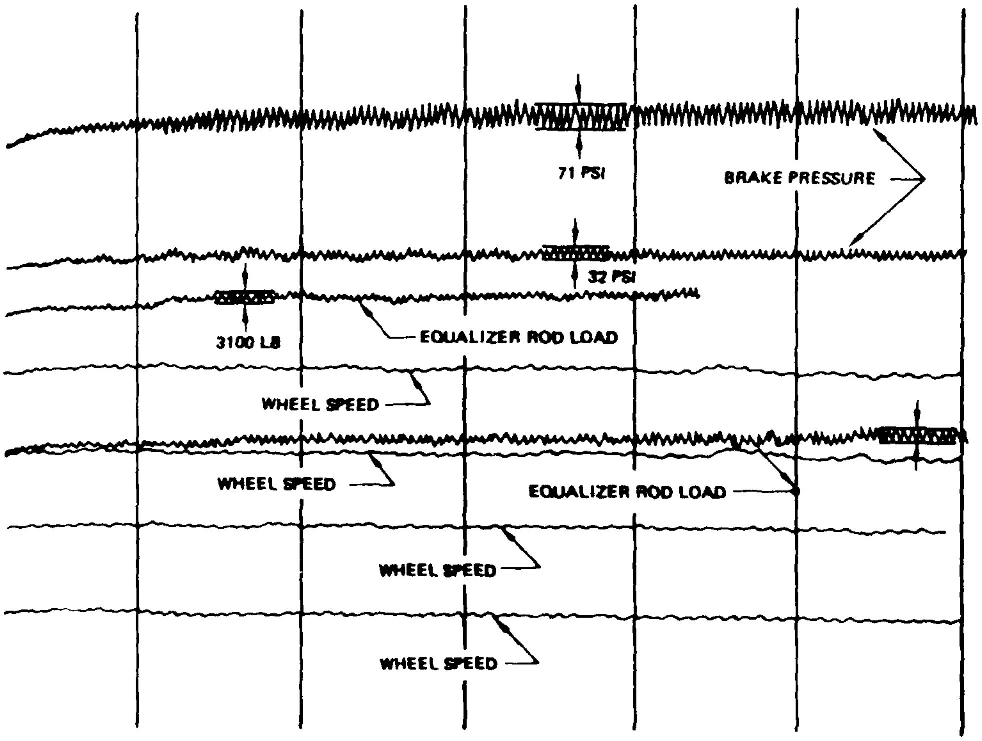
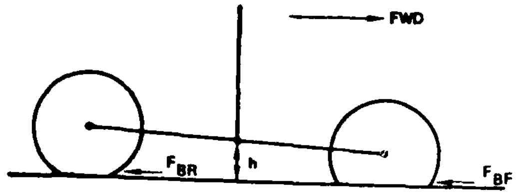
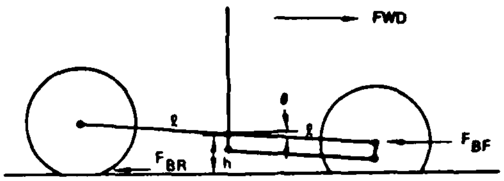


Figure 8. A Typical Flight Test Record Showing Brake Squeal (KC-135)



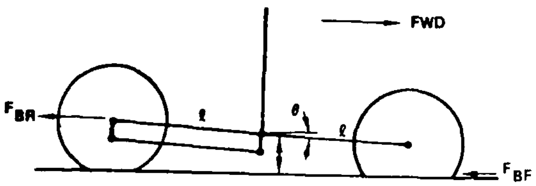
PITCHING MOMENT = $h(F_{BF} + F_{BR})$

Figure 9a. Illustration of a Non-Equalized Truck



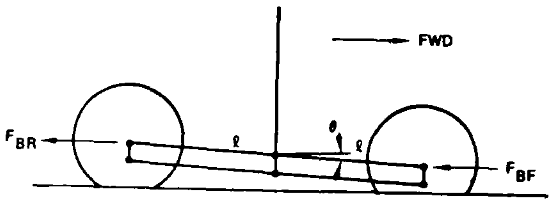
PITCHING MOMENT = $hF_{BR} + F_{BF}\ell\sin\theta$

Figure 9b. Front Brake Equalized Truck



PITCHING MOMENT = $hF_{BF} - F_{BR}\ell\sin\theta$

Figure 9c. Rear Brake Equalized Truck



PITCHING MOMENT = $(F_{BF} - F_{BR})\ell\sin\theta$

Figure 9d. Fully Equalized Truck

Table 1. Tire Properties and Related Terms

Foot Print Area	Pneumatic Caster
Braking Force Coefficient	Rolling Radius
Cornering Force Coefficient	Slip Angle
Vertical Force Coefficient	Yaw Angle
Yawed Rolling Coefficient	Wheel Angular Velocity
Drag (Fore and Aft) Force	Lateral Hysteresis
Lateral or Cornering Force	Torsinal Hysteresis
Fore and Aft Spring Force	Vertical Tire Deflection
Lateral Spring Force	Polar Moment of Inertia
Circumferential Decay Length	About Axle
Fore and Aft Spring Constant	(Tire, Tire and Wheel)
Torsional Spring Constant	Pneumatic Trail
Lateral Spring Constant	
Cornering Power	
Inflation Pressure	

Table 2. Typical Brake Data from Dynamometer Tests*

VARIABLE	HOW MEASURED	STATUS		OTHER COMMENTS
		ACCEPTABLE	NEEDS IMPROVEMENT	
Torque	Strain Gage or Pressure Gage	X	X	Pressure gage data not suitable for antiskid improvement
Speed	D. C. Tachometer	X		
Load	Load Cell	X		
Temperature	Thermocouples	X		
Accelerations	Accelerometers	X		

*Simulate aircraft K.E. and load tire so no skidding occurs

Table 3. Typical Anti-Skid Data - Simulation and Flight Testing

VARIABLE	HOW MEASURED	STATUS		OTHER COMMENTS
		ACCEPTABLE	NEEDS IMPROVEMENT	
Wheel Velocity	Antiskid Output	X		
Valve Current	Antiskid Output	X		
A/R/Ground Switch	Antiskid Output	X		
Brake Pressure	Pressure Transducers	X		
Metering Valve Pressure	Pressure Transducers	X		
Return Line Pressure	Pressure Transducers	X		
Drag Strut Load	Strain Gages	X		These variables are directly available from the simulation during lab testing.
Brake Torque	Strain Gages	X		
Spoiler Position	Potentiometer	X		
Throttle Position	Switch	X		
Airplane Position	APACS Camera	X		
Airplane Velocity	APACS Camera	X		
Airplane Acceleration	APACS Camera	X		

Table 4. Typical Tire Data - From Dynamometer Test (TSO Reqm't)*

VARIABLE	HOW MEASURED	STATUS		OTHER COMMENTS
		ACCEPTABLE	NEEDS IMPROVEMENT	
Load	Prescheduled Load Cycle is Applied	X		
Speed	Speeds are prescheduled to represent duty cycle	X		
Time		X		
Tire Yaw Angle		X		Recent USAF Reqt
Tire Deflection	Pre-selected			Typical 33-35%

Other factors are burst pressure, tread design, slippage, tire airworthiness

*Simulated to reproduce A/C K.E.

Table 5. Typical Data - From Landing Gear Drop Tests

VARIABLE	HOW MEASURED	STATUS		OTHER COMMENTS
		ACCEPTABLE	NEEDS IMPROVEMENT	
Vertical Load	Load Cell Under Tire	X		
Drag Load	Load Cell Under Tire	X		
Axle Bending Load	Strain Gage	X		
Oleo Displacement	Potentiometer	X		
Upper/Lower Chamber Pressure	Pressure Transducers	X		
Tire Deflection	Photography	X		
Gas/Cylinder Wall Temp	Thermocouple	X		
Carriage Drop Height	Preset	X		

Table 6. Typical Data Obtained During Shimmy Tests

VARIABLE	HOW MEASURED	STATUS		OTHER COMMENTS
		ACCEPTABLE	NEEDS IMPROVEMENT	
Load-Drag/Side	Strain Gage	X		Use is made of all instrumentation available for antiskid and taxi tests
Pressure	Strain Gage Pressure Transducer	X		
Acceleration	Accelerometers	X		
Nose Gear Angle		X		

RECHERCHE DES LIMITES DE FONCTIONNEMENT D'UN ROTOR D'HELICOPTERE EN VITESSE ET FACTEUR DE CHARGE

par
B. Certain et J.M. Besse
BP 13, 13772 Marignane
France

1. INTRODUCTION -

Le rotor principal de l'hélicoptère a toujours été l'origine de la plupart des limitations du domaine de vol.

Sur les machines modernes, les charges transmises au rotor se sont encore accrues pour plusieurs raisons :

- Puissance installée importante due à la bimotorisation et au désir de conserver des performances correctes au décollage après panne d'un moteur et par temps chaud ou en altitude.
- Vitesses de translation élevées, conséquence bien sûr de la motorisation mais aussi d'un affinement important des fuselages, surtout par l'adoption de trains d'atterrissage rentrants et de capotages de mât.
- Charge au disque élevée, permise par la surmotorisation.

De plus, d'importantes évolutions technologiques permettent aux ensembles mât, moyeu, pales, de supporter des taux de contraintes élevés : Il s'agit bien sûr de l'utilisation des fibres, verre et carbone, des articulations lamifiées ou encore des amortisseurs viscoélastiques.

Cette recherche quasi systématique d'un fonctionnement aux limites aérodynamiques des profils pose des problèmes nouveaux d'ordre théorique (aérodynamique instationnaire) mais aussi de technique d'essais, les réponses des profils modernes étant peu connues en regard de l'expérience acquise sur le bon vieux NACA 0012 utilisé exclusivement sur nos hélicoptères jusqu'à un passé récent.

A partir de l'exemple du SA.365N dit DAUPHIN "COAST GUARD" (bimoteur 2 x 440 kW) qui, à la masse maximum de 3850 Kg, réalise en palier plus de 160 kt au niveau mer, on va rappeler brièvement les origines des "murs" qui bordent le domaine de vol de l'hélicoptère, puis on décrira les moyens de mesure, de surveillance en vol et de dépouillement. Enfin, la méthode d'essai sera décrite et montrée par un petit film pris au cours des phases de vol aux limites du domaine.

2. PHENOMENES PHYSIQUES CONDUISANT A DES LIMITES -

2.1. - Le décrochage -

C'est le problème classique de la pale reculante qui doit garder une certaine portance alors que la vitesse relative de l'air devient faible et même s'annule vers le centre (cercle d'inversion).

L'incidence sur les profils augmente et l'on note l'apparition de contraintes élevées sur les bielles de pas.

La planche n° 1 montre l'évolution de ce signal en fonction du facteur de charge sur la première génération de pales du 365N.

On remarque que la divergence devient visible en pale avant, s'installe dans la plage jaune et commence à décroître au milieu de la plage bleue, c'est à dire 3/4 arrière.

Au signal qui pour n = 1 g est du 1ω presque pur, se superpose une forte modulation en 6ω qui correspond à une réponse du mode de torsion de la pale.

La planche n° 2 donne une explication schématique de ce phénomène connu sous le nom de "STALL FLUTTER".

Les courbes noires représentent la polaire d'un profil obtenue en soufflerie ($Cz = f(\alpha)$ - $Cm = f(\alpha)$).

Les courbes en couleur sont les polaires instationnaires, c'est à dire les relations reliant la portance et le moment à l'incidence alors que le profil oscille en incidence autour d'une valeur moyenne de $\pm$ 5°.

Sans entrer dans les détails, on peut dire que l'incidence de décrochage n'est plus fixe mais dépend de la vitesse de sa variation, le décrochage étant repoussé lorsque l'incidence augmente et au contraire persiste d'autant plus que l'incidence décroît vite.

Le fonctionnement hélicoptère correspond aux deux premiers cas ($\alpha = 7 + 5 \sin \omega t$ et $= 15 + 5 \sin \omega t$).

Dans ce second cas, on note une évolution curieuse du $Cm = f(\alpha)$ en forme de 8 et l'on démontre que dans la partie hachurée, l'amortissement est négatif, le profil absorbe de l'énergie venant de l'air. On constate par ailleurs que ce phénomène se produit pour les faibles incidences du cycle et on retrouve le fait constaté sur le signal biellette de la planche n° 1, que l'on voit diverger à partir de la pale avançante alors que l'incidence est encore faible et converger dans une zone où elle est encore forte. /..........

Ce phénomène n'est jamais divergent, l'amortissement sur un cycle étant positif mais il peut engendrer des contraintes très importantes par excitation du mode de torsion.

Une solution a été trouvée à ce problème en centrant l'extrêmité de pale plus avant. L'augmentation de vrillage en pale reculante due à cet accroissement de rappel à plat diminue notablement l'incidence en bout de pale.

2.2. - Le Mach limite -

En pale avançante, le Mach atteint par l'extrêmité peut être grand et l'on se heurte aux problèmes de transsonique, c'est à dire augmentation rapide de la traînée et apparition de moments dus au recul du foyer des profils.

La planche n° 3 montre la complexité du phénomène sur quatre paramètres :

(1) Contrainte de l'amortisseur de traînée viscoélastique,
(2) Contrainte de traînée emplanture pale,
(3) Battement emplanture pale,
(4) Biellette de pas.

Au cours de cette phase de vol, le Mach est proche de 0,97 réalisé avec des pales à profil OA 209.

On constate sur la traînée une forte modulation en $\omega/2$ et une forte variation du signal biellette à la limite des zones de couleurs, juste en position pale avançante. Ce dernier signal est aussi modulé en $\omega/2$ et le phénomène n'apparaît visiblement qu'un tour sur deux.

Ceci a pour conséquence un déréglage en track du rotor, chaque pale ayant deux trajectoires possibles qu'elle décrit en alternance.

Vu de la cabine, le rotor semble faire un peu ce qu'il veut. Là encore les contraintes peuvent devenir très importantes.

Pour reculer l'apparition de ce phénomène, un effort tout particulier a été fait sur les profils et la planche n° 4 donne quelques résultats comparatifs en prenant comme référence le NACA 0012.

On caractérise un profil par la valeur de Mach pour laquelle le gradient de traînée dCx/dM atteint 0,1.

La partie gauche de la planche n° 4 donne le Mach critique à portance nulle. On voit que le NACA 0012 atteint 0,8, que les profils type SA.131 (PUMA) permettent d'aller jusqu'à 0,85 et la série des OA2 jusqu'à 0,87.

On notera l'importance de l'épaisseur relative.

La partie droite donne les mêmes caractéristiques mais à portance non nulle. Les profils modernes gardent un avantage jusqu'à 0,3 de Cz environ.

Bien que très significatives(et sur le plan performances aussi), ces améliorations de profils ne suffisent pas à couvrir le domaine revendiqué aujourd'hui par les hélicoptères les plus modernes. Il a fallu travailler l'extrêmité de pale, avec des saumons en flèche, dont l'effet n'est pas celui de la flèche d'une aile d'avion, mais sert à créer un centre de poussée en arrière de l'axe de torsion de la pale, ce qui la stabilise et diminue les variations de vrillage dues aux moments aérodynamiques.

3. MOYENS D'ESSAIS -

De ce rapide exposé, il ressort que la surveillance de certains efforts est nécessaire pendant le vol afin d'alerter l'équipage de l'approche de phénomènes dont le déclenchement est assez brutal, même s'ils ne sont pas explosifs.

Par ailleurs, d'une manière générale, ces augmentations de niveau de contraintes ne sont sensibles à bord par des vibrations que lorsque les phénomènes sont déjà profondément installés, l'aide de la télémesure est donc impérative.

En dehors de cette fonction de surveillance, l'installation d'essai permet d'enregistrer à bord un grand nombre de paramètres vibrations et contraintes ainsi que tous les paramètres de vol à évolution lente traités en numérique (PCM).

Au sol, pendant les vols et après ceux-ci, cette masse d'information doit être traitée et analysée.

Les planches n° 5 à 16 vont nous aider à décrire une telle installation en prenant toujours le SA.365N comme exemple.

3.1. - Installation embarquée -

Les photos n° 5 et 6 représentent un moyeu STARFLEX équipé de ces jauges et de son collecteur dans lequel il passe 40 voies environ comprenant les contraintes battement et traînée pales, biellette de pas, contraintes moyeu dans les bras, la partie centrale, déplacement adaptateur ....

de fréquence, flexions mât, accéléros 3 axes sur la tête, etc...

Ces paramètres et ceux venant de la structure et du rotor arrière sont amenés à une baie dont la planche n° 7 donne le principe. Les paramètres sont calibrés dans une série d'amplificateurs. Un tableau de mélange permet d'envoyer chaque voie dans l'un des 12 canaux des 8 multiplex. En effet le choix de la position n'est pas arbitraire, tous les canaux n'ayant pas la même bande passante.

(La bande passante de chaque voie est proportionnelle à la fréquence de la sous porteuse correspondante. Les sous porteuses sont échelonnées afin de séparer l'information au moment de la combinaison des voies et de permettre sa reconstitution par filtrage lors de la lecture de la bande).

Ensuite, les signaux, après mélange, sont enregistrés sur magnétique avec une base de temps et les conversations équipage et télémesure.

Une partie de l'information peut être visualisée à bord sur des scopes (cas des gros hélicoptères), mais surtout 12 voies sont envoyées en télémesure par un émetteur (20 W bande P).

Les photos n° 8 et 9 représentent cette installation à bord du 365N. On notera à gauche l'enregistreur magnétique, à droite les amplificateurs et en-dessous les platines des multiplex.

Le poids total embarqué, compte-tenu des fils et de la partie PCM, est de 250 Kg. environ.

3.2. - Au sol -

Télémesure -

La planche n° 10 décrit cette installation : L'opérateur est relié par VHF à l'hélicoptère en essais. Il dispose essentiellement de deux scopes sur lesquels sont envoyés les paramètres à surveiller et leurs limitations (il reçoit aussi, non représenté ici, le message PCM complet qui lui permet d'avoir en lecture directe position des commandes, vitesse, alt...etc).

Ici, les moyens de calcul sont limités à des analyseurs analogiques capables de donner les spectres des signaux reçus. En effet, compte-tenu du temps de réaction très court dont dispose l'opérateur, ce sont essentiellement les scopes qui sont utilisés.

Les photos n° 11 et 12 montrent cette pièce avec ces scopes, son récepteur et son enregistreur magnétique.

Dépouillement -

Pendant le vol, mais surtout après le vol, l'information contenue dans la bande magnétique de bord va être exploitée par un ordinateur. La planche n° 13 résume les différentes fonctions réalisées par un MITRA 125.

L'information est constituée par les 12 voies d'un multiplex (qui sont traitées les unes après les autres) d'une part et par un paquet de cartes constituant les étalonnages ou les niveaux de contraintes α et β ou toute autre information nécessaire telle que les temps codés des séquences à dépouiller.

Trois programmes principaux sont en mémoire, les deux premiers font des analyses harmonique et spectrale, le troisième calcule les endommagements des pièces principales au cours de vols types. Les durées de vie seront calculées à partir de ces résultats par le Bureau d'Etudes.

Les sorties possibles sont la bande magnétique, surtout utilisée pour stocker les résultats du troisième programme en vue d'un traitement statistique ultérieur.

Les analyses sont sorties soit sous forme de listing soit sur table traçante soit sur écran Techtronix avec "hard copy".

Les photos n° 14, 15 et 16 donnent une idée de la salle des machines (il y a en fait deux MITRA) et de leurs périphériques.

La tendance est d'éviter au maximum les listings qui sont fastidieux à exploiter et d'obtenir des traces élaborées.

Exploitation -

Elle se ramène toujours à vérifier des niveaux de contraintes afin d'assurer la sécurité des vols à venir d'une part, à chercher d'autre part les causes des réponses importantes, c'est à dire déterminer les zones des spectres qui contiennent le plus d'énergie. Tout le jeu consiste alors à déplacer les modes propres des pales afin de diminuer leur réponse en les éloignant des excitations fondamentales.

La planche n° 17 est une recopie de "hard copy" d'écran Techtronix concernant l'analyse spectrale d'un signal biellette de pas à 13.000 ft pour des vitesses croissantes.

L'analyse de signaux semblables, en battement et en traînée, permet de déterminer les modes de pales.

/...........

La planche n° 18 donne un exemple de résultat obtenu au banc par excitation du rotor par les commandes de vol pour différents régimes rotor. L'intérêt d'une telle méthode est qu'au banc les excitations sont connues en phase et amplitude.

En vol, on commence à arriver à des résultats mais en utilisant des méthodes plus complexes basées sur les corrélations croisées entre deux paramètres, ce qui revient en quelque sorte à supposer que l'un est l'excitation et l'autre la réponse.

Une fois connue,ou supposée comme telle, l'origine d'un problème, il faut encore pouvoir effectuer des changements sensibles sur les pales. La technologie des pales en stratifié nous aide beaucoup car elle permet d'avoir des masses linéiques non constantes par introduction de masses concentrées. Par ailleurs, la raideur en torsion peut être ajustée par variation de l'angle des tissus de recouvrement.

4. METHODE D'ESSAIS -

Le domaine de vol est ouvert progressivement par des piqués effectués à pas constant, à des vitesses croissantes, basse altitude et pour des masses moyennes. Par la suite, la masse est augmentée jusqu'à la valeur maximum désirée puis viennent les vols en altitude.

Pour les facteurs de charge, l'évolution est la même, l'essai se faisant à pas constant et vitesse constante.

La difficulté de ces essais réside dans la complexité de la mise en oeuvre (nombre important de paramètres, fragilité des jauges, étalonnages évolutifs, parasites ...), dans la rigueur des points d'essais surtout en ce qui concerne les facteurs de charge pour lesquels la rapidité de mise en virage (ou de sortie) peut être un paramètre important.

Un suivi précis des configurations machine est indispensable (masse, centrage) ainsi que toutes les modifications, leur importance n'étant pas évidente a priori.

Les équipes de l'AEROSPATIALE sont bien rodées à ces essais qui demandent une confiance sans limite dans une surveillence au sol, d'ailleurs jamais mise en défaut pendant les vols qui ont permis la mise au point et la certification d'un nombre respectable d'hélicoptères au cours des 20 dernières années.

Les fruits de ce travail constant où tous ont participé peuvent se résumer dans les chiffres ci-dessous qui fixent le domaine de vol du 365N ouvert au début de l'été 1980.

- Masse maximale = 3850 Kg.

A cette masse :

- VD basse altitude = 183 kt
- VD à 13.000 ft = 140 kt
- Facteur de charge maxi à 140 kt basse altitude à la puissance maximale continue = 1,8 g.
- Facteur de charge maxi à 100 kt à 13.000 ft à la puissance maximale continue = 1,6 g.

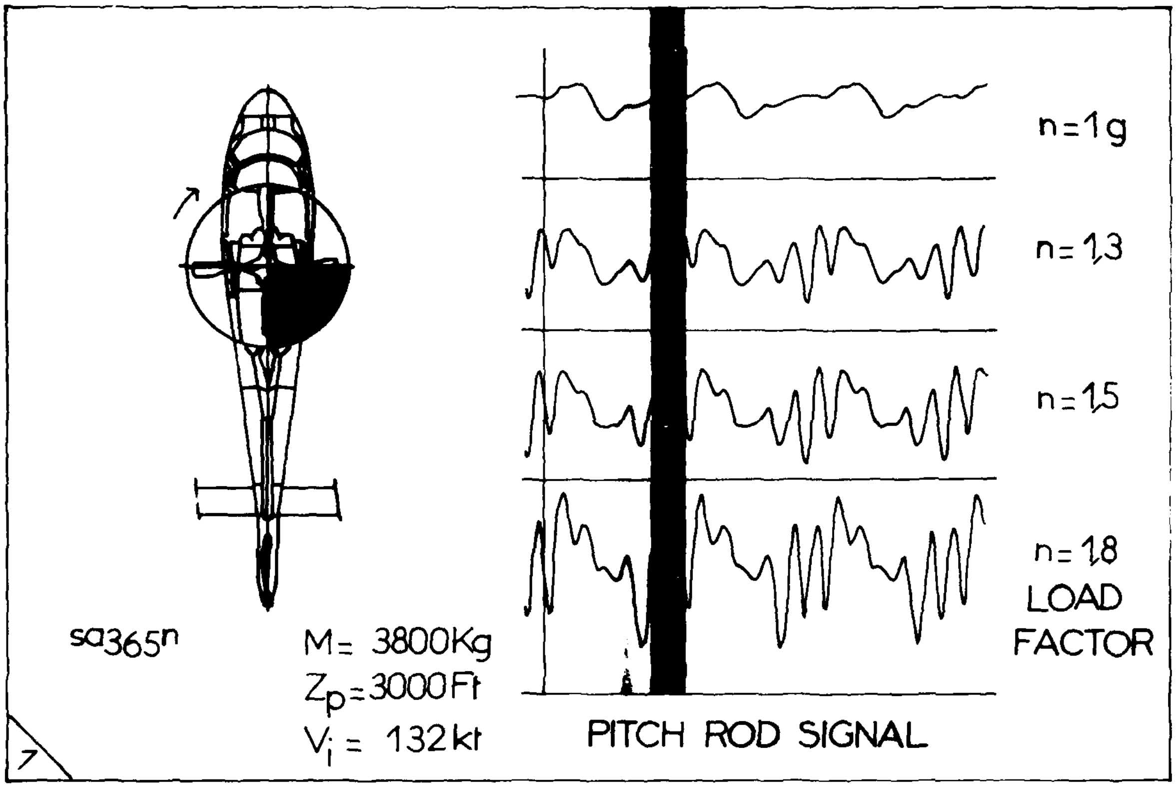


Figure 1

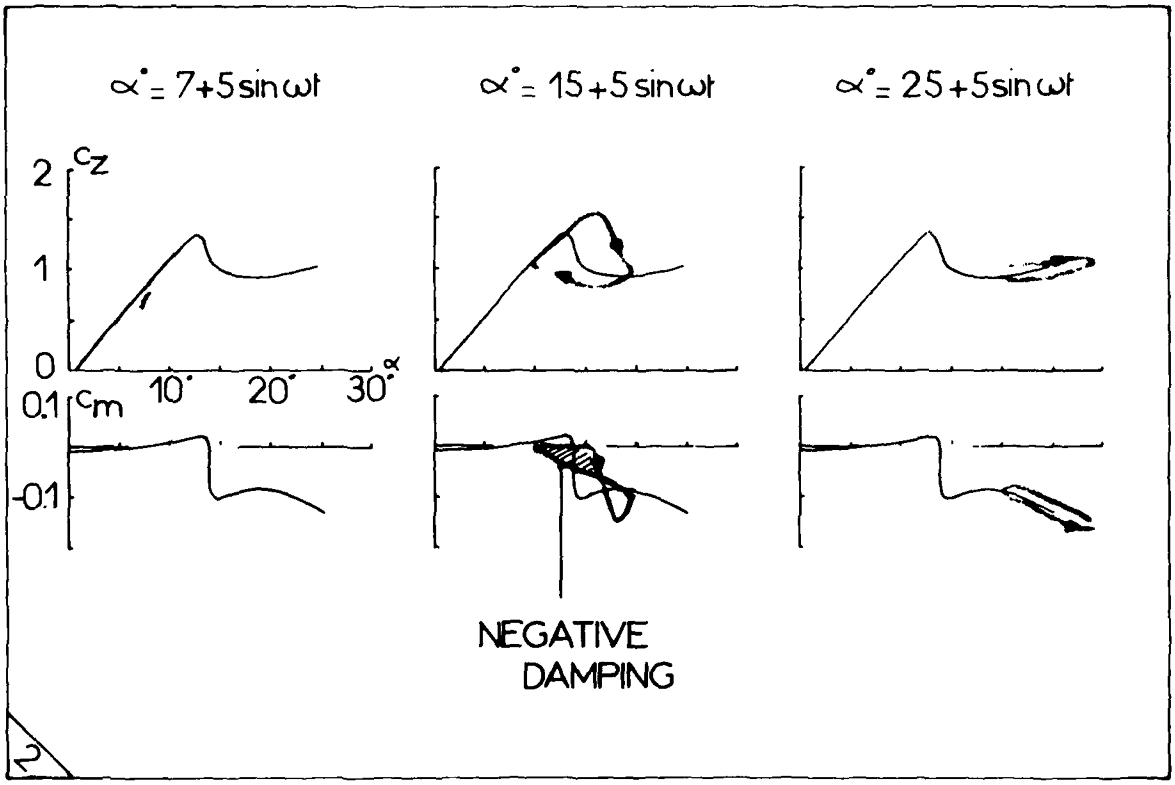


Figure 2

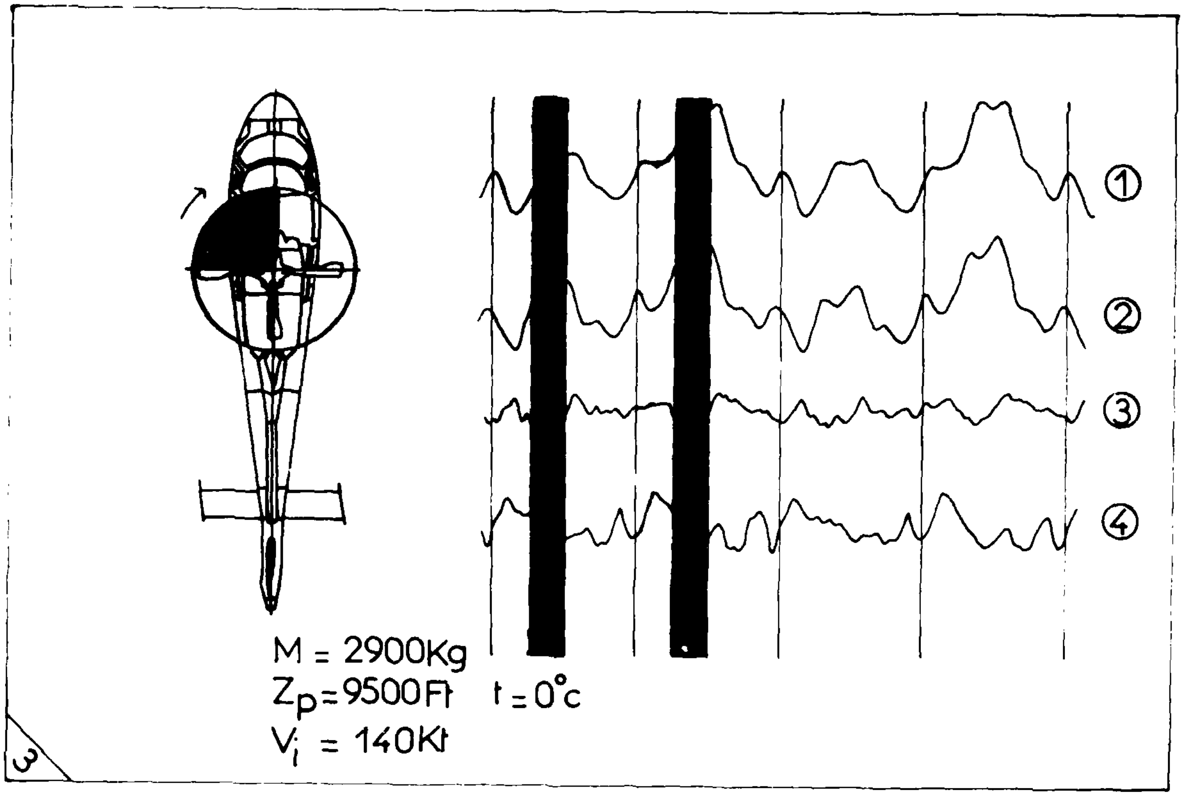


Figure 3

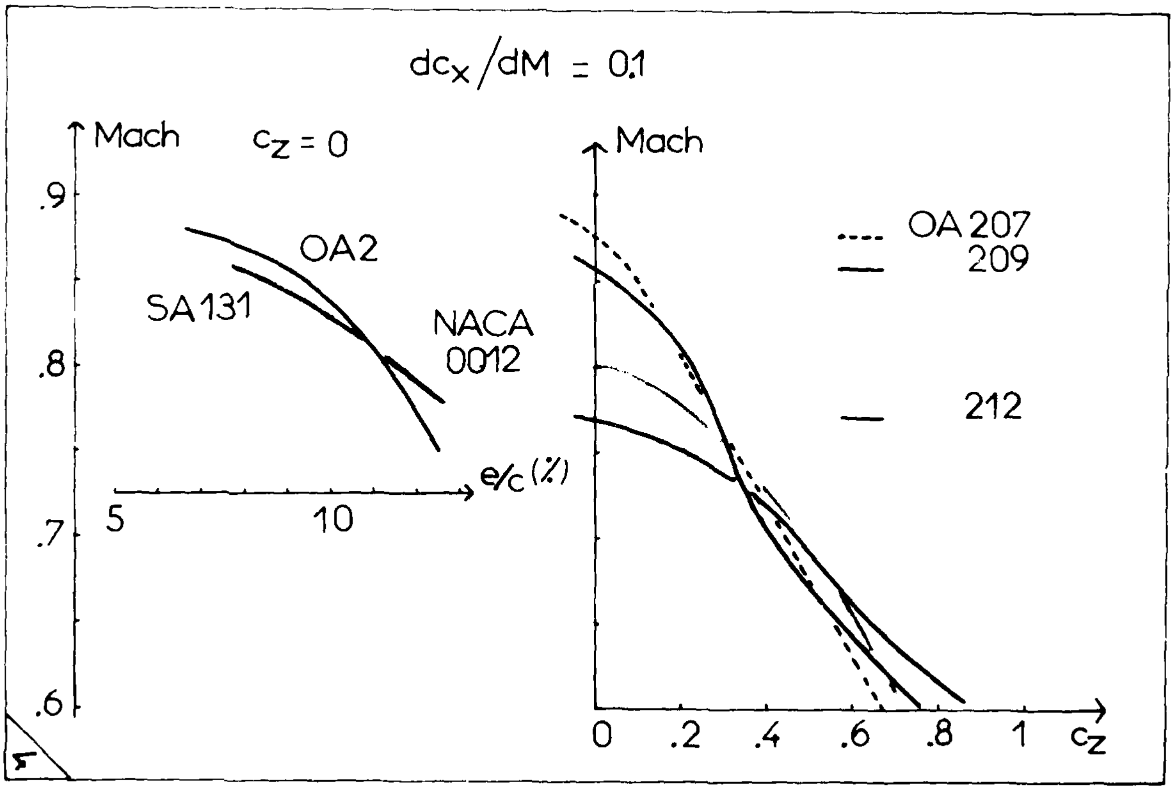


Figure 4

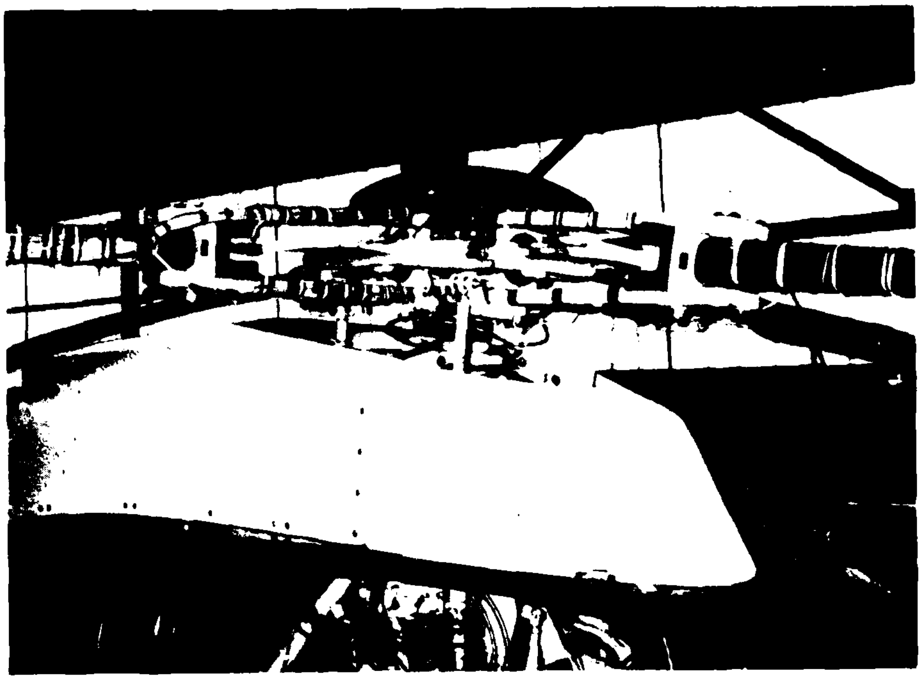


Figure 5 COWLING AND STARFLEX ROTOR HEAD

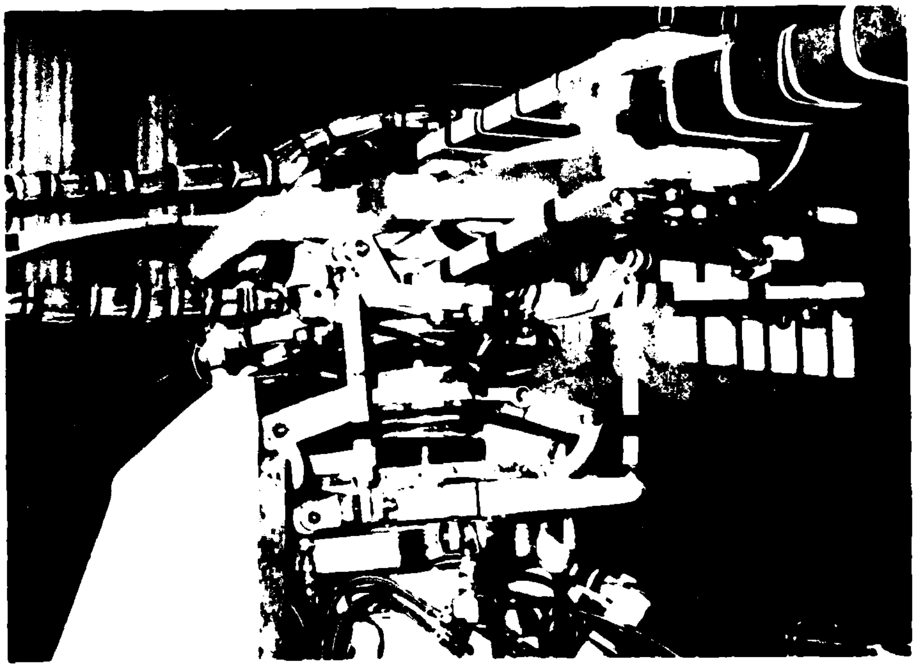


Figure 6 STARFLEX ROTOR HEAD WITH STRAIN GAUGES

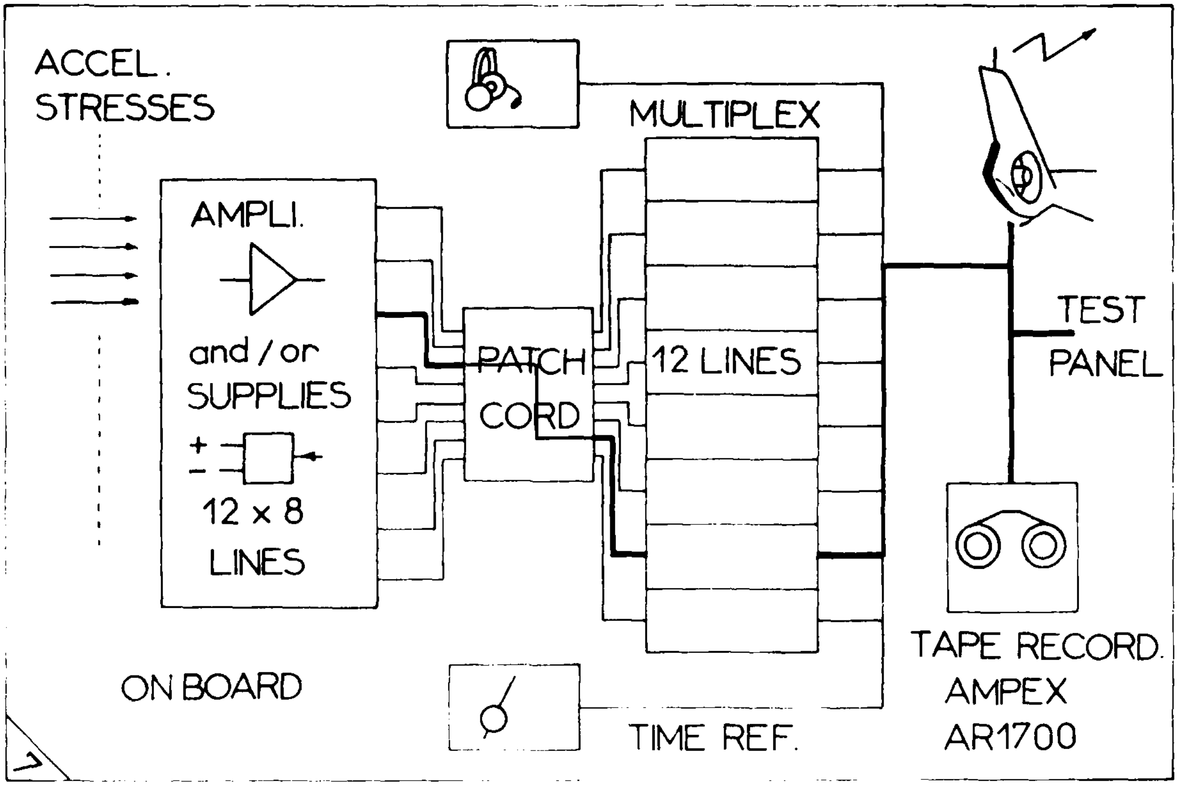


Figure 7

Figure 8 AIRBORNE TEST UNIT

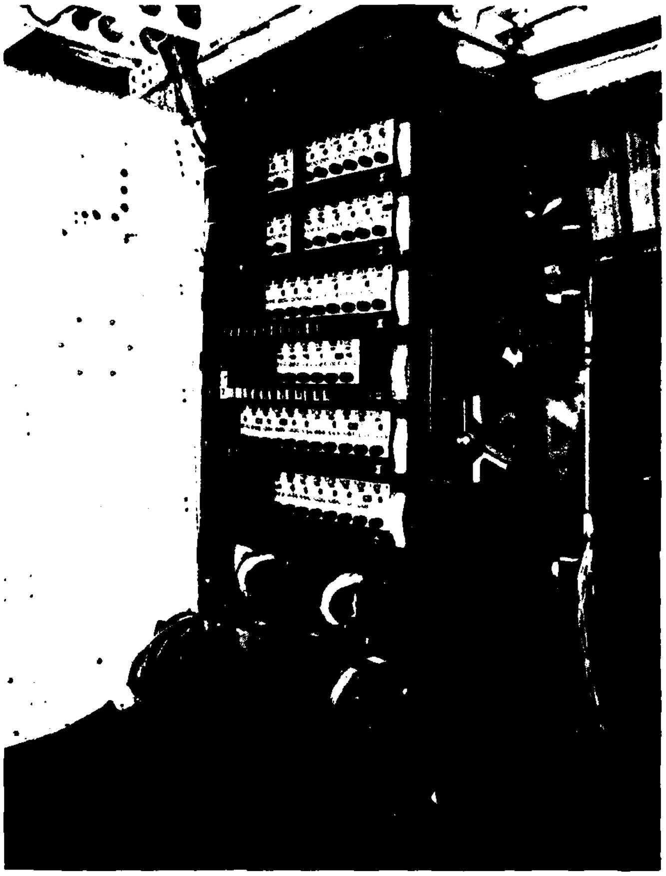


Figure 9 AIRBORNE STRAIN GAUGE AMPLIFIER

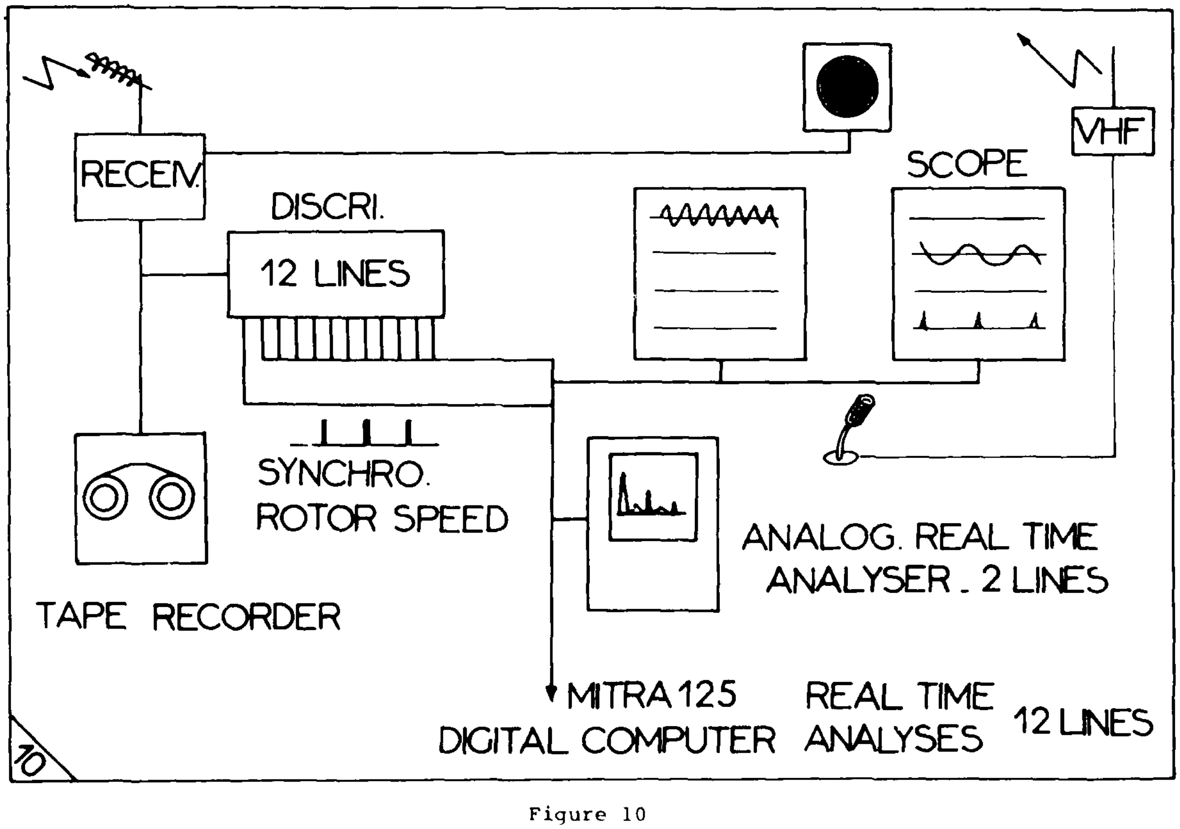


Figure 10

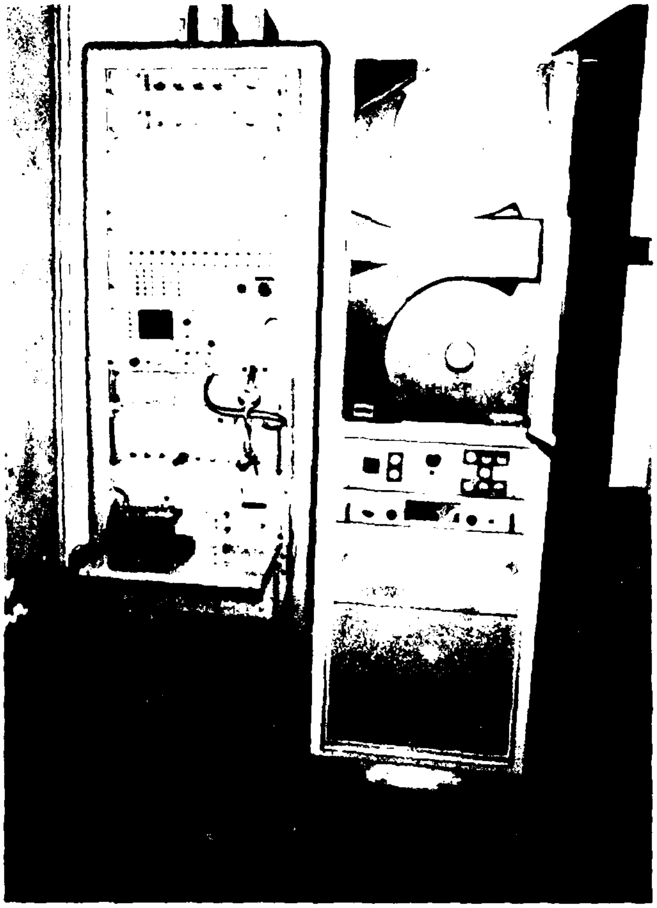


Figure 11 TAPE READING UNIT

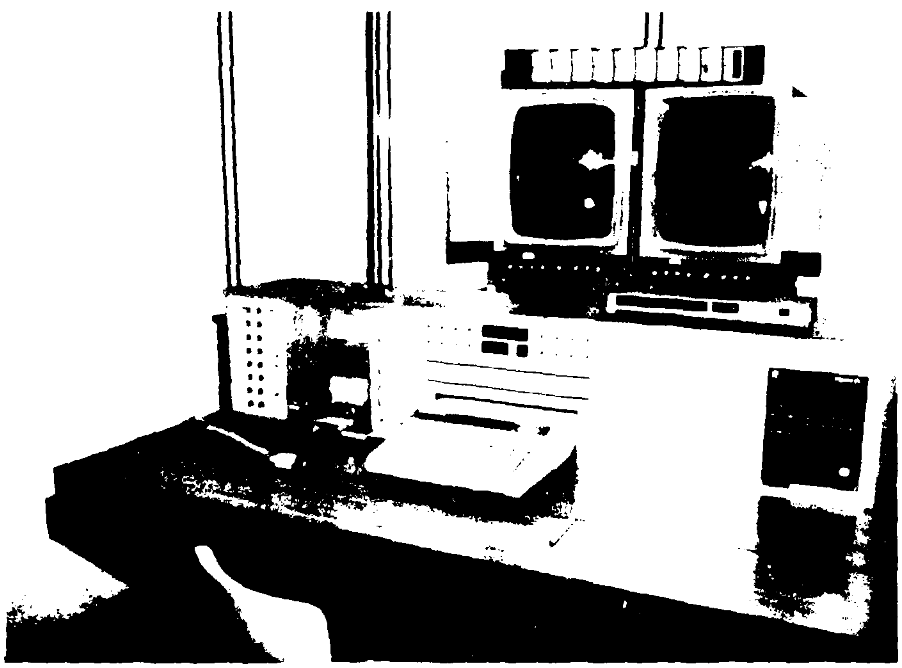


Figure 12 TELEMETRIC DISPLAY UNIT (12 CHANNELS)

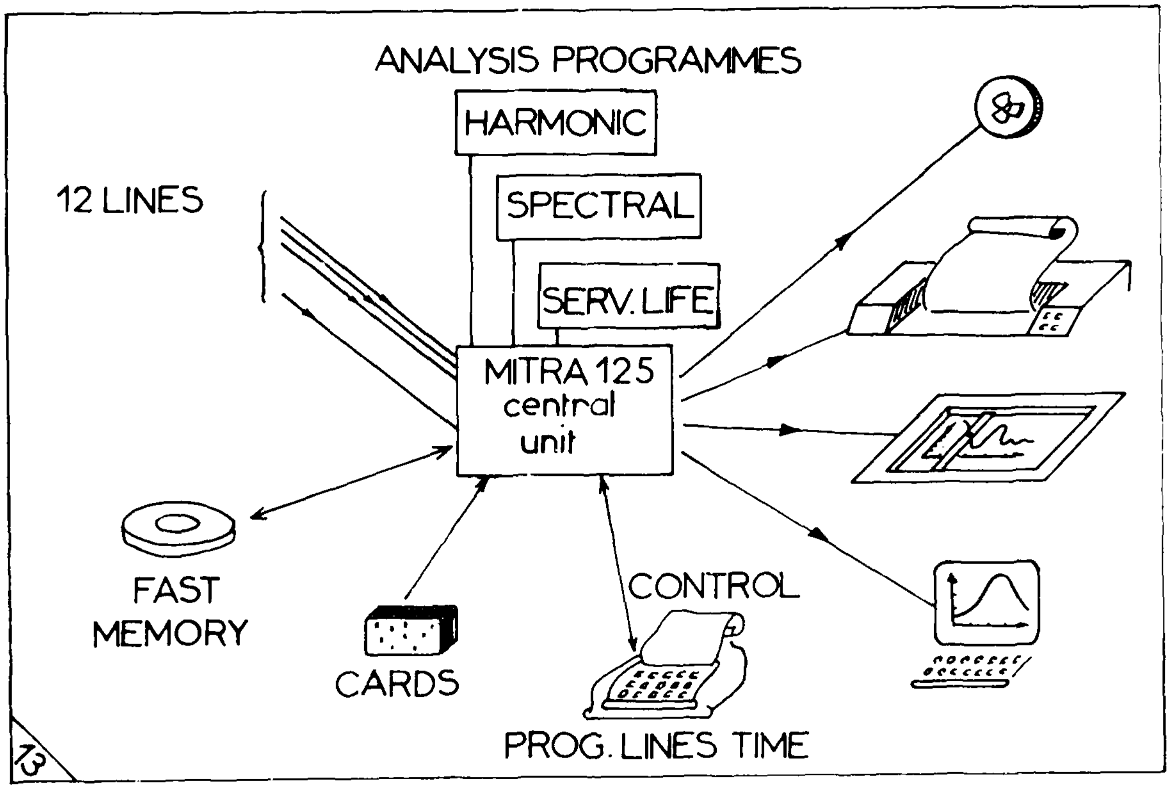


Figure 13

Figure 14 TAPE CONTROLLER

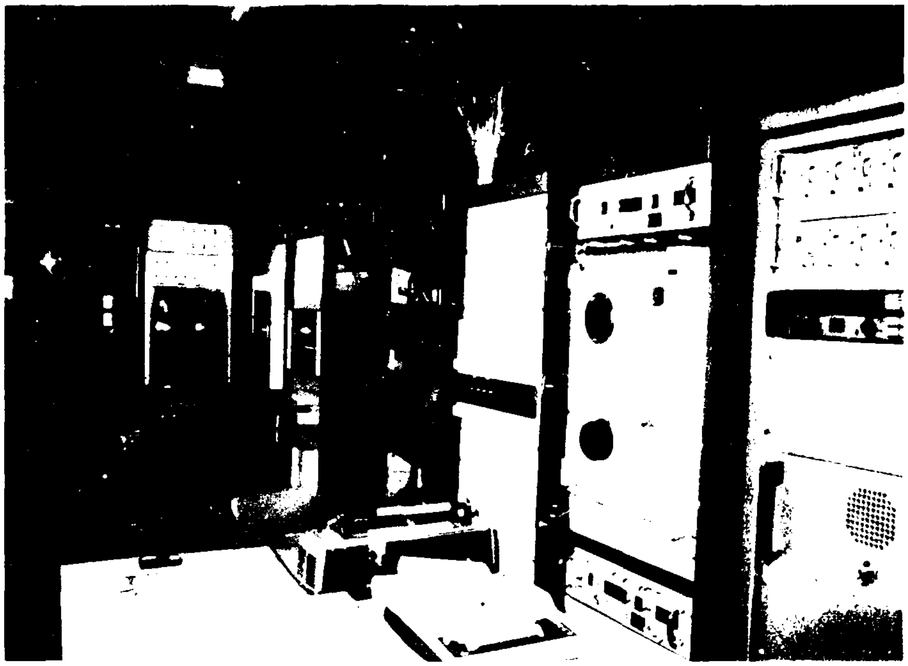


Figure 15 DATA PROCESSING ROOM - GENERAL VIEW

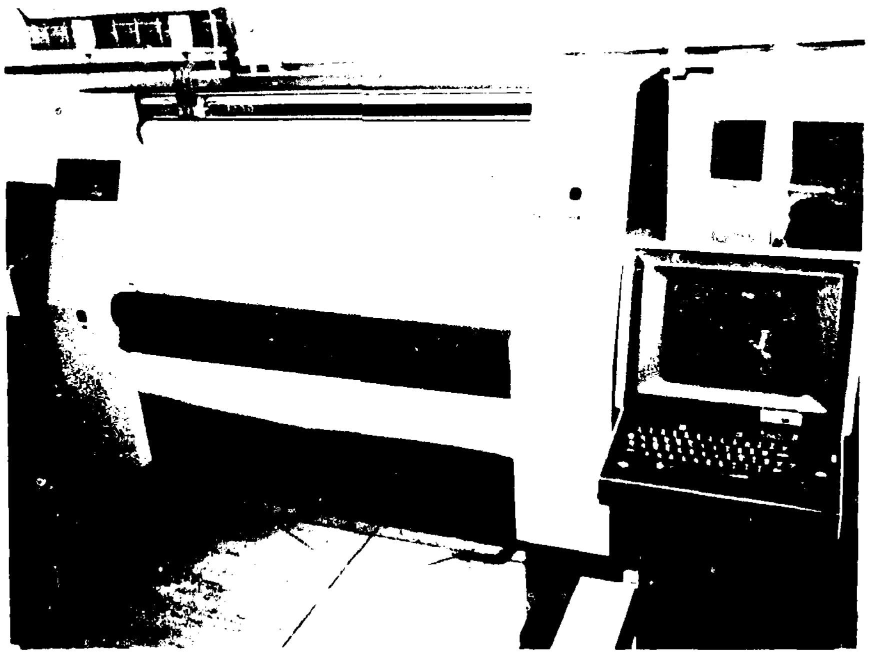


Figure 16 DATA PRINTING UNIT

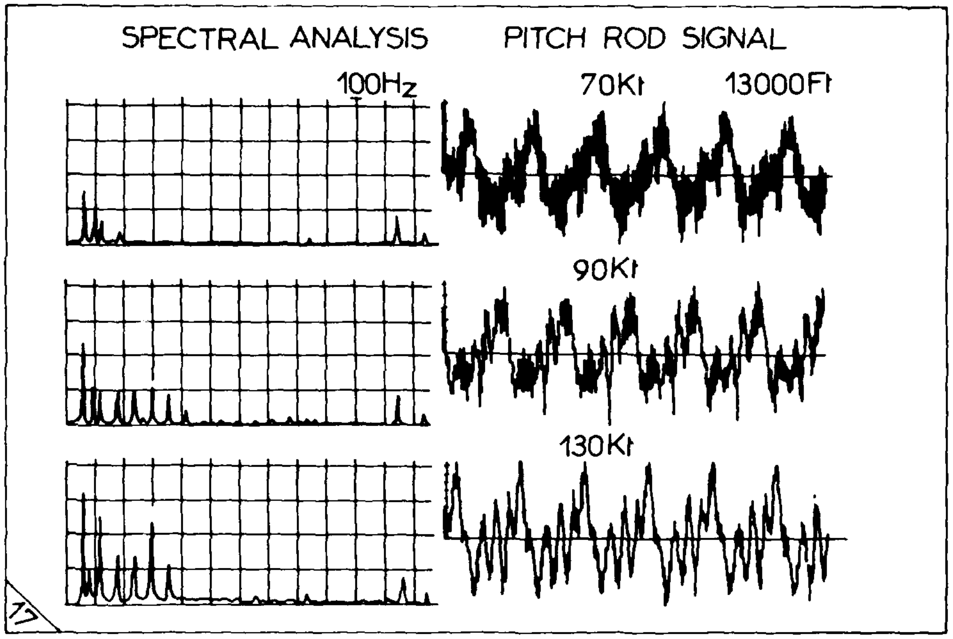


Figure 17

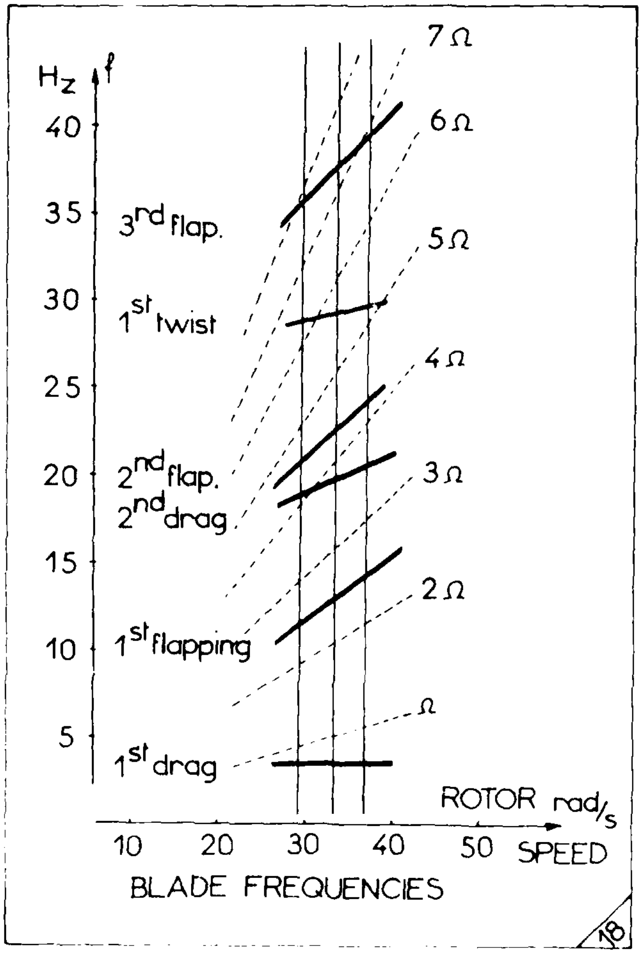


Figure 18

ELECTRO-MAGNETIC COMPATIBILITY
THE DETERMINATION OF SAFETY FOR CRITICAL SYSTEMS

G M Smith, BEM, C Eng, MIERE
Procurement Executive
Ministry of Defence
Aeroplane and Armament Experimental Establishment
Boscombe Down
Salisbury
Wilts, UK.

Summary

The problems of certification of the fitness of military aircraft to enter service have increased significantly with the introduction of electronic equipments into areas of the aircraft which directly relate to primary flight safety. In addition to the effects of self-generated interference, effects due to the external environment must also be considered.

The establishment of adequate margins of safety for these systems requires changes to equipment test methods and procurement procedures. The problems are reviewed and alternative approaches described.

1. Introduction

Where two or more electrical or electronic systems are installed close to each other and are required to function at the same time then there is a possibility of mutual interference. The effect of interference may have two forms - an output from an equipment where none is intended or, alternatively, a failure to respond to an intended signal. Such effects may range from a nuisance - as with domestic radio receivers or recording equipment - to hazardous - as with the inadvertent initiation of electrically detonated explosives.

In the past, it has generally been sufficient to conduct functional tests of the electrical, navigation, radio and radar equipments of aircraft in combination to establish the more easily recognised interference effects. Where no significant effects were found, or had been found and effectively dealt with, a state of electro-magnetic compatibility (EMC) was considered to have been established.

In the case of explosives initiated by electrical energy, which may also be induced by radio transmissions and electrical system disturbances, a more rigorous approach is taken. The minimum energy required for initiation is determined for the detonator in isolation and, using a simulated detonator, the energy levels present in the aircraft are determined during operation of the aircraft systems. EMC is only considered to be established if the ratio of energy levels is greater than an acceptable minimum.

Recent developments in aircraft design have required the introduction of complex electronic equipments into engine and flight control systems to replace mechanical and electrical components. The authority of these systems is becoming such that the effects of interference could present a hazard to the safety of flight and it follows that a margin of safety, similar to that quantified for explosives, needs to be established. The probability of the continuing performance of such equipments is carefully considered during reliability studies. Since interference can well result in an equal loss of performance then equal attention is required to the establishment of EMC.

The performance of these safety critical equipments is achieved by the use of modern semi-conductor components which, due to the lower operating signal levels required and their capability of dealing with a wider range of frequencies, have a higher potential sensitivity to interference. Similar sensitivities are now to be found in existing aircraft where new versions of the older types of equipment are introduced. These vulnerable equipments are now being installed in strike aircraft and helicopters where the most severe interference environments are to be found due to the high packing density of both equipments and wiring. The introduction of digital computers and associated data transmission raises further problems in the definition of appropriate aircraft tests.

Experience has also shown the need to assess the effects of the electro-magnetic fields through which the aircraft must fly. These external fields are produced by communication and radar transmitters sited on land, on other aircraft and on ships whilst the aircraft itself behaves as a combination of crossed dipole antennae. The total problem is illustrated in Figure 1.

2. Safety Margins

The need to achieve EMC in aircraft (and other major systems) including the need to establish safety margins for those equipments which are adjudged to be flight critical, is recognised in Military Specification -E-6051D dated September 1967. This specification requires that the prime contractor shall establish an overall EMC programme and that a board, consisting of contractors, sub-contractors and the Government representatives, shall be set up to control the programme and to expedite solutions.

System requirements are set out, including consideration of the external environment together with the exhortation "that every effort shall be made to meet these requirements during initial design rather than on an after-the-fact basis". A demonstration of compatibility, with the equipments in the final system operated in combination, is required and demonstration of compatibility between aircraft during formation flying is specifically mentioned. The need to establish safety margins for equipments whose upset due to interference could lead to a disastrous result such as loss of aircraft/life or mission failure is stated. The safety margin required is 6dB (a voltage/current ratio of 2/1) for general equipments and 20dB (10/1) for explosives although no justification is given for either value.

A margin of safety in this context can be defined as the ratio by which the level needed to upset an equipment (the susceptibility threshold) exceeds the maximum level of interference likely to be experienced by the installed equipment from aircraft interference sources and from the external environment likely to be encountered during aircraft operation. Present equipment susceptibility tests, such as MIL-461 which is required by MIL-E-6051D and is applied to equipment in isolation during qualification, specify levels which the equipment must withstand. Although these levels have been raised over recent years, the margin by which the equipment meets the specification - which may range from a large to a small amount is not quantified.

This lack of information is particularly acute with regard to the control and signal inputs of equipments which have been found to be the major paths of interference. The radiated susceptibility tests which are designed to establish the adequacy of these inputs suffer from uncertainty of the amplitude of the representative fields generated in existing test rooms. Even if these uncertainties could be removed and the tests could be re-designed to establish the susceptibility threshold, such information would only be of direct use if a comparison with the field conditions present at the equipment installation within the aircraft could be made. The measurement of the parameters of the field within a strike aircraft (E, H and Z - since we are concerned with power transfer) using conventional antenna and measuring receivers is often physically impossible.

Apart from the procedure used with explosives (mentioned earlier) a form of margin has been established by recording the conditions found in aircraft cables and subsequently re-injecting these signals into the aircraft wiring at a higher level. This process (developed by the West German Government agency at E 81) includes amplification of the output of radio transmitters before coupling to the aircraft antenna. This "antenna enhancement" method has been used in the UK in some frequency bands. Apart from the power handling limitations of the aircraft antenna, the output of the transmitter which is to be enhanced presents a problem since the transmitter specification only requires a minimum output to be generated. The maximum output is therefore unknown. A recent UK survey (Reference 1) has shown the conditions at one equipment can vary by a ratio of 4/1 across an aircraft fleet due to the variation of transmitter outputs and differences in the airframe and wiring. Thus the adequacy of the 6dB margin (2/1 ratio) required by MIL-E-6051D must be in doubt.

High power external sources such as the Radio Frequency Environment Generator at A&AEE (Reference 2) have been used as a means of enhancement but, because of the uncertainty of external field effects on an aircraft (Reference 3) it is necessary to measure the normal and enhanced levels at the installed equipment for comparison. Such high power sources are subject to frequency management and a prediction of the frequencies at which the equipment may be susceptible is required before an effective test can be set up. As stated earlier, such predictions cannot be derived from present qualification test results.

3. The Alternative Approach

The optimum solution is seen to be to establish a method whereby the susceptibility thresholds of equipments determined during qualification can be directly compared with the levels found in the installation and the margin of safety established directly. In order to find such an alternative approach we must examine the paths of interference from the various sources to an equipment. These are shown in Figure 2. Interference may enter an equipment through the cables - power supply, signal, control and antenna - or by fields penetrating the equipment case. The interference on the cables is generated by other equipments connected to the same cable, is coupled from adjacent cables within a wiring loom, or results from the effect of fields internal to the aircraft coupling with the loom or cable. Similar fields are incident upon the case of the equipment. The internal fields result from radiations from other equipments and cables and from 'penetration' of fields external to the aircraft. The external fields are, in turn, produced by the aircraft antenna systems or occur as the external environment. The external fields may of course couple directly to the equipment through its own antenna.

The possible alternative to the measurement and simulation of the fields within the aircraft is the measurement and simulation of the interference conditions within the cables at their point of entry to the equipment. This approach assumes that the field does not penetrate the case of the equipment (metal) or is not the dominant interference path. Present experience suggests that this is so except for equipments where such penetration could have been predicted from visual inspection. The only routine measurements that can be contemplated in strike aircraft (and then not without difficulty) are those of the currents in cable harnesses using a clip-on current probe. The special measures required to break into the cables to determine the currents in individual cores, or to measure the resulting voltages, are not usually possible without removing other equipments due to lack of space. Such detailed measurements must therefore be confined to special investigations. The measurement of voltage, besides requiring the disturbance of cables, presents the further problem of the use of high impedance probes which may also be susceptible to the interference environment.

A further problem arises with the measurement of conditions within the aircraft (including current) in that if a cable is used to extract the signal to a measuring instrument outside the aircraft, the interference entering the aircraft through the measuring cable can exceed the interference already present. This problem has been overcome by the use of commercially available fibre optic links which are driven by light emitting diodes and can transmit over a distance of up to 50 metres or more to a convenient measurement point. Frequencies up to 500 MHz may be transmitted with the latest system. The battery powered transmitter which is some 21 cm long and 8 cm diameter must be accommodated near to the measurement point and access provided for the signal and remote control fibres. Since the largest amplitudes of interfering signals have been found to result from the narrow band outputs of the on-board and external transmitters at frequencies related to the aircraft dimensions, the transmitted information may be conveniently displayed and examined on an oscilloscope.

Work is in hand in UK to develop a method of establishing the susceptibility by a bulk current method in which interference is injected into equipment cable harnesses during qualification (or subsequently) by means of a current probe. The difficulty with this approach, which is necessary for large multi-input systems if the test time is to be minimised, is to establish that the cable current divides between the individual cores in a manner similar to the division in the aircraft.

For less complex systems with up to eight inputs/outputs the susceptibility of each port has been established separately by direct voltage injection with the equipment susceptibility being taken as the lowest composite of the levels obtained. The harness (bulk) current has been measured during this process for direct comparison with the aircraft conditions. It is argued that since harness current will not exceed the core current during the equipment tests and that the core current will be less than the harness current during the aircraft tests, a meaningful (although imprecise) comparison can be made. The process is shown in Fig 3, and the results of a comparison made by this method in Fig 4.

4. Design for Compatibility

So far in this paper the problems of aircraft testing, the need for the establishment of safety margins and alternative methods of measurement have been considered. For critical systems, the margin is required to establish a satisfactory state, to allow for variation between aircraft and equipments during production and to allow for changes to the characteristics of both during their life. What margins may be readily obtained and what may be considered to be adequate has yet to be determined.

The procurement of military aircraft is a complex activity which involves the resolution of the requirement for a total weapon delivery system - as a modern aircraft must be regarded - into its constituent parts and the placing of contracts for their design, development and construction. The various parts are then brought together and integrated into the total system. The final measurement of the performance of the aircraft - against the original requirement - is carried out in the UK by the Aeroplane and Armament Experimental Establishment at Boscombe Down and leads to a recommendation for the release of the aircraft to the Services. A simplified form of a procurement cycle and the EMC test process is shown in Fig 5.

More recent UK aircraft requirements have included a clause on the following lines. The aircraft and its installed equipment shall be designed to operate without malfunction or unacceptable degradation of performance in the electro-magnetic environment generated by all installed equipments and when operated in the external environment corresponding to the operational scenario. Responsibility for the achievement of compatibility, including the selection of qualification tests and subsequent integration into the airframe, is placed with the aircraft constructor. This responsibility extends to existing equipments and Government furnished items through a co-ordination contract. An EMC Control Board is set up by the aircraft constructor and visibility of its activities is provided to the Government Project Director and EMC specialists at the periodic meetings of an EMC Working Group.

The highest confidence levels that can be achieved in the compatibility of an aircraft will result from the confirmation of adequate safety margins by aircraft tests on equipments which have been purpose designed and adequately pre-qualified. To this end, the problems of final clearance and the level of confidence required are addressed from the outset of the project.

Whilst many manufacturers are engaged on the design of equipment for compatibility, the lack of the definition of the problem in terms with which they are familiar - the signals to be processed - prevents the establishment of the necessary design measures. It is expected that the specification of the cable currents to be considered will provide a way of meeting this need. The design problem can be reduced to whether or not the equipment can be made into an interference free environment by the addition of filtering to the input and output circuits. If this is not possible, as with high data rate digital transmission, then the interference free zone must be extended to include the wiring harness and drive circuits in such a manner that continuing protection can be assured through the life of the aircraft.

The ultimate solution lies in the understanding of the mechanisms of field penetration and coupling which will allow the prediction of equipment conditions by calculation. Research is in hand and progress is being made, although it is likely to be a number of years before a sufficient understanding will be available.

5. Conclusions

Existing equipment qualification tests, particularly those covering control and signal inputs, do not provide the information required for the formulation of aircraft test plans, or for comparison with the conditions present in aircraft to provide a measure of the safety margins available. It is now necessary to establish these margins for the electronic equipments in flight safety critical areas of aircraft in addition to explosives.

The requirements of MIL-E-6051D can be met by a proper comparison of equipment wiring harness currents measured during equipment and aircraft tests.

Responsibility for the achievement of compatibility is now placed with the aircraft constructor with the objective of achieving compatibility during the design process. It is anticipated that this can be achieved through the estimation of the interference cable currents likely to be incident on the equipment in the final aircraft installation.

Reference

1. The variation of induced currents on aircraft wiring. P M NEWTON, IERE conference on EMC - Southampton 1980.

2. Some UK techniques for assessing RF susceptibility of electro-explosive devices in aircraft systems. Sqn Ldr T J JENKINS and Mr W OLDHAM, IEEE International Symposium on EMC - Washington DC 1976.

3. The testing of aircraft under near field conditions. M ELLIOTT, IERE conference on EMC - Southampton 1980.

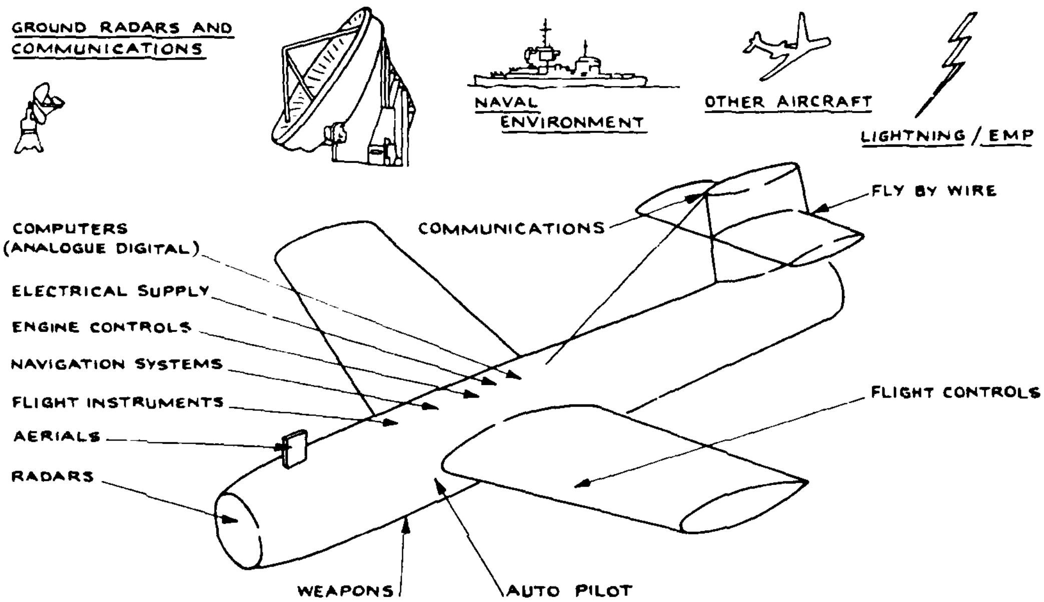


FIG. 1 THE AIRCRAFT PROBLEM

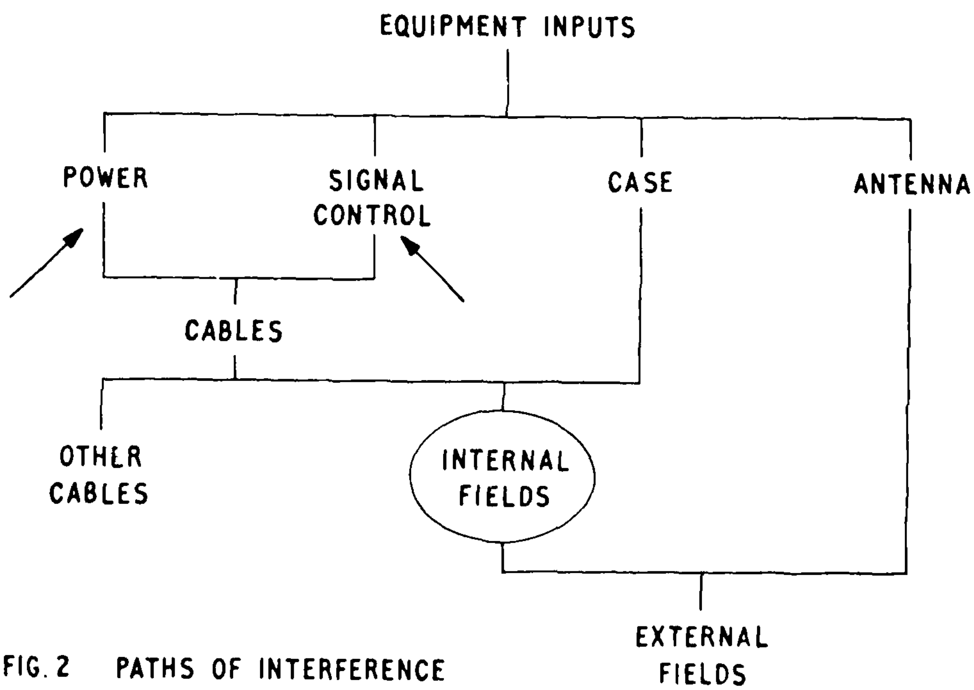


FIG. 2 PATHS OF INTERFERENCE

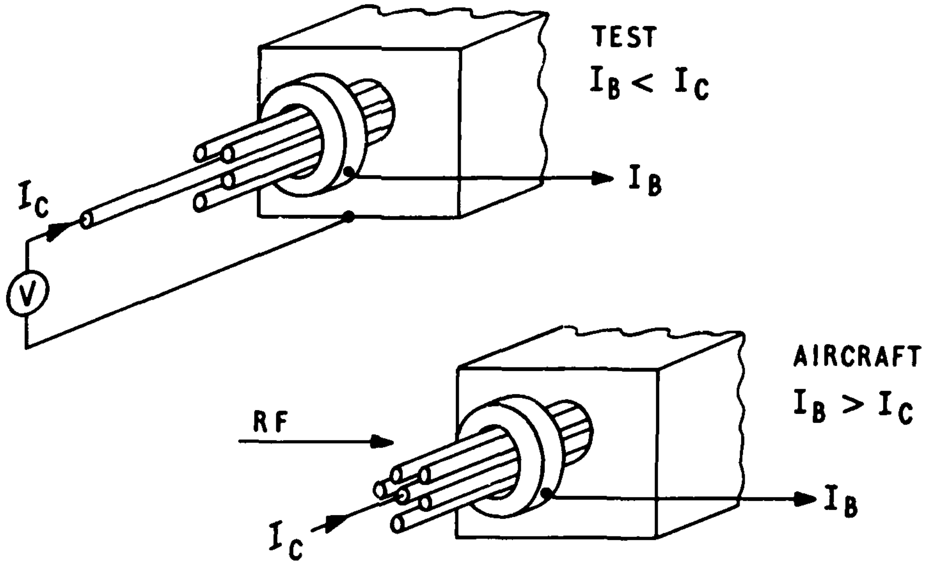


FIG. 3 COMPARISON OF BULK CURRENTS.

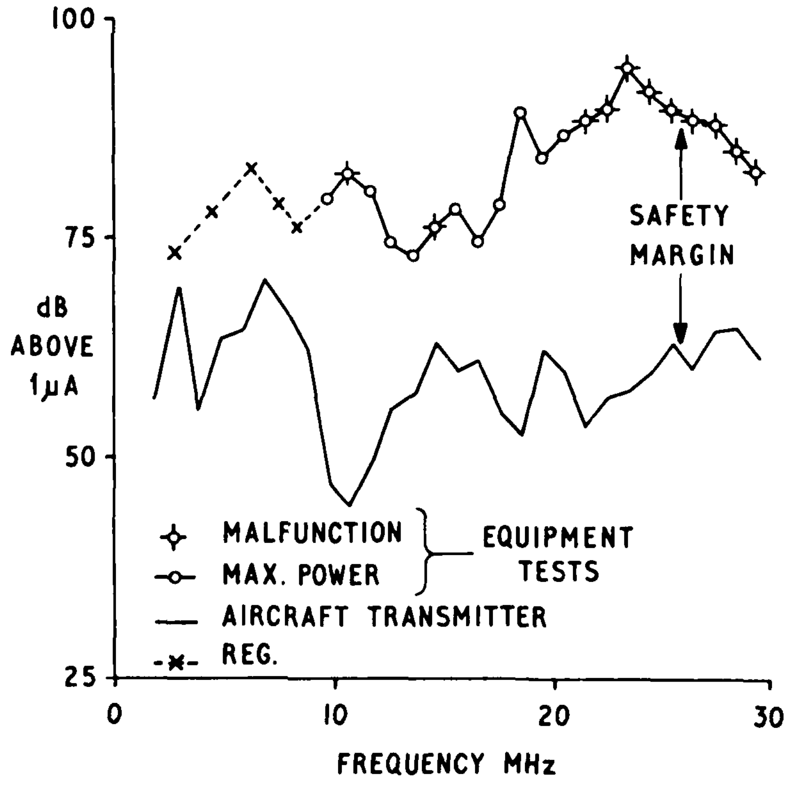


FIG. 4 BULK CURRENT COMPARISON.

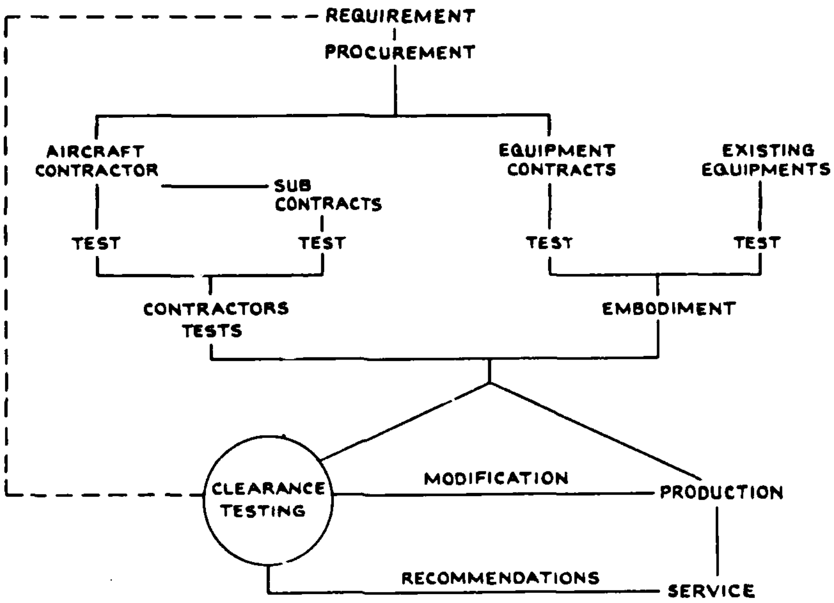


FIG. 5 AIRCRAFT PROCUREMENT CYCLE.

RELIABILITY AND MAINTAINABILITY EVALUATION DURING INITIAL TESTING

by

Jan M. Howell
6510 Test Wing/TEEES
Edwards AFB, California, U.S.A. 93523

SUMMARY

Extensive reliability and maintainability (R&M) evaluations, both quantitative and qualitative, are accomplished during initial testing to ensure that the highest quality weapons system is delivered to the user within existing acquisition cost and schedule constraints. Because of the ever increasing cost-of-ownership of modern weapons systems, the emphasis on these evaluations is increasing proportionately. This paper presents an overview of these evaluations. The objectives, methodology used and information available from such evaluations are discussed. The statistical analysis and methods normally associated with the R&M engineering discipline is deemphasized or relegated to appendices when essential.

INTRODUCTION

As modern weapons systems continue to increase in capability the complexity and acquisition cost increase also. These ever upward spiraling costs dictate that a weapons system be as reliable and as maintainable within cost, performance and delivery schedule constraints. To achieve a satisfactory level of reliability and maintainability a system must be tested to determine the baseline characteristics. Then appropriate action must be taken to overcome discrepancies.

The complexity which caused initial high acquisition cost also caused lengthy time delays in performing any redesign or buying different components to implement a redesign. It is common for problems discovered in early testing to not be fixed before 100 production systems have been deployed. To minimize the number of systems which have to be retrofitted, evaluations must be accomplished as early as possible in the weapon system development cycle.

In the late 1960s, the USAF Flight Test Center recognized these requirements and pioneered the development of techniques to accomplish R&M evaluations during initial testing. During the F-111, C-5A, A-7D and UH-1N test programs this methodology was developed and served the test program with steadily increasing degrees of success. By 1973 this methodology had matured and has since been used in every new aircraft test program conducted by the USAF Flight Test Center. Most recently this methodology was used during the Air Launched Cruise Missile competitive evaluation between the AGM-86 and the AGM-109.

OBJECTIVES

To achieve a quality weapons system at minimal cost requires that certain specific objectives be attained. Specifically, manufacturer compliance with contractual requirements must be evaluated, resource requirements for the mature system must be estimated and deficiencies (or opportunities for improvement) must be identified.

Contractor Requirements:

From the standpoint of measuring the contractor's success in meeting requirements, every evaluation must be tailored to measure the parameters contained in a given contract. Table 1 contains a list of some of the more critical parameters commonly used to specify reliability and maintainability characteristics in contracts for new aircraft weapons systems. Often these parameters are tailored to suit the specific weapons system.

It is important to note that both the buyer and the seller should realize that the first versions of a new weapons system will not demonstrate the R&M characteristics that can eventually be achieved by a mature version of the weapons system. Design error, lack of established quality control procedures, untrained manufacturing personnel, and lack of production tooling are among the many reasons that the first produced versions of an aircraft are not as reliable as aircraft produced later in the production run. The rate of improvement and the factors affecting it are the subject of a well known paper by Mr. J. T. Duane of the General Electric Corporation (reference 1). The Duane reliability growth theory has been the subject of some controversy but the controversy concerns the rate of reliability improvement, not the existence of reliability improvement. Appendix A presents more detailed information on the Duane reliability growth model. Figure 1 shows the Mean Time Between Maintenance Action growth for several recently deployed USAF aircraft.

The most accurate measurement of R&M characteristics can be made during testing of the weapons system when the environment is controlled to a large degree. Unfortunately at this point in the systems' life cycle, the ultimate R&M values have not been achieved. Later in the systems life cycle, mature R&M characteristics exist but the environment is poorly

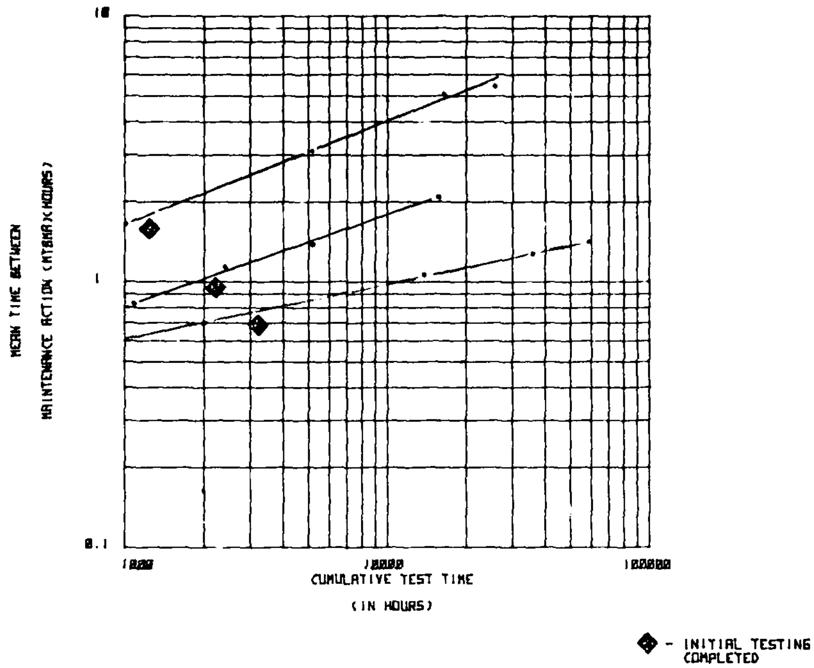


FIGURE 1

DUANE RELIABILITY GROWTH MODEL FOR

SEVERAL RECENTLY DEPLOYED USAF AIRCRAFT

known and accurate failure data is generally not available. To resolve this paradox, many weapons systems contracts specify intermediate R&M characteristics that must be achieved prior to the end of testing. It is these intermediate or "degraded" values of R&M characteristics that the weapons system is measured against during initial testing.

Table 1

TYPICAL RELIABILITY AND MAINTAINABILITY PARAMETERS
SPECIFIED IN AIRCRAFT CONTRACTS

Logistics Reliability

- Mean Time Between Failure
- Mean Time Between Corrective Maintenance Action

Operational Reliability

- Mean Time Between Mission Critical Failure
- Probability of Mission Success

Maintainability

- Maintenance Man-Hours per Flight-Hours
- Maintenance Man-Hours per Sortie
- Maintenance Personnel per Aircraft Squadron
- Mean Time to Repair
- Maximum Time (90th percentile) to Repair
- Mean and Maximum Specified Task Time (preflight, turnaround, etc.)

Often the contractual definition of R&M parameters is different than the accepted (or real world) definition used by the USAF. The underlying reason for this is that the contractor can only be held responsible for those problems which are caused by the system design and manufacture. The contractor cannot be held responsible for anything which is a function of the environment in which the system is operated or maintained unless that environment can be clearly defined in the contract. Examples of occurrences which must be excluded from the calculation of R&M parameters include: operator/maintenance error, foreign object damage and accident. Many contracts also exclude "wear out" items such as tires from the definition of failure. It is extremely important that these definitions be delineated as completely as possible in contractual documentations to prevent misunderstandings and potential litigation.

It is also important that the contractual definition of R&M parameters be as close as possible to the accepted definition and that any differences be well understood by the contractor, program office, test agency and eventual user. Figure 2 shows two Mean Time Between Failure (MTBF) statistics for a system where the contractual definition of failure was considerably different than the accepted or standard definition. In this case, the only failures which counted contractually were those problems which would prevent completion of the planned mission and which were discovered after the aircraft had been given a preflight inspection. These rules excluded a large number of failures. As a result, the difference between the contractual and actual MTBF was about one order of magnitude.

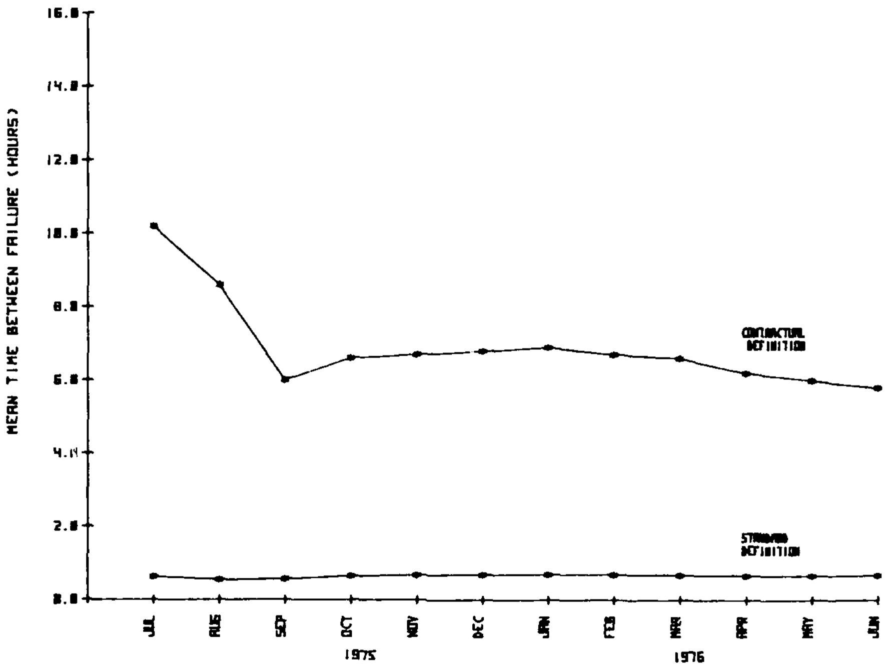


FIGURE 2

CONTRACTUAL AND STANDARD

DEFINITION OF MEAN TIME BETWEEN FAILURE

Estimating Mature System Resource Requirements:

A second objective of initial testing is to forecast resource requirements and operational achievements of the mature system. Table 2 lists some of the characteristics measured during initial testing. Some words of amplification must accompany that table. Obviously, a 1,000-hour test program cannot provide sufficient data to enable one to determine the number of spare main landing gear axles required when the axle has a projected 50,000-hour life. But accurate usage projections can be made for components with a more limited life. For example, the average number of landings per tire life can be accurately determined. Similarly, the mean time between maintenance action for a simple, highly reliable subsystem such as an aircraft interphone is hard to determine with confidence, but for a more complex system such as the fire control radar the failure frequency is correspondingly higher and thus easier to measure during a short test program.

Although many individual results from initial testing are obviously either valid or invalid, a large class of results cannot be so easily classified. For this reason it is necessary to resort to some statistical tools. Various tests of significance have been used in the past but the most meaningful approach seems to be the use of lower confidence limits. The use of confidence limits allow one to state that if a given test were to be repeated a large number of times then for any given percentage of the time the results would be no worse (or better) than the lower (or upper) confidence limit. For example, if an article is tested 1,000 hours and fails 10 times then MTBF is 100 hours and the 90th percentile lower confidence limit is 66 hours. This states that 90 percent of the time the MTBF will not be lower than 66 hours. These limits are useful in conveying the relative uncertainty of results while simultaneously presenting the possible "worst case" that may exist. For these reasons, lower confidence limits are calculated and reported along with actual results.

Table 2

RELIABILITY AND MAINTAINABILITY CHARACTERISTICS
MEASURED DURING INITIAL TESTING

Reliability

- Component Life
- Mean Time Between Demand (time between need for a spare)
- Retest OK Rate
- Repairable Rate

Maintainability

- Task Times
- Facilities Requirement
- Support Equipment Utilization

Operational Availability

- Mean Down Time
- Sortie Generation Rate
- Operational Ready Rate

Identification of Deficiencies:

The last and most challenging objective is the identification of deficiencies (or opportunities for improvement). Of the new aircraft weapons system tested at the USAF Flight Test Center in the last ten years an average of 800 deficiencies were found on every test program. These 800 deficiencies range from the trivial, easily fixed category to the very significant problems requiring major redesign efforts. Approximately 50 percent of the 800 deficiencies were directly related to reliability and maintainability problems.

One example of a serious yet easily fixed problem was an aircraft that was delivered with inadequate markings to show what areas the maintenance technicians could walk on. The potential for costly damage was high but the problem was quickly fixed with paint. Another serious, but easily fixed problem involved the rubber bumper pads the main landing gear tires rested against when the landing gear was retracted. The original installation pads were glued on the mounting structure and, after some brief usage, fell off. More serious than the pad loss was the fact that without the pads to stop the tire, the gear traveled too far during retraction and damaged the electrical switches which indicated when the landing gear was in the up and locked position. After the switches and pads were replaced, the aircraft had to be placed on jacks and the landing gear cycled between the up and down position to insure correct operation. This checkout procedure was costly in terms of man-hours and aircraft nonavailability. Riveting the pads in place solved the problem.

Unfortunately, the problems that are easily fixed are those that shouldn't have occurred in the first place. Generally they are a result of oversight or just plain sloppy design work. Problems that are more difficult to correct often result when weapons system performance is expanded at the cost of reliability or maintainability. A good example is the enhanced vision or "bubble" canopies used on modern tactical aircraft. When the desired vision is achieved, the mechanism made to function throughout the required temperature range, and the entire system made to function in icing conditions, very little room is left for the designer to optimize maintainability.

As a result, the early versions of the "bubble" canopies were very difficult to adjust. When these canopies are maladjusted, the canopy either refuses to lock closed (ground abort), leaks pressure (a probable air abort) or refuses to unlock after landing (which upsets aircrews somewhat). The task is so difficult that one aircraft required several hundreds of man-hours to change a canopy. As a result, the manufacturer undertook at his own expense a major redesign to lower the man-hours required for initial canopy installation and adjustment because the cost of reengineering the system would be recouped during production.

During a test program identification of deficiencies is a major consumer of R&M engineering man-hours. When an 18-month, 1,000-hour flight test program uncovers 400 R&M related deficiencies this means that the R&M engineering group averages one new problem every working day. Each problem must be thoroughly researched to insure that all causal aspects of the problem are adequately documented. This research amounts to gathering all pertinent evidence regarding the problem. Such evidence might consist of statements from operations or maintenance personnel, photographs (or video tape) of damaged parts or problem processes, drawings or instrumentation data. The analogy between the test engineer documenting a problem and a lawyer building a case is strong and appropriate. Sufficient analysis must be performed to determine the impact of the particular problem. Any recommended changes to alleviate the problem must be identified and validated. Most importantly, the problem, impact, and any recommendations must be thoroughly documented and promptly reported to the agency which has the authority to approve needed changes. Lastly, when a fix is implemented, it must be tested to insure that the problem was really fixed. A surprising percentage of fixes either do not correct the problem or introduce new problems. Perhaps this follows from the fact that if the design was difficult to accomplish the first time, it is no easier the second or subsequent times.

Once the information concerning deficiencies has been disseminated to program management and contractor personnel the R&M engineering group is often called upon to further explain the problem, to defend judgments concerning problem impacts and provide rationale for suggested changes. But the return on engineering hours expended is absolutely worth the effort. The reliability improvement discussed earlier does not occur if the problems are not isolated and effective corrective actions taken.

PROGRAM PLANNING

The first step towards accomplishment of these objectives is proper R&M evaluation program planning. To accomplish the planning tasks, a Joint Reliability and Maintainability Evaluation Team (JRMET) is formed six months to a year prior to the actual start of testing. This team is composed of representatives from the developing agency, the eventual using command, the supporting commands, the test organizations and the contractors. Among the tasks accomplished by the JRMET during the planning phase are:

1. Insure adequate training for operations and maintenance personnel prior to start of evaluation.
2. Insure availability of technical data.
3. Insure availability of ground support equipment.
4. Determine evaluation ground rules.
5. Determine requirements for data collection and processing.
6. Determine responsibility for data collection and processing.

R&M DATA COLLECTION

The tool most essential to the accomplishment of these objectives is a comprehensive R&M data collection system. Information gathered by this system must be of sufficient detail to allow engineering personnel to assess the need for changes and estimate the effect of any such changes. While such detailed data collection would be cost effective for a mature, well understood weapons system, it is essential for new systems. Generally, the data required to perform an R&M evaluation can be divided into two broad categories: Operations (or usage) data and maintenance data.

Operations (usage) Data:

The operations data must gather all information regarding equipment usage. This must include the time or cycles the system was operated, the environment the system was operated in, and the type of mission that was being attempted. For aircraft where not all subsystems were needed or used for every mission, the subsystems that were used and their effectiveness must be recorded. Any anomalies noted by the system operator must be documented in sufficient detail to facilitate further investigation by maintenance and engineering personnel.

Maintenance Data:

Aircraft maintenance actions can also be divided into two categories: scheduled maintenance (servicing, inspection, etc.) and unscheduled maintenance (corrective maintenance). Data from either type of maintenance must show all resources that were used during the action. These resources include:

Personnel (Types/numbers of maintenance technicians and time required)

Facilities (Engine run pad, fuel cell repair area, etc.)

Expendables (Fuel, oil, lox, etc.)

Ground Support Equipment (Types and time used)

Spares (Replacement for broken parts)

The resources used measure the cost of doing maintenance and must be documented accurately. For corrective maintenance, a complete and detailed description of the problem must be recorded. Additionally, a detailed description of the actions taken to investigate and remove any anomaly must be recorded.

The required accuracy does not come automatically. The maintenance technicians must be trained to fill out the required forms completely. Most importantly, the technicians must be motivated to provide the required documentation accuracy. The necessary motivation is achieved when the technicians understand that the information they provide is their method of participating in the test and that it is needed and used to obtain a better weapons system.

Another facet of R&M data collection that requires particular attention is failure analysis information. Generally, this information must be obtained from the contractor or his vendors. The failure analysis process can be expensive but it is necessary to know the exact cause of failure before any corrective redesign can be considered. Provisions for detailed failure analysis should be contained in contractual documents with the results to be provided to the customer. Even with such appropriate contractual requirements, it is necessary for an R&M engineer to monitor the flow of failure analysis information to insure that the test agency receives the needed information.

Data Processing:

Once the raw data forms are completed, they must be processed to arrange the information in a manner most suitable by engineering personnel. The processing system used at the USAF Flight Test Center is called the Systems Effectiveness Data System (SEDS) and is a computerized information processing system designed to be used on large scientific computers such as the Control Data Corporation's Cyber series machines.

The data forms are keypunched and the cards entered into the system where the data is validated by software routines which identify and reject data containing machine detectable errors. Once the data is free of machine detectable errors, the system processes the information into a data base for subsequent analysis. The most important facet of the data base is the organization of the data entered. The maintenance actions required to complete the investigation or repair of a single anomaly occur at different locations (flightline, intermediate shop, depots or contractor) and at different times. Many months often elapse between the discovery of the original problem and the eventual repair at a vendor's facility. The system collects all maintenance actions related to a given problem occurrence, groups them together and in chronological order, and calculates such parameters as total man-hours expended on that problem occurrence. Once all related data is gathered and linked, the engineer is able to see the complete sequence of actions from when the problem is discovered until final resolution is reached.

TEST AND EVALUATION

When the planning is complete and arrangements are made to acquire and process the necessary data the stage is set for the actual evaluation to begin. It is important to note that no aircraft flight time is specifically dedicated to R&M evaluations. Instead, the R&M data results from the flight operations needed for other testing. Some dedicated time on the aircraft is needed to perform maintenance demonstrations, but these demonstrations are performed in conjunction with routine maintenance or the technical data verifications and do not significantly impact the overall test program.

Data Review:

In addition to the time spent identifying deficiencies discussed earlier, an important part of the evaluation is the review of R&M data which is the primary JRMET function during the evaluation. Every maintenance action must be reviewed to:

1. Insure accuracy of data by making necessary corrections and additions.

2. Classify failures as critical or noncritical, relevant or nonrelevant, etc.

3. Determine need for corrective redesign.

Once the data is reviewed/corrected, R&M results such as failure rates, maintenance man-hour requirements, etc., can be computed and reported. Judgment can be made regarding contractor success in meeting requirements. Mature system resource consumption estimates can be made and any required adjustments to planned resource allocations can be effected.

CONCLUSIONS

In order to identify deficiencies as early as possible in the weapon system development cycle, R&M evaluation must begin with the start of initial testing. With adequate planning, trained R&M engineering personnel and a suitable data collection system a majority of the R&M deficiencies should be discovered before the system is delivered to the eventual user. Similarly, manufacturer success in meeting R&M requirements can be determined and resource requirements for the mature system can be estimated.

REFERENCES

1. Duane, J.T., Technical Information Series Report DF62MD300, General Electric Company, DCM+G Department, Erie, Pennsylvania, 1 February 1962.

2. Codier, E.O., "Reliability Growth in Real Life", Proceedings of 1972 Reliability and Maintainability Symposium, February 1972.

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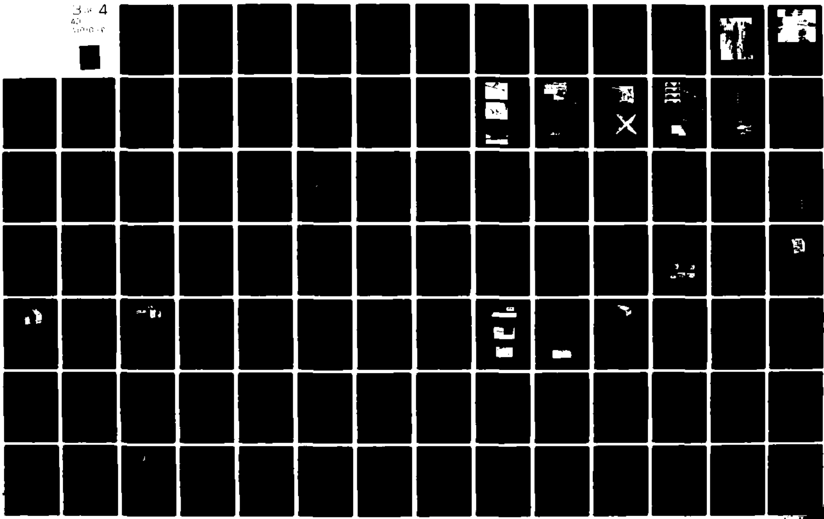


APPENDIX A

RELIABILITY IMPROVEMENT

Reliability improvement has been a subject of significant discussion for many years and numerous techniques to forecast the rate of reliability improvement have been used. In 1962, Mr. J. T. Duane of the General Electric Corporation published a paper describing a reliability growth model that has, over the intervening years, come to be accepted as the standard for explaining positive changes in reliability. The Duane theory postulates that:

$$\lambda_{\Sigma} = \frac{F}{H} = KH^{-\alpha}$$

where:

λ_{Σ} = Cumulative Failure Rate

H = Total Test Hours

F = Number of Failures During Time H

K = Constant Determined by Circumstances

α = Growth Rate

From an applications viewpoint, this model represents practical reliability growth so well that when empirical data does not agree with the model, the data should be considered suspect (reference 2). In addition to the accuracy with which the Duane Theory models reality there is another significant point in its favor. When plotted on logrithmic chart paper (both axes), reliability improvement appears as a linear relation of test time as in figure A-1.

There is one slight drawback to presenting data in the form of figure A-1 and that is the question: "If reliability is improving, how come the graph goes down?" To answer this question (usually originating from management) it is necessary to depart from the basic Duane theory and plot the reciprocal of failure rate, (i.e., mean time between failure) as in figure A-2.

One additional amplification is also useful. By definition, the basic Duane Theory deals with the cumulative reliability but it is normal to want to know the reliability at different specific times. Differentiating the cumulative failure rate with respect to time in Duane's expression yields:

$$(1-\alpha)\ KH^{-\alpha} \text{ or } (1-\alpha)\lambda_{\Sigma}$$

Then, the instantaneous failure rate is: $(1-\alpha)$ times the cumulative failure rate. Similarly, the current mean time between failure rate is $1/(1-\alpha)$ times the cumulative mean time between failure.

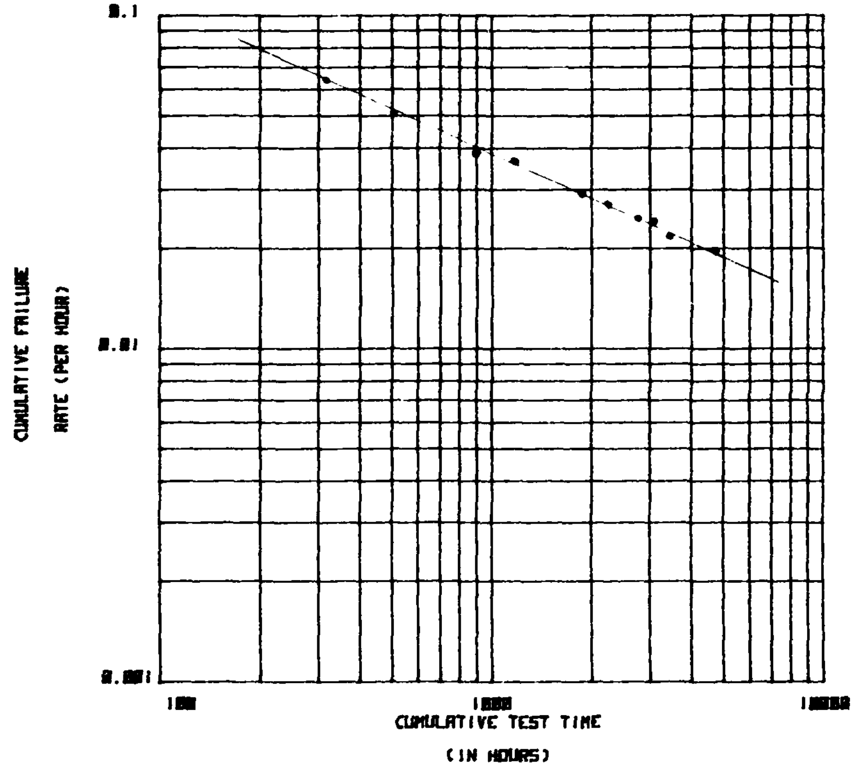


FIGURE A-1
DUANE RELIABILITY GROWTH MODEL
FAILURE RATE VERSUS TEST TIME

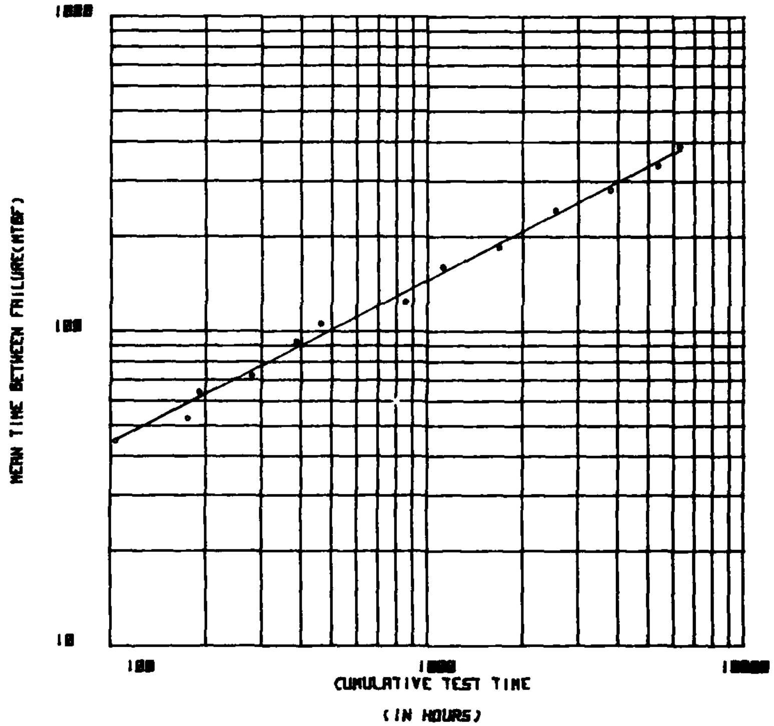


FIGURE A-2
DUANE RELIABILITY GROWTH MODEL
MTBF VERSUS TEST TIME

SUMMARY OF SESSION II - AIRCRAFT SYSTEMS TESTING

Though the basic harmonization of guns and gunsights can be evaluated by air-to-ground gunnery, air-to-air gunnery is essential for the final evaluation because of factors such as fuselage bending during high g maneuvers. The air-to-air evaluation is accomplished using towed targets with miss-distance measurement capability which utilizes the shock waves of the bullets for counting and distance calculation. At the RAE, not only the number and the range but a quadrant direction of the bullets can be detected by an acoustic miss-distance indicator based on the signals of four microphones placed on the target. At the AFFTC, evaluation software utilizes data from head-up display film and on-board instrumentation to generate a "bullet-line" that would have occurred if bullets had been fired at the test conditions. Though the equipment and therefore the costs of tow targets are increasing, the inherent accuracy can be used to reduce the number of flights in the development and evaluation phase of gun and gunsight testing.

The engine installation in military aircraft requires close cooperation between the aircraft and the engine manufacturer to optimize the engine-airframe integration. The discussion pointed out the large differences between the requirements for civil and military engines. The military engine is operated nearer to its limits, so compressor stall margins, high angle of attack operation, hot gas reingestion from gun or missile firing and the dynamic behavior at abrupt changes in the power setting are potential problems, whereas in civil engine performance, fuel consumption and the certification are the main concerns of the flight testing.

The problem of terrain following (TF) flight is to select the optimal TF height above the ground as a compromise between being low enough not to be detected and being too low to have adequate obstacle clearance. The major task of testing these types of systems is the confirmation of performance and safety. As the TF will be largely operated in an automatic mode during operation under all-weather conditions it is evident that the pilot must be involved in the very early test and simulation phase to define the limits of the system for the high risk missions over rough terrain and to know all the effects of various failure modes.

The necessity for interrelated design of the entire landing gear system was strongly presented. The system concept must include the gear structure, tires, brakes, control devices, and must also cover all modes of operation of the gear and aircraft. Continued advances are being made in the design and construction of each sub-system and test techniques to allow determination of system performance. Considerable discussion evolved on tires, tire pressure measurements, and runway friction measurements. Some ideas were expressed to require better standards for tires - especially retread tires- and to give pilots more instrumentation relating to the condition of their tires or landing gear. Others preferred to have the aircraft operators devise better preflight inspection and control systems for landing gears, i.e., no more governmental standards. Little trust was expressed for any of the so-called runway friction measurement techniques now in use.

Design techniques and the instrumentation to check performance for helicopter rotors operating at vehicle speeds of 200 knots were presented. Data were taken from the rotor through slip rings to on-board instrumentation to allow the pilot to determine safe stress levels as flight conditions were changed. Rotor vibration levels as felt through the structure by the crew were not a safe method for limiting stress levels. A telemetry unit was used to transmit data from the aircraft to a ground station for complete analysis.

The problems produced by unwanted electromagnetic signals in aircraft were outlined. Although electronic equipment in aircraft is shielded by metal boxes and metal aircraft structuring, the wiring systems in aircraft act as antenna and pick up various signals from sources on-board the aircraft as well as outside. The wires conduct these signals through bulkheads and safety covers. Math models have been developed for some simple wiring systems. A strong point was made for requirements to be made early in the design stage of an aircraft to reduce electromagnetic interference problems.

The deficiencies and their effects on reliability of new flight systems were outlined. Methodologies to accomplish R&M evaluations during initial test programs of a vehicle were presented. There was agreement that good R&M started with proper requirements for the vehicle or system. Thereafter, R&M should be a special subject for the contractor. Techniques to check R&M and to correct defects are available, but fixes are expensive. Do it correctly the first time.

UNIQUE TEST CAPABILITIES OF THE EGLIN AFB MCKINLEY CLIMATIC LABORATORY

Richard D. Toliver
McKinley Climatic Laboratory
Eglin AFB Florida 32542

SUMMARY

The technical facilities of the Eglin Air Force Base McKinley Climatic Laboratory are described. The major facilities include the 93,000 cubic meter Main Chamber and the 2,440 cubic meter Engine and Equipment Test Cell.

INTRODUCTION

Do simulated climatic tests yield results similar to field tests under natural conditions? This question is often asked and has been studied extensively at the Climatic Laboratory. Problem areas in successful simulation include precise identification of degrading environmental factors, determining appropriate length of laboratory and field tests, interaction effects of multiple natural influences, variability of results in nature, over and under severity of laboratory tests and lack of quantitative scaling factors between laboratory results.

By comparing reports of natural environmental tests and reports of testing accomplished at the Climatic Laboratory, it has been determined that chamber tests are highly preferable to field tests in terms of time, cost, convenience, and precision. However, chamber tests often do not predict long term effects of environmental exposure.

Most problems of simulated testing disappear when tests are performed on full scale, production or pre-production hardware. Because of the enormous size of the Climatic Laboratory facilities, "field" testing may be accomplished to a great extent, inside the test chambers. At the Climatic Laboratory a test team has the flexibility of testing in a natural environment and with precision obtainable only inside a controlled chamber.

HISTORY

Interest in environmental testing of Air Force equipment dates back to 1934 when the Baker Board recommended that tactical units be trained in winter conditions and at least one squadron undergo all-year training in Alaska. This led to the establishment of Ladd Field at Fairbanks Alaska, in 1942.

The Cold Weather Test Detachment faced many difficult problems. Transportation was difficult and sometimes extremely hazardous. Weather, because of its uncertainty, played havoc with schedules.

These problems, plus the fact that much of the Air Force equipment at this time could not be used in temperatures below -20°C, made it clear that a more positive means of cold weather testing must be found.

In September 1943, the Army Air Force cold weather testing mission was assigned to what is now the Armament Division and through the efforts of Colonel H. A. Russell and Lt Col A. C. McKinley, plans were completed for constructing the Climatic Laboratory in May 1944.

Testing began at the Laboratory in May 1947. Army Air Force tests proved to be so successful that the Climatic Laboratory became a facility utilized by all agencies of the Department of Defense.

MISSION

The mission of the Climatic Laboratory is to provide global (surface) climatic environmental testing conditions in order that the United States Air Force and other Federal agencies may develop and test systems. The Laboratory has supported development and testing programs for many agencies, including Air Force, Coast Guard, Army, Marine Corps, National Aeronautics and Space Administration, National Weather Service, National Science Foundation, Air Force Geophysics Laboratory, and the Tennessee Valley Authority.

To accomplsih the mission of the Climatic Laboratory, the assigned personnel provide the following services:

a. Operate and maintain 11 testing facilities to provide environments.

b. Provide consultant engineering services for users of the facilities.

c. Design, manufacture, and install test support equipment.

d. Design, fabricate, and install instrumentation and data collecting systems.

In addition to the 11 major test chambers, 14 small test chambers are available at the Fuze Test Facility for environmental testing of explosive items such as bomb fuzes and ammunition.

MAIN TEST CHAMBER

The 93,000 cubic meter Main Chamber of the Climatic Laboratory is unique in that it is the largest and most complex climatic environmental test chamber in the world. Inside dimensions are 76.8m wide, 61.3m deep, 21.3m high in the center, with an appendant floor area 18.3 by 26 meters. In this chamber the Laboratory can produce special conditions such as rainfall to 40cm per hour, winds up to 160 knots, snow, solar radiation, humidity 10 to 95 percent, freezing rain, in-flight ice accretion of different types, and temperature extremes from -55°C to +75°C. Items varying in size from a tent to the C-5 Galaxy aircraft have been tested in the chamber.

Figure 1 shows the physical layout of the Main Chamber. Access to the chamber is through the main doors which open the full 76.8 meters of the chamber width. The other major opening into the chamber is the 4.6 meter high by 15.2 meter wide vertical lift door. All doors may be opened while the chamber temperature is being controlled.

Figure 2 shows the C-5 Galaxy transport aircraft undergoing testing at -40°C inside the Main Chamber. Because of the tremendous size of th- chamber, more than one item can be tested simultaneously. At one time during the C-5 test, two other aircraft, a F-100 and A-7, were in the chamber with the C-5.

AIR MAKE UP SYSTEM

A unique characteristic of the Main Chamber is an air make up system used to cool or heat air to the test temperature and inject this air into the chamber to allow the operation of jet engines during climatic tests. Up to 295 kilograms of air per second may be injected into the Main Chamber. Sustained engine runs at -55°C are possible without loss of chamber temperature.

Figure 3 shows a typical installation of a high performance aircraft in the Main Chamber. The F-16 aircraft is positioned on hydraulic jacks to afford a stable platform for the tie-down. Steel cables are then secured to the aircraft. This type of tie-down allows the aircraft to be run at full afterburner if necessary. Hot exhaust gases from the engine are taken out of the chamber through a duct fitted aginst the tailpipe of the F-16 and extending through the wall of the chamber.

Test personnel and test instrumentation are housed in the white buildings in the left portion of Figure 3. Shirt sleeve environments are maintained inside these rooms regardless of the chamber temperature.

ICING

Icing of a number of different types may be accomplished in the Main Chamber. Helicopter blade icing can be performed by a spray frame suspended vertically in front of the helicopter, or from a frame suspended horizontally above the aircraft. Icing tests may also be performed on a fixed wing aircraft. Liquid water contents between $.1g/m^3$ and $5g/m^3$ are routinely produced. Typically, a spray frame is positioned in front of the aircraft, and large fans are used to blow the spray cloud toward the aircraft. Cloud velocities are normally below 60 knots.

The amount of water to be injected into the flow field in order to obtain the desired liquid water content is determined analytically. Because the spray frame functions as a large evaporative cooler, the required water amount is the sum of the cloud liquid water content and the amount of spray that is being evaporated. The amount of water which is evaporating is calculated from a mathematical model developed at the laboratory. To verify analytical calculations, cloud liquid water content, particle size distribution, particle velocity distribution, cloud dew point, and chamber dew point are recorded and reduced by computer in real time during the actual icing test. Obviously, real time data reduction provides a powerful tool for spray frame control.

In situ measurements of particle size and velocity of polydispersed transparent spheres (rain drop and icing cloud particles) are made with a laser-based particle sizing interferometer. The system is non-intrusive, and as such, does not interfere with the phenomena being measured.

A sample probe is formed by the intersection of two equal intensity laser beams. The laser beams issue from a transmitter, and information is picked up by a receiver. The receiver may be located up to six meters from the transmitter anywhere in a 360° circle around the transmitter. Figure 4 shows a simplified optical configuration of the inteferometer. Particles passing through the probe region scatter light from the two beams. The period of the doppler shifted scattered light signal is measured and used with a set fringe spacing to determine the velocity. By also measuring the ratio of the oscillatory part of the signal to the overall pedestal amplitude, the size of the particle is determined. Liquid water content calculations are made in real time from the particle size and velocity information coupled with a known value of the sample probe size. All particle data is reduced in real time and displayed as a histogram of particle size and particle velocity and a table of numbers of particles versus size and velocity. Particles from five microns to five milimeters can be measured. A typical measurement range would be from eight microns to 80 microns. Each particle is measured, but particles are normally grouped into 50 to 60 bins separated by one to two microns for data presentation.

An icing spray frame can be seen in Figure 3 directly in front of the F-16. The small white boxes beneath the nose of the F-16 are the transmitter and the receiver of the interferometer.

Because of the ability to precisely control icing conditions, it is not unusual to be able to perform several completely different icing tests per day. Five different liquid water content and temperature combinations for helicopter rotor icing have been accomplished in one day.

TEST ITEM DATA COLLECTION

Test item data collection is provided through three instrumentation terminals each having 192 pairs of number 16 shielded instrument lines. One terminal is located on the south wall and the other two on the east and north walls.

Data lines are terminated in a patch panel located in the Laboratory computer room. A total of 400 data channels can be recorded at one time.

MISCELLANEOUS CAPABILITIES OF THE MAIN CHAMBER

Electrical Power: 440V, 3-Phase, 60Hz; 220V, 3-Phase, 60Hz; 120V, 1-Phase, 60Hz; 28VDC; 120/208V, 3-Phase, 400Hz.

Bomb Pit: The bomb pit, located in the center of the chamber, is 3m wide, 2.7m deep, and 25m long. Munitions can be dropped into this pit to evaluate release mechanisms on aircraft.

Gun Firing: A projectile trap permits gun firing of automatic weapons up to 30mm. Requests for gun firing support are evaluated for safety based upon rate of fire, velocity, caliber, and length of burst.

Snow: Snow can be made in the Main Chamber in varying densities and in quantities up to $6m^3$ per hour.

Control and Monitoring: An Observation room with three large nonfrosting observation windows is located in the south wall and a similar room on the north side of the chamber.

ENGINE AND EQUIPMENT TEST FACILITY

The Engine Test Facility is used for environmental qualification and external and internal icing of turbojet engines. This facility is supplied with refrigeration from the same system that supplies the Main Chamber. Therefore, due to its smaller size, the temperature range is greater (+70°C to -77°C), and the maximum cooling rate is much faster (15°C to -65°C in 7 1/2 hours). Permanently installed instrumentation includes equipment for recording temperatures from -70°C to +165°C, gauge lines for recording pressure at over 100 pickup points and a vibration indicating system. A high speed Digital Data Acquisition System can be used to write computer input tapes. In addition, oscillographs are available for recording transient data and a closed circuit television network can be used to facilitate testing. Specific details of the facility are:

Size: 39.6m long, 9.2m wide, 6.7m high. Usable floor space is limited to approximately 30m by 9.2m because of two fuel storage tanks located within the room to facilitate fuel cold/hot soak for engine testing. These fuel tanks may be removed if necessary.

Air Make Up: Same as Main Chamber.

Access to Facility: An electrically operated door, 9.2m by 7.6m opens the entire south wall of the room. An air lock, 6.7m square, is provided in the southeast corner of the room. Two personnel doors 1m by 2.1m are provided in the east wall.

Figure 5 shows the physical characteristics of the Engine and Equipment Test Facility.

Control and Monitoring: A control room with three nonfrosting observation windows is located in the east wall. Portable heated booths up to 3.7m by 3.7m may also be used inside the chamber for monitoring and control.

Engine Test Stand: The test stand is located at the south end of the room and is adaptable to aircraft engines of all types. The test stand may be removed from the facility if necessary.

Thrust Measurement: The engine thrust is transmitted through the test bed to load-sensing elements and then displayed on different instrumentation as required.

Icing Capabilities: Icing spray frames can be fabricated to meet the particular requirement for engine icing. Fuel icing is accomplished by controlling the liquid water content of the fuel, the temperature of the fuel, and the test space temperature to produce icing conditions.

Humidity: Same as Main Chamber.

Electrical Power: Same as Main Chamber.

Snow: Same as Main Chamber.

Altitude Capability: None.

SUN, WIND, RAIN, DUST FACILITY

The Sun, Wind, Rain, and Dust Facility is designed to facilitate rain, solar radiation, and dust testing. Rainfall, .03cm/hr to 40cm/hr, can be simulated over the entire 15.2m by 15.2m floor of the chamber. Winds up to 60 knots can be simulated, and these winds can be in conjunction with the rain or the sand and dust.

Size: The inside dimensions of the chamber are 15.2m by 15.2m by 9m.

Temperature Ranges: 21°C to 70°C.

Humidity Control: From 95% at 71°C to 20% at 20°C.

Solar Radiation: 46.5m^2 of solar radiation up to 1300 watts/m^2.

Access: Four vertical lift doors, 4.6m high by 4.9m wide, are provided in the chamber. The room also has two personnel doors. An observation window and several ports are located in the control room for viewing and access to instrumentation.

Wind Machines: Up to three wind machines can be used at one time. Each machine can produce volocities from 5 to 60 knots.

Sand and Dust: Sand and dust are introduced into the chamber from outside sources and are maintained in suspension by the wind machines. The dust is filtered out of the air, and the air is then conditioned and returned to the chamber where more dust is added to it, thus assuring precise temperature and humidity control during tests.

Electrical Power: 480C, 3-Phase, 60Hz; 120/208V, 3-Phase, 60Hz.

Figure 6 shows the physical characteristics of the Sun, Wind, Rain, and Dust Test Chamber.

PHYSIOLOGICAL STRATO CHAMBER

The Physiological Strato Chamber is the Laboratory's only chamber in which altitude as well as temperature can be simulated. Temperatures to -70°C and simulated altitudes as high as 25,000m have been attained. With pre-cooling, the temperature in this chamber can be reduced from +20°C to -56°C and, at the same time, pressure lowered to 87mm Hg (15,000m) in 12 minutes.

A lock is provided so that personnel may enter and leave the chamber with minimum interference to test conditions. In addition, instrumentation panels are provided to facilitate the recording and control of test data.

The chamber is constructed of welded steel, insulated with 13 sheets of reflective metal insulation in the chamber and 7 sheets in the lock. The chamber is constructed to withstand pressures from zero absolute to one atmosphere.

Size: The chamber is 4.1m long, 2.9m wide, and 2.2m high. A lock adjacent to the chamber is 3.05m wide and 1.32m long.

Temperature Range: +60°C to -70°C.

Communications: An aircraft-type intercommunication system is provided for communication between the outside and the chamber. Six observation windows are provided.

Access: Door size for both lock and chamber is .65m by 1.1m.

Electrical Power: Same as All Weather Room.

SALT TEST CHAMBER

The Salt Test Chamber is designed to provide salt fog and salt spray conditions in accordance with applicable testing standards.

Size: Interior dimensions are 17m long, 4.9m wide and 4.9m high.

Temperature Range: 10°C to 77°C; relative humidity from 30% to saturated.

Access: End doors 4.6m wide and 4.6m high; one personnel door .9m by 2m.

ALL WEATHER ROOM

One of the most versatile facilities in the Climatic Laboratory is the All Weather Room. In this chamber, arctic to jungle conditions can be produced on an overnight basis. Rainstorms up to 40cm of rainfall per hour can be created. Sand and dust storms as well as snow can be produced as the test requirements demand. Its specifications are:

Size: 13.2m by 6.7m by 4.9m.

Temperature Range: +77°C to -62°C. Maximum cooling rate is from +15°C to -62°C in 24 hours.

Humidity Control: From 5% to 95% with regulation to within ±2%.

Access: A vertical lift door, 4.3m wide by 4.7m high, opens to an outdoor ramp on the east end of the room. A personnel door, .8m by 2m, is provided on the south wall. Small doors, ports and windows are located within the walls for viewing and access to instrumentation and power leads.

Sand and Dust: Sand and dust are introduced into the chamber from outside sources and are maintained in suspension by wind machines.

Electrical Power: 440V, 3-Phase, 60Hz; 120/208V, 3-Phase, 60Hz; 120/208V, 3-Phase, 400Hz; 28VDC.

WHO MAY USE THE CLIMATIC LABORATORY

The Climatic Laboratory is a technical facility operated for the USAF by the Armament Division (AD) and is available for use by all Department of Defense agencies, commercial non-Government contractors with Federal executive agency sponsors, and foreign allies. Inquiries concerning the technical capabilities of the Laboratory should be addressed to: Headquarters, AFSC/DLXL, Andrews AFB MD 20334.

Testing periods for the Main Chamber must be scheduled well in advance; however, they are flexible where possible in order to provide maximum chamber utilization. Testing periods for the smaller auxiliary chambers are arranged according to the specific requirements of the testing agency.

It is suggested that, whenever practicable, an advance planning meeting be held at AD, with representatives from the sponsoring agency and the testing agency or contractor in attendance. This meeting should be planned with the following purposes in mind:

a. Afford all test personnel the opportunity to review, define, clarify, and reach agreement on the test support requirements. It should be possible at this meeting to identify those requirements which can be satisfied by resources at AD, those which will require procurement action by AD, and those which the sponsoring agency may be required to provide.

b. Provide prospective users the opportunity to become familiar with the details of operation and the capabilities of the Laboratory.

No action can be taken on requests for tests which are received directly from industry; however, contractors holding contracts with any agency of the Department of Defense may conduct tests in the facility when sponsored by the DoD agency with whom they are under contract.

Upon receipt of a test request from the DoD testing agency or the DoD sponsoring agency, AD will review the support requirements and will advise all concerned of AD acceptance or rejection, and of any recommended changes to the test procedures or support requirements. No test, however, will be supported without an official letter or message from the responsible DoD agency requesting support of the test and approving expenditure of necessary funds. After receiving an acceptance notice from AD, the requesting agency must notify AD of its firm intent to conduct the test. Upon receipt of this notice, AD will schedule the test.

For emergency tests, AD will immediately review each request for test support on its own merits, and will consider providing support on an "as available" basis, making every effort to cooperate with the responsible DoD agency.

CONCLUSIONS

The large size of the Climatic Laboratory chambers allows complete systems to be tested under a large number of climatic conditions. Not only can the machine be tested, but also the man-machine interface.

Out of years of testing experience has grown a philosophy equally applicable to all services and agencies of the Department of Defense. This philosophy embraces the concept that environmental testing of all systems is a vital necessity in the improvement of overall system reliability. Through both simulated and natural environmental testing we are afforded an opportunity to continually study and understand the mechanics of failure and so approach total reliability. Because of the flexibility and relative low cost of simulated testing as compared to field testing, it is evident that the Climatic Laboratory is one of the most outstanding examples of money saving and methods of improvement within the Department of Defense.

REFERENCES

1. Toliver, Richard D. "Instant Weather," Maintenance Magazine, Summer 1978.

2. "Unique Features of the McKinley Climatic Laboratory," The Journal of Environmental Sciences, July 1978.

3. "The Ability of the Climatic Laboratory to Produce a Wide Variety of Climatic Conditions," Summary Report, Rotary Wing Icing Symposium, USAAEFA Project No. 74-77.

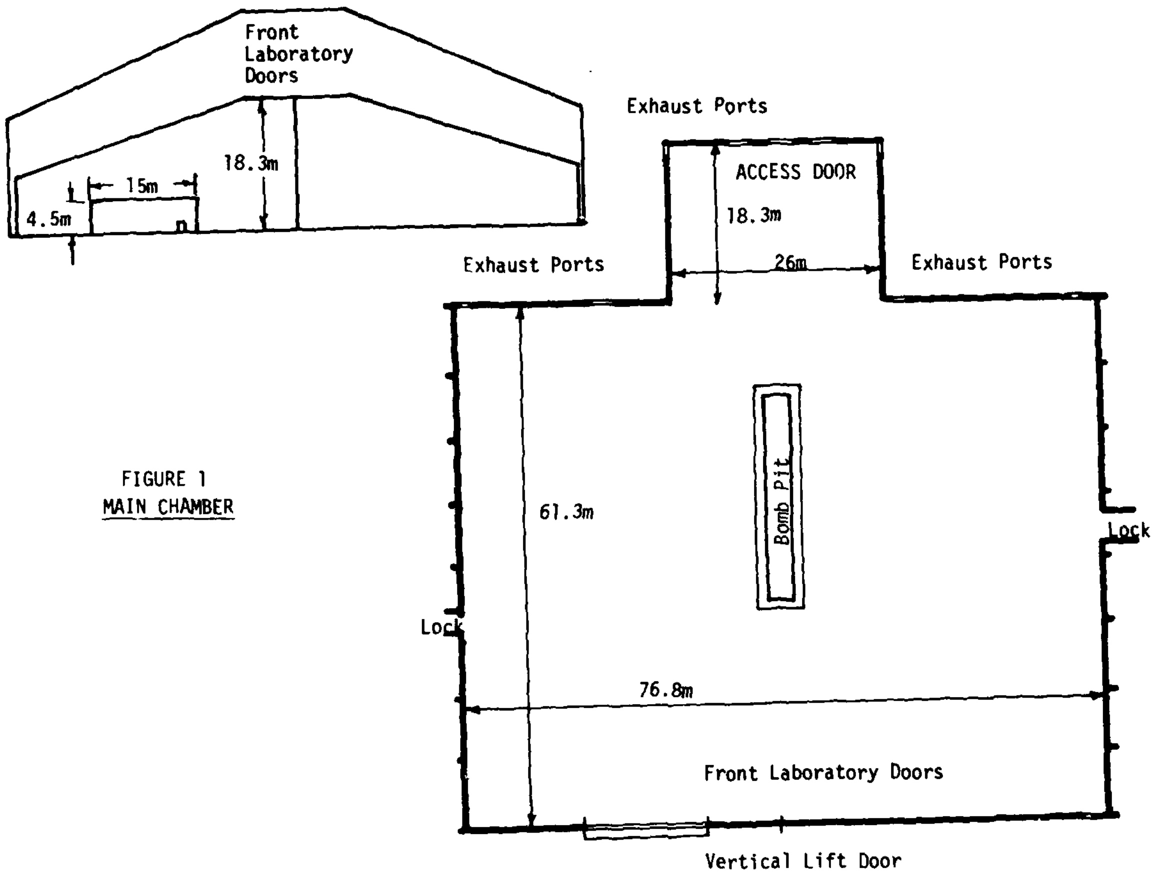


FIGURE 1
MAIN CHAMBER

FIGURE 2   C-5A AIRCRAFT IN MAIN CHAMBER

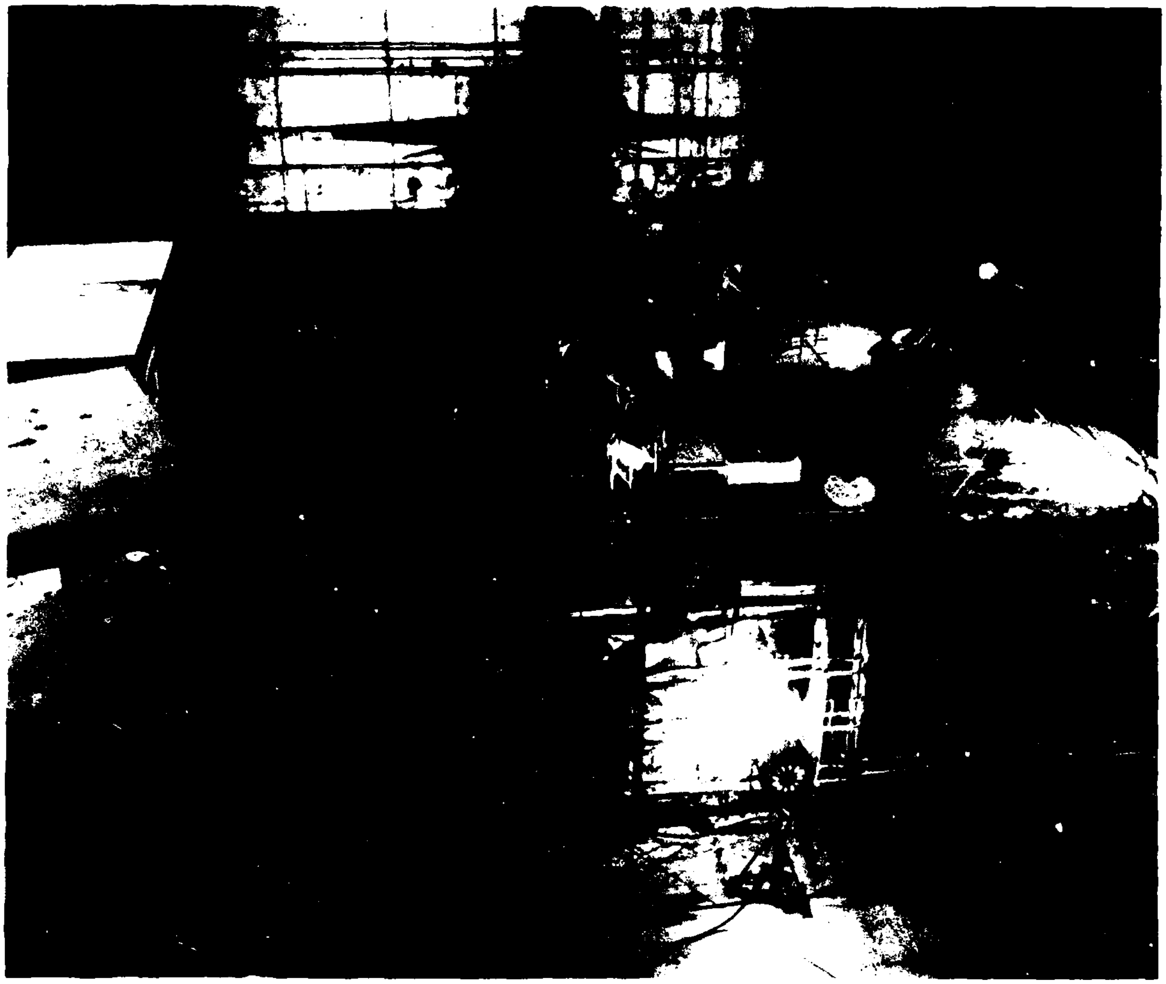
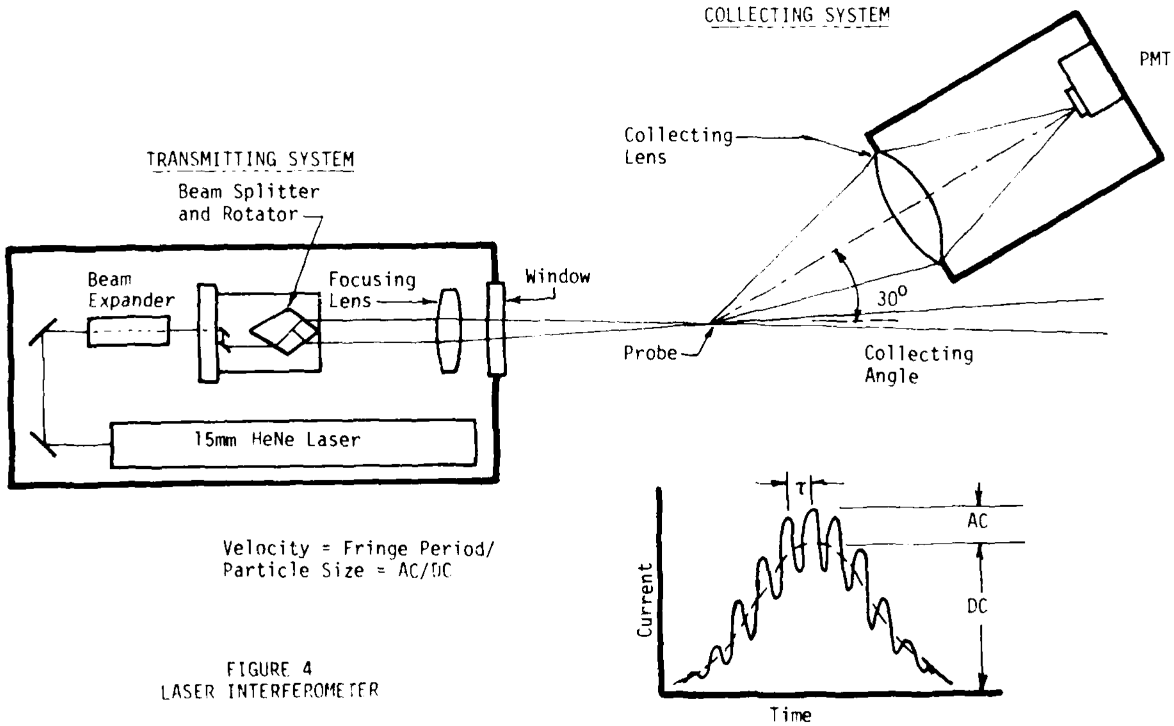


FIGURE 3 F-16 IN MAIN CHAMBER



Velocity = Fringe Period/
Particle Size = AC/DC

FIGURE 4
LASER INTERFEROMETER

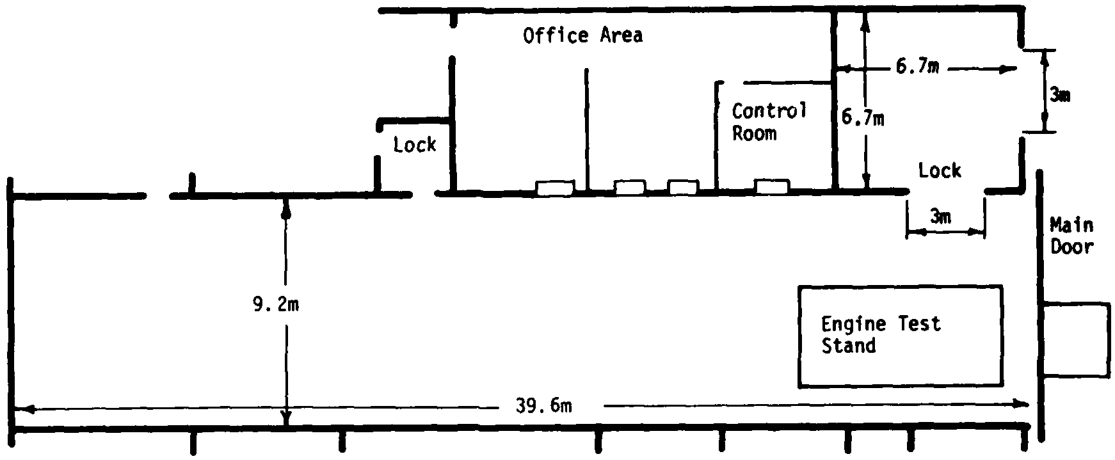


FIGURE 5
ENGINE AND EQUIPMENT TEST FACILITY

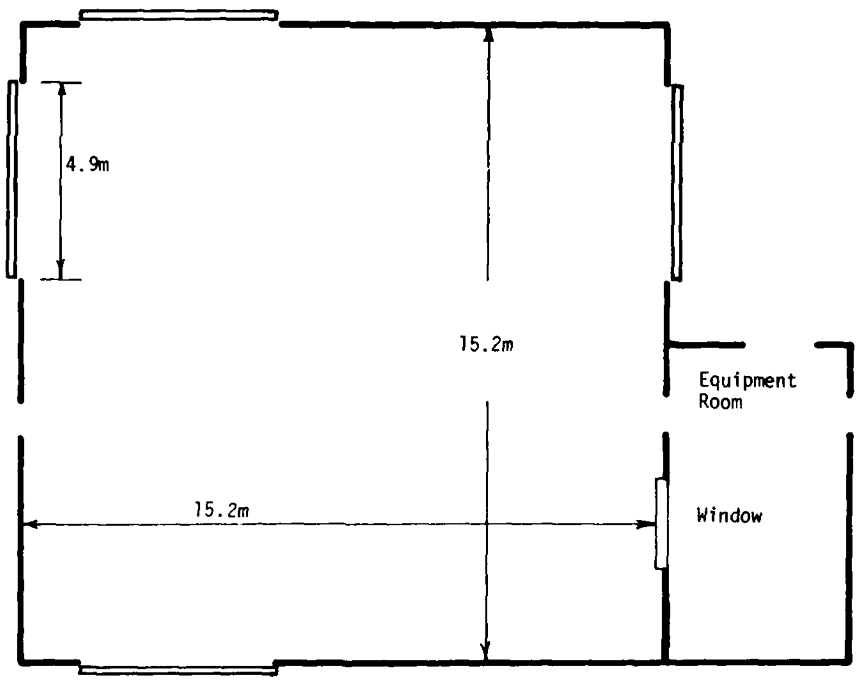


FIGURE 6
SUN, WIND, RAIN, DUST FACILITY

DEVELOPMENT FOR HELICOPTER FLIGHT IN ICING CONDITIONS

by

David Gibbings C. Eng., M.R.Ae.S.
Helicopter Icing Trials Manager
Westland Helicopters Ltd.,
Yeovil, Somerset, England, BA20 2YB.

SUMMARY

Helicopters have reached a stage in their development where prolonged flight in icing conditions has become an operational necessity. Over the last decade technology proving programmes have been undertaken in Europe and North America and all weather operations are now a practical proposition.

The requirement for airframe, intake and rotor deicing systems and the instrumentation considered necessary for safe flight development are discussed. The problems involved in carrying out icing and snow flight development as part of a full prototype development programme are considered and some of the favoured icing simulation test methods are covered.

The conclusion is devoted to a description of the way in which an icing release may be substantiated including a statement on the current state of the art.

DEVELOPMENT FOR HELICOPTER FLIGHT IN ICING CONDITIONS

1. INTRODUCTION

Since its emergence as a practical flying machine the helicopter has evolved over the past thirty five years an established place in the aviation scene. During this time they have been regarded with some condescension by the fixed wing fraternity as a mechanical novelty whose rightful place is to provide support for real aeroplanes, indeed the next best thing to flying.

The situation is changing and the toy has grown up. The modern helicopter can boast Automatic Flight Control systems and avionics sufficient to gain entry into the most hallowed regions of restricted airspace. The armed helicopter has a terrifying capability of destruction and has changed the whole concept of anti submarine warfare. Helicopters dominate the rescue and oil rig support scene and are expected to maintain their role regardless of weather.

It follows that requirements and specifications for the next generation of helicopters are beginning to contain some very positive statements regarding sustained flight in icing and it will therefore come as no surprise that helicopter manufacturers in Europe and the U.S.A. are all actively working on icing development programmes.

2. THE PROBLEM

Icing has always been a bogey man to the helicopter operator, helicopter operations are rarely above 10,000 ft. A region where icing is most likely to occur.

The term icing covers a range of climatic conditions. Firstly our old enemy natural icing, this takes the form of supercooled water droplets in suspension in sub zero cloud, which freeze on contact with any object passing through. Ice subsequently builds up on all forward facing surfaces including rotating blades and controls. With the power available these days fuselage accretion in its own right does not cause a great deal of embarrassment but it can effect the efficiency of aerials, block vents and air inlets and in the process of shedding can find its way into intakes and tail rotor.

Ranging as it does from hover to fast forward flight in climb or descent and accompanied by vibration, ice shedding does present a helicopter problem. The trajectory of shed ice is less easy to predict and repositioning the offending protuberance will often only transfer the problem to another flight regime. Assymetric ice shedding from rotor blades will result not only in a very unpleasant ride but severe stress penalties and of course engine intake icing is a major problem.

The main rotor has distinctive problems of its own. The build up of ice on the leading edge changes the pitching moment of the blade resulting in a massive increase in control loads, continued excursions into icing would subject the controls to oscillatory stresses far beyond those upon which the component lives were calculated and could eventually lead to catastrophic failure. There is of course an even more obvious effect, change in blade profile brought about by ice accretion brings about a loss of efficiency which will eventually force the helicopter down. (Fig. 1)

Snow does not generally lead to heavy ice accretion but it does tend to build up in all sorts of awkward places and if very fine gets into cracks and under fairings. It then melts and runs out as water to freeze elsewhere. (Fig. 2) Snow also effects the helicopter in another way. Flying in the near hover close to the ground it will re-circulate to envelop the aircraft totally, obscuring the ground just when you need to know where it is. (Fig. 3) More than one helicopter has been brought back to earth rather forceably having ingested a blanket of snow from the structure in front of the engine which a careless crew have failed to remove before starting. Because of their mode of operation clear vision is probably even more critical for helicopters and the maintenance of this presents its own problems. A heavily iced up helicopter coming into the hover can shed large chunks of ice from the rotor at near lethal velocities, something which the next generation of shipborne operators will have to think about. Having got the thing on the deck there will then be the problem of removing residual ice during a quick turn around

3. ROTOR DE ICING

Keeping the rotors clear of ice is more specifically a helicopter problem which has to be solved if prolonged flight in all weathers is to be entertained and it is in this area that the main effort has been concentrated.

Over the last decade European and U.S. projects have been developed to varying degrees utilising cyclic electrical heating on the blade leading edges whereby heater mats are energised in turn. (Fig. 4) Centrifugal force and vibration tend to work in your favour so that once the heat loosens the bond the ice does fly off. In fact some helicopter rotors do self shed under certain conditions without the help of a system, but none to the extent that it could form the basis of certification and it does vary with the wide range of conditions and from type to type.

The amount of electrical power necessary to maintain anti icing is far too high to be seriously considered and all the systems under development or in service utilise some form of cyclic programming. The demand on electrical power can be further reduced in rotors with even numbers of blades by deicing blades alternately as diametrically opposed pairs. Assymetric shedding of ice results in severe vibration and high stresses.

The areas of accretion on leading edges have been well established and credit must be given for the early work carried out in this field by the National Research Council of Canada. From this and subsequent flight work the requirements for leading edge mat coverage have evolved.

Accretion on the leading edge will result in a rapid rise in control loads and in severe conditions this will force the helicopter to vacate the condition in minutes. (Fig. 5) Regular cycling of leading edge mats has to be maintained to sustain flight. The problem with any heat generating system is that having melted the ice the water created will run back to freeze on the first available cold surface. This results in a build up of ice aft of the leading edge, progressively reducing the rotor efficiency. The tail rotor and stabilizer also require protection.

Test programmes run by the U.K. to date have been concerned with technology proving to confirm feasibility over a wide temperature range and to provide a good base from which confident design decisions regarding mat coverage. ON times, cyclic sequences and heat intensities can be made.

4. INSTRUMENTATION OF DEVELOPMENT AIRCRAFT

Icing is clearly one of the high risk areas in any test programme and should be approached with caution. A comprehensive instrumentation fit is an important factor in the rate at which progress can be made and is one of the main assurances for safety. (Fig. 6)

The aircraft must carry reliable instruments to provide atmospheric information such as temperature and liquid water content.

Although flight may still be maintained with heavy ice accretion, this may result in very high stresses on blades and controls not necessarily reflected by aircraft behaviour or handling. A comprehensive strain gauge fit is consequently necessary and the results will have to be monitored throughout the programme to ensure that vital component lives are not at risk.

A full set of performance data is essential and certain key stresses and engine intake condition will need to be available as visual presentation to the crew to provide sufficient data so that the decision to stay or retreat in good order can be made before the situation gets out of hand.

Extensive use has been made of video recording on the U.K. aircraft to monitor a sensitive engine air intake but also to add visual and voice recording capability to provide a back up for the primary recorded data.

Accurate temperature measurement and liquid water content are vital in order to ensure that the crew have a reliable readout of atmospheric conditions with which to carry out the ice hunting phase of the operation and all of this has to be on record for later analysis.

A technique which has been applied extensively in the Wessex is the use of accurate lag angle measurement to determine a comparative torque contribution for individual blades. In order to use these data it is necessary to obtain a good clear air calibration of blade datum lag angles over the speed range. This used with cameras and other data is very useful as a test technique we refer to as 2 x 2 whereby diametrically opposite pairs of blades can be subjected to different cyclic sequences and compared when flying under the same icing conditions.

An important feature is to include the ability to change the sequence in which heater mats are energised allowing a wide range of cyclic switching options to be tested. If this can be accomplished easily it is possible to take full advantage of a spell of icing weather covering a wide range of experiments.

4.1 Camera

Vital to any meaningful rotor icing assessment is the ability to visualise what is going on up there. In the U.K. since the early 70's a considerable amount of effort has been devoted to the development of effective camera systems and has evolved from 16 mm cine to the current use of 70 mm colour using the F 95 low level reconnaissance camera. (Fig. 7)

Two distinctly different systems have resulted to provide coverage of the upper and lower blade surfaces.

The upper surface is covered by a camera viewing vertically upwards into a four way mirror. The result is a single picture showing all four blades. (Fig. 8)

Photographing the under surface requires a much more complex system whereby the passage of a single blade is sensed by a doppler radar and this with appropriate time delay triggers a 70 mm camera and flash mounted on the tail boom. Four individual pictures are required to cover a four bladed rotor and a control system allows this camera to be synchronised with the main rotor head camera. (Fig. 9)

Both cameras are referenced to a central time base and time appears in digital form on each frame to within 0·1 seconds allowing accurate cross reference to trace and video recordings.

The systems were originally conceived by the Aircraft and Armament Experimental Establishment at Boscombe Down and have been in use very successfully for the past three years and have played a key part in the U.K. development.

All the above require comprehensive and rapid on site analysis so that full advantage can be made of good icing weather.

4·2 Radio Navigation Equipment

Icing work requires that the aircraft will spend most of its time in instrument flight and it is very important that a comprehensive installation of radio and navigational equipment be included. The aircraft will require dual communication radios, ADF, VOR, ILS, Secondary Surveillance radar.

5. THE ICING TRIALS ENVIRONMENT

A considerable amount of discussion has taken place concerning the choice of an icing trials venue. Icing trials programmers have spent a great deal of time and effort studying climatic statistics to find the ideal site. Icing has always been difficult to forecast and the occurences of icing from one winter to the next are erratic and impossible to predict.

It would be difficult enough even if climate were the only consideration but this unfortunately cannot be so and an icing trials manager must consider a much wider permutation.

(a) Climate must, of course, be the primary factor but it does not follow that good natural icing will produce equally good snow conditions.

(b) Air traffic - it is essential to have good air traffic surveillance and preferably a section of airspace dedicated to icing flying. A co-operative air traffic control organisation can make a major contribution to the success of an icing trial.

(c) Terrain - it is very important to be able to operate over reasonably level terrain with a relatively low population density.

(d) Safety Services - good S.A.R. support is vital.

(e) Domestic and Servicing Facilities - good hangarage with at least some workshop facilities are required. The site needs to be accessible and within reach of a regular air cargo service. Accommodation for analysis facilities and domestic accommodation for the trials team throughout a long winter trial have to be considered. The site must also be secure.

We do not live in a perfect world and all the above are rarely possible in a single site. Failure to recognise any aspect will reduce the flying rate and the ability to respond to suitable weather opportunities.

Most icing trials in the Western world take place in the Northern United States, Canada or Scandinavia. Having found a suitable venue good sense says that the environment should be approached with caution. The ideal, which is rarely available, is a consistent layer of icing cloud up to 1,500 metres with a cloud base 300 metres A.G.L. and 300 metres forward visibility. Given this situation it is possible with some confidence to take an icing encounter to the point where the aircraft is forced out of the condition with the knowledge that a controlled precautionary landing is an available option.

6. SIMULATION OF ICING AS AN AID TO DEVELOPMENT

Confronted with the need to develop a new helicopter with a good icing flight capability the designer and development engineers are faced with the problem whereby climatic trials generally come late in the programme and invariably bring out a number of problems concerned simply with the hitherto untried aspects of temperature, water ingress, humidity etc. accompanied with the need for caution in the early stages, the result of which is liable to be insufficient answers too late in the programme.

There is no easy answer to this dilemma. It is essential that in order to achieve a productive icing trial the aircraft must have good component life margins, a well explored flight envelope and that all systems are to a reasonably advanced state of development.

A considerable amount of ingenuity has gone into simulation of icing and a number of facilities are now available in which advance testing can be undertaken.

6·1 Icing Tunnels

Facilities whereby icing can be simulated in a wind tunnel are available in all sizes and it is consequently possible to install whole airframe rigs in large icing tunnels and establish airframe and intake accretion areas to determine the heating required for de-icing and to give some indication of ice shed trajectories. The largest of these facilities in the U.K. is at Cell 3 West in the National Gas Turbine Establishment and this runs a very full programme on Fixed Wing, Helicopter, Engine and Research projects.

There are numerous smaller facilities available in which icing instruments and component icing can be assessed.

6·2 The N.R.C. Spray Rig

This installation has been in operation for a considerable time and has a well deserved place in helicopter icing history. (Fig. 10)

The disadvantage of this facility is that it will only allow assessment in the hover and the arguments concerning droplet size and cloud consistency are still waging. Situated as it is in Canada, from the U.K. viewpoint, a spray rig test is as difficult to mount as an icing trial.

6·3 In Flight Spray System

Extensive use is made in the U.S. of Inflight Spray aircraft. The Helicopter In Flight Spray System (H.I.S.S.) using a Chinook tanker has featured extensively in U.S. programmes.

The advantage is that testing can be undertaken in the comparative safety of clear air and provides a useful supplement to a natural icing trial in that it allows work to continue during the interminable VMC days which seem to prevail whenever one undertakes an icing trial. There are the inevitable arguments concerning droplet size and cloud distribution and an H.I.S.S. test involves a fair degree of co-ordination to control chase aircraft, test aircraft, tanker and photographic coverage. (Fig. 11)

6.4 Shed Wax Trajectory Tests

One of the worries concerning airframe icing is the trajectory of the ice shed during melt off. Considerable success has been achieved by the use of a simple mousetrap release which can be mounted on vulnerable points and which release prepared wax blocks when triggered.

Such tests can be carried out regardless of temperature and because there is no risk of icing, engine intakes can be protected by grilles which also show witness marks of ice strikes. (Fig. 12)

The system was used to good effect in Lynx when 663 wax blocks were released covering the flight envelope and a wide range of aerial, weapon and role equipment sites were assessed.

A well planned programme will show full use of rig and simulation facilities at early stages to support the design development phase but any simulation of the complex natural phenomenon which forms icing cloud can only be taken as a guide and can never be accepted as the full answer. Eventually one has to face up to taking the aircraft into the real thing and any icing programme will culminate in a number of icing trials as a final demonstration.

7. DEVELOPMENT OF FLIGHT IN ICING

The development programme to achieve an effective flight in icing release is by its nature a protracted process and in a new project with new airframe, transmission and power plants, the design and development process needs to be properly integrated from the project stage.

The fundamental problems of getting a new project fully developed and into service with a satisfactory flight envelope and component lives are more than enough to undertake and all the basic issues of development have their influence on the progress of the ice protection systems development.

Let us now consider a helicopter for which full ice protection is a stated requirement and which must maintain its role regardless of weather.

7·1 Intake

The fact that our aircraft must have a flight in icing capability will have an influence on the intake lines and the first line of defence for engine protection will lie in the geometry and siting of the intake. The designer may choose to place them well forward or accept the loss of ram efficiency and adopt sideways facing.

Choice of engine will also have its influence. Engine designers are attempting to build in a degree of survivability from ingestion and many engines now include F.O.D. separators in their design. All the above will offer a degree of assurance but our all weather aircraft will almost certainly be given some specific ice protection.

The tortuous process of sub contractor selection will then take place and design of the chosen system will proceed. The sub contractor will be called upon to show his own programme of work to demonstrate compliance with environmental, climatic, vibration, structural integrity, E.M.C. and a wide range of criteria which a wise prime contractor will lay down in his requirement.

The main pre flight intake development will probably take place in a number of sessions using a major icing tunnel facility with an airframe rig. From these tests it will be possible to decide heat distribution, sensor positions, area coverage and power requirements for introduction into the prototype aircraft.

7·2 Rotor Protection

The pre flight phase of rotor protection will follow similar lines but meaningful testing of the full system is much more reliant on flight testing. The introduction of heater mats into the blades can be expected to have an influence on the overall structural integrity and such issues as thermal cycling and mat adhesion must be tested early in the programme. Icing tunnel testing to determine the icing sensitivity of the chosen aerofoil section may also be included to finalise mat coverage.

It will be vital that the sub contractor can demonstrate that the overall integrity of the blades is not down graded by the presence of heater mats and the control system will have to satisfy a wide range of requirements. Slip ring assemblies are an area of technical risk and bench or rig tests to demonstrate reliability will be included.

The main and tail rotor blade fatigue programme will also be under way at this time and all fatigue specimens will include dummy mats. The presence of mats and wiring can have an influence on the blades' characteristics under lightening strike and tests will be carried out to clear this aspect.

7.3 Airframe

Resistance to severe ice accretion will be given full consideration during the design stage. The siting of aerials will be decided with this in mind. Practical assessment of these issues will take place in the major icing tunnel tests probably in parallel with the intakes work and will include such items as heated windscreens and wipers.

The need for some antenna to be de-iced in order to function will be determined at an early stage.

7.4 *Systems*

It is inevitable that ice protection systems will make demands on electrical power and it is highly likely that a dedicated power supply will be provided. The icing systems will be included in electrical rig testing which will include assessment of power transients, switching failure simulation and E.M.C.

During these early stages it will also be necessary to ensure that adequate instrumentation such as strain gauges, temperature sensors are included in intakes and blades planned for development aircraft and decisions must be made regarding which prototypes will carry full systems and to what standard.

All the above activity will lead to a high degree of assurance that design has been properly accomplished and can be expected to lead to Experimental Flight Approval for the Flight Development Programme to proceed.

8. FLIGHT DEVELOPMENT

Confronted with the task of developing a new aircraft with all the complexities of flight envelope, vibration, stress, handling, mechanical, testing and A.F.C.S. to consider, it cannot be expected that ice protection will receive a great deal of attention during the initial phase of flying, it is even probable that the first machines will not include any of the equipment.

Once an aircraft with complete ice protection systems is flying, time will be devoted to functioning, EMC and Failure Simulations. During the main flight test programme provision will also be made for some wax shedding tests to provide assurance for the main icing trials work.

Dealing as we are with a U.K. based programme some opportunities for limited excursions into icing can be expected and will be cautiously undertaken.

One of the main problems is that production decisions have to be made at an early stage in the programme and nearly always in advance of the first major icing trials and the best that one can hope for is a brief icing assessment during this phase with minimal opportunity to make a short (say 6 week) excursion to an icing venue but this cannot occur too early in the programme simply because an overseas trial with an unproven aircraft is very difficult to support and component lives are still cautiously low at this stage.

By the time the first major development Icing Trial takes place planning will already have taken place to introduce ice protection systems in the early production aircraft which are intended for certification testing and Service Trials.

It can be expected that the major issues for icing and snow flying will be investigated during the first trial, most icing practitioners agree that it will be necessary to keep the aircraft at an icing venue for the full winter. Given the icing weather, it should be possible to formulate a cautious experimental icing release and to make provision to introduce the essential changes which may arise from the trials into the production machines as a retrofit.

A second development trial should consolidate this work and make it possible to make a qualified statement of conformity with the requirement and to pinpoint areas where modification and further development are necessary. All the above work will have been under the leadership of the manufacturer with active involvement by the certification agency.

The first trial with a production aircraft will represent the main contribution towards a full flight release and will heavily involve the certification agency. The aircraft will embark upon the trials with the highest achieveable standard of ice protection and the ideal is that it will dramatically demonstrate complete conformity with the requirement.

With three major icing trials completed, the capability and potential of the ice protection will be very much clearer and it should be possible to consolidate the release by close monitoring of service experience. (Fig. 13)

9. THE WAY AHEAD

Numerous technology proving projects have been undertaken over the past five years and there are many well substantiated claims to success. The result is that helicopters with ice protection are currently under development or in production.

There are variations of opinion regarding the way in which certification or release is underwritten, whether by complex simulation or dramatic demonstration in the real environment, costly icing trials with specialised test aircraft or through cautious monitored extension through service experience. There are now few who doubt that success can and will be achieved.

Helicopter flight in icing has reached a stage of development similar to that through which fixed wing aircraft passed during World War 2. Unlike our fixed wing counterpart the helicopter will be forced to remain at low level to rescue the unfortunate or the irresponsible, to defend the nations lawful needs, provide the support necessary to ensure continued flow of oil and to carry the impatient captains of industry quickly but surely to their destination, all regardless of weather.

The technology is available but it will not come cheaply and must be hurried with care but the time cannot be far away when flying could well be considered the next best thing to a good all weather helicopter. (Fig. 14)

Disclaimer

The views expressed in this paper are those of the author and do not necessarily represent the views of Westland Helicopters Limited.

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SUMMARY OF ROTOR THERMAL DE-ICING PROJECTS

Bell 47	Research (N.R.C.)	Canada
CH 46/CH 113	Development	Canada
SH 3D/CH 124	Development/Service Trials	Canada
Alouette	Research	France
Super Frelon	Technology Proving	France
Puma/Super Puma	Production	France
Bolkov Bo 105	Technology Proving	Germany
Bolkov Bk 117	Proposed Development	Germany/Japan
Kawasaki Kv 107	Development	Japan/U.S.
Sycamore	Research	U.K.
Wessex 5	Technology Proving	U.K.
W.G. 34	Design Stage for Production	U.K.
CH 47 Chinook	Development/Production	U.S./U.K.
Bell UH-1	Technology Proving	U.S.
HH 2D	Experimental	U.S.
H 34	Research	U.S.
UH 60A	Development/Production	U.S.
UH 61	Development	U.S.
YAH 64	Development/Production	U.S.
Mil 8	Production	U.S.S.R.
Hind	Production	U.S.S.R.

Note: The true status of U.S.S.R. projects is unconfirmed.

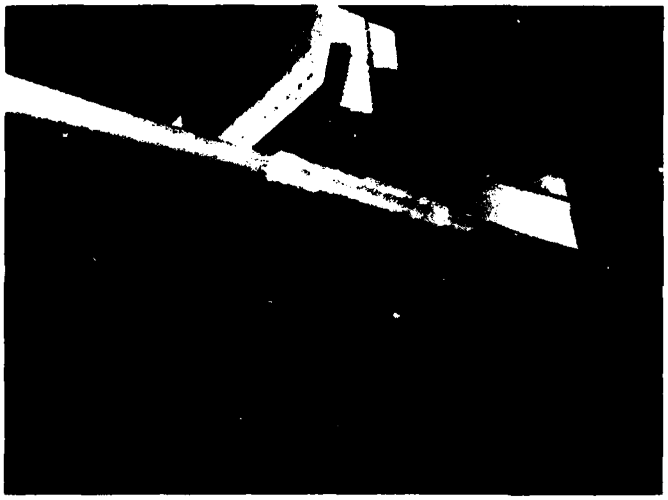


Figure 1.

ICE FORMATION ON UNPROTECTED ROTOR BLADE

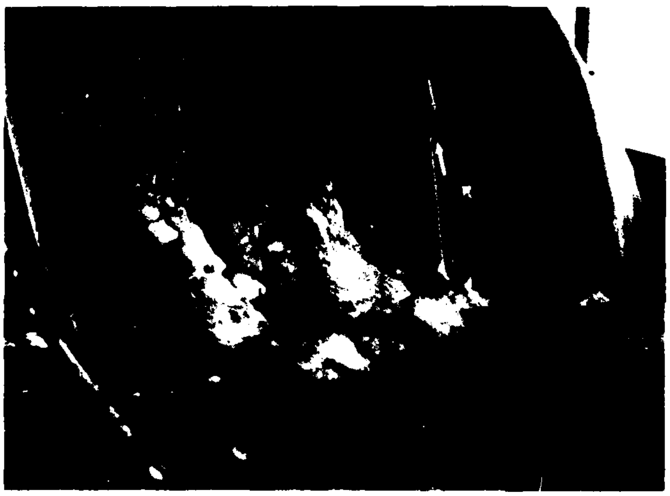


Figure 2.

REFROZEN SNOW AFTER RUN OUT FROM GEARBOX AREA

Figure 3.

FLIGHT IN RECIRCULATING SNOW

Figure 4.

DEVELOPMENT OF HEATED ROTOR BLADE DEICING SYSTEM IN A WESSEX HELICOPTER

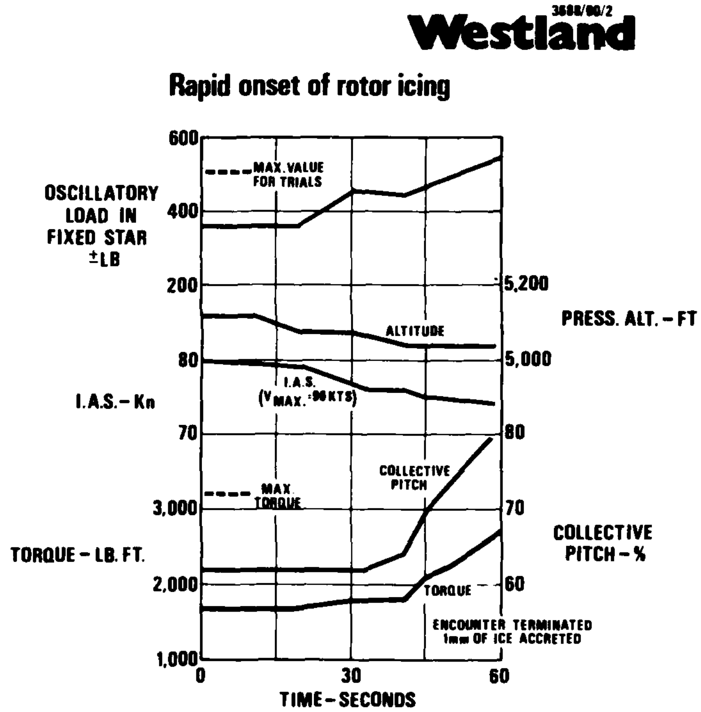


Figure 5.

RAPID ONSET OF ROTOR ICING

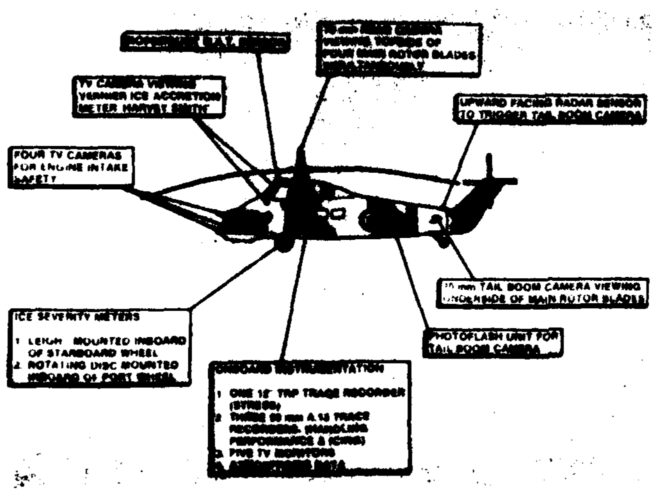


Figure 6.

AN ICING INSTRUMENTATION PACKAGE

Figure 7. WESTLAND ICING TRIAL CAMERA INSTALLATION

Figure 8. 70mm MAIN ROTOR HEAD CAMERA PHOTOGRAPH

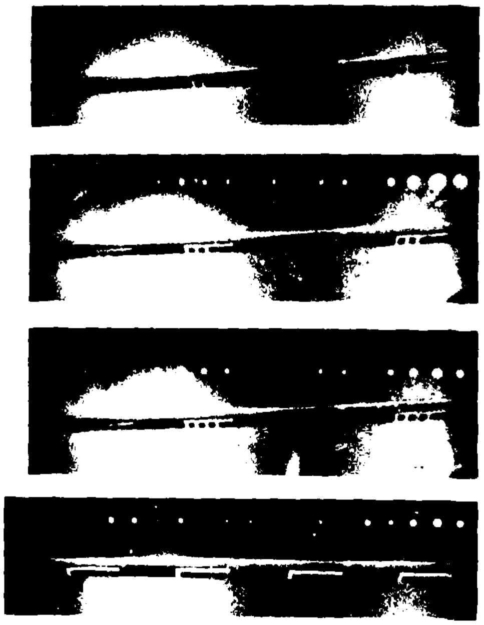


Figure 9.
70mm TAIL CAMERA PHOTOGRAPH
SHOWING THE BLADE UNDERSIDE

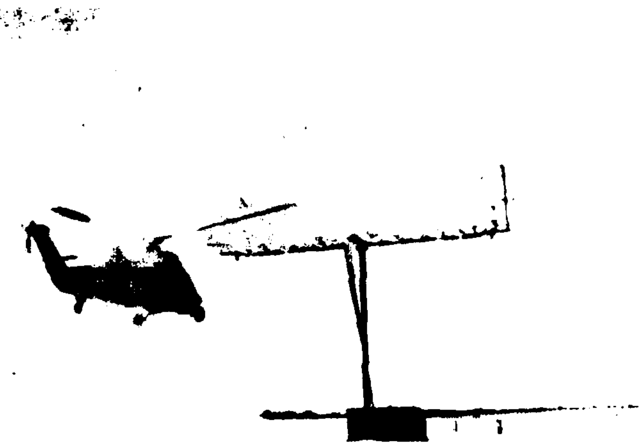


Figure 10. THE NRC SPRAY RIG

Figure 11.
HELICOPTER IN-FLIGHT
SPRAY SYSTEM (HISS)

Figure 12.
WAX SHEDDING EQUIPMENT
TO SIMULATE MELT OFF

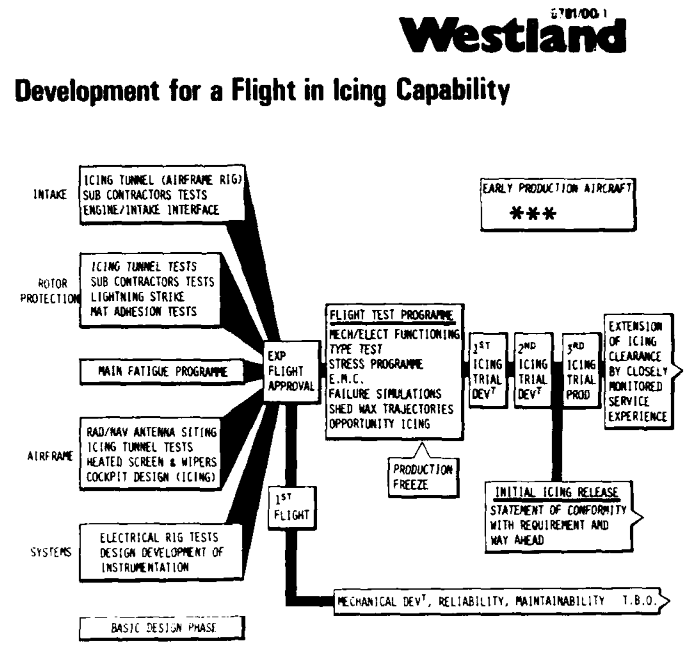


Figure 13. DEVELOPMENT FOR A FLIGHT IN ICING CAPABILITY

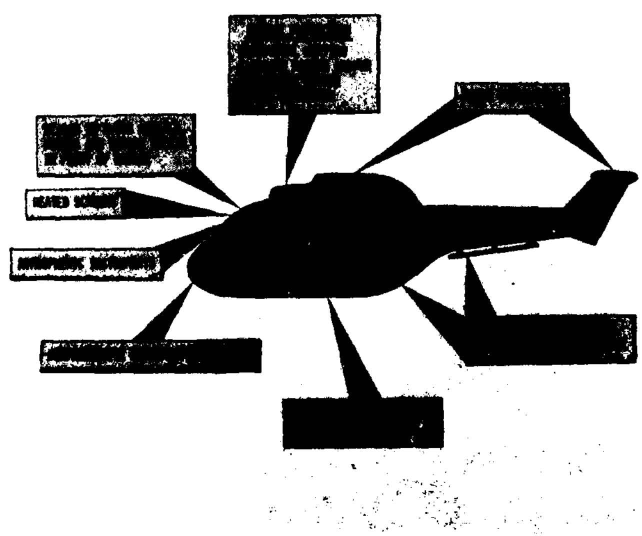


Figure 14. THE FLIGHT IN ICING HELICOPTER

METHODES D'ESSAI DU COMPORTEMENT DES AERONEFS EN CONDITIONS GIVRANTES ET DES SYSTEMES DE PROTECTION CONTRE LE GIVRAGE

par

Martin FRIEDLANDER
Ingénieur Navigant d'Essais
Essais-Equipements
CENTRE D'ESSAIS EN VOL
91220 - BRETIGNY-SUR-ORGE
(France)
Tél. 084 95 70 - poste 3242

Le présent exposé portant sur le même sujet que celui traité par M. GIBBINGS dans la présentation n° 19 au FMP, a été rédigé de façon à lui être complémentaire.

Aussi ne revient-il pas sur certaines définitions.

1 - LE GIVRAGE -

Il est important d'insister sur l'aspect dynamique du phénomène givrage rencontré en vol, ce qui le distingue d'une formation de glace par simple gel statique d'eau liquide. L'inclusion de bulles d'air, de cristaux de glace parfois, et la rapidité de passage à l'état solide de gouttelettes d'eau qui se trouvaient précédemment dans un état de surfusion particulièrement instable, créent des dépôts de givre qui adhèrent très bien sur les parties exposées des aéronefs. Si l'on ajoute à cela l'état de surface finement rugueux des profils lorsqu'ils sont érodés, on peut imaginer les difficultés rencontrées parfois pour faire partir les dépôts.

De nombreux paramètres influent sur la formation du givre, mais en fait, il suffit de retenir les principaux :

- la température de l'air chargé de gouttelettes ,
- la "teneur en eau", généralement exprimée en grammes par mètre cube de nuages,
- le "diamètre volumique médian", paramètre un peu artificiel qui cherche à rendre compte du spectre dimensionnel des gouttes d'eau ,
- la vitesse aérodynamique.

De façon encore plus simplifiée, on considère que la "sévérité" du givrage est directement liée à la teneur en eau, à vitesse donnée bien entendu, ce qui permet d'utiliser des témoins de givrage comme indicateurs simplifiés de teneur en eau. La sévérité de givrage est chiffrée en "vitesse de dépôt", généralement exprimée en mm/mn.

Diverses combinaisons de températures et de diamètres de gouttes entraînent des formations de givre "en pointe", c'est-à-dire assez aérodynamiques, d'autres combinaisons donnent le givre "en cornes" ou "en champignon", si pénalisant sur le plan aérodynamique.

La présence de cristaux de glace ou de neige dans les nuages surfondus augmente de façon très importante les dépôts de givre capté.

Malheureusement, il n'existe pas à l'heure actuelle de moyen simple d'identififer la présence de cristaux de glace, ni d'en mesurer les dimensions et la concentration dans les nuages.

2 - LES CONSEQUENCES DU GIVRAGE -

Les dépôts de givre agissent de plusieurs façons sur la mécanique du vol des aéronefs .

Pour des aéronefs moyennement ou faiblement motorisés, la combinaison de perte de puissance des moteurs, de dégradation des conditions aérodynamiques du vol et d'accroissement de la masse peut entraîner l'impossibilité de conserver le niveau de vol, la diminution de vitesse, puis un décrochage prématuré, à moins que l'aéronef n'ait percuté le sol auparavant !

Pour les aéronefs fortement motorisés, donc à performances souvent élevées, le problème est moins crucial mais il subsiste durant la phase d'attente à moyenne ou basse altitude, surtout pour les ailes à profils modernes, très sensibles aux modifications de formes.

Pour un avion comme Concorde, le problème est un peu différent ; de par sa forme géométrique et l'emplacement des moteurs, l'aile sert partiellement d'entrée d'air ; c'est à ce titre qu'elle a été protégée contre le givrage.

Pour les avions de combat, l'absence de phase d'attente prolongée permet de voler sans protection, sauf éventuellement pour les avions "école" qui sont appelés à tourner à vitesse réduite en circuit d'aérodrome.

Les conséquences du givrage des hélicoptères sont un peu particulières. Elles découlent du régime aérodynamique cyclique des pales et du dégivrage naturel anarchique des rotors crée par la force d'inertie centrifuge.

3 - PRINCIPES DE PROTECTION CONTRE LE GIVRAGE -

Dans l'état actuel de la technique, les constructeurs n'appliquent que deux grands principes :

- l'antigivrage local ,
- le dégivrage local.

Il est en fait dommage d'employer une énergie importante à faire fondre de la glace qui a tendance à se former sur les profils, alors que l'instabilité même du phénomène de surfusion des gouttelettes devrait permettre, au prix d'une quantité d'énergie bien moindre de transformer l'eau en cristaux de glace avant l'arrivée sur les profils. Il n'y aurait plus alors formation de dépôts.

C'est regrettable mais le télé-antigivrage par rayon sophistiqué n'existe pas encore.

En l'attendant, on construit des systèmes assez lourds et gourmands en énergie ... donc en carburant.

Le dégivrage pneumatique par déformation brutale de chambres en caoutchouc collées sur les bords d'attaque des voilures est perfectionné et utilisé sur les avions volant relativement lentement.

L'antigivrage et le dégivrage par liquide existent toujours. Ils ont même été perfectionnés par l'emploi des bords d'attaque en matériaux poreux, permettant l'exsudation contrôlée des produits chimiques.

L'antigivrage et le dégivrage thermiques sont assurés soit par air chaud, soit par résistance électrique ; les tuyauteries et échangeurs, ainsi que les systèmes de cyclage et de sécurité sont bien au point, mais lourds, encombrants et coûteux.

Depuis quelques années, les soviétiques ont développé un système de dégivrage à impulseurs électro-magnétiques, qui reprend le principe de l'ancien dégivrage cyclique pneumatique, mais n'entraîne plus qu'une déformation imperceptible des profils lors de fonctionnements impulsionnels du système ; ce principe, très économique en ce qui concerne l'énergie, paraît promis à un bel avenir si les applications ne font pas apparaître de problème technologique majeur.

Pour protéger les hélicoptères, les constructeurs utilisent les mêmes principes que pour protéger les avions, en ce qui concerne les moteurs, les points sensibles de la cellule et les pales. Les techniques de protection des pales ont dû être patiemment adaptées et c'est surtout l'arrivée de la génération des pales plastiques qui a permis de maîtriser le problème.

4 - MISE AU POINT ET ESSAIS DES PROTECTIONS -

La mise au point et la démonstration de validité des systèmes de protection contre le givrage nécessitent en général une cascade de moyens d'étude et d'essais.

La première étape est, bien évidemment, l'ETUDE. La réflexion basée sur l'expérience, permet rapidement de retenir tous les points techniques qui mériteront d'être vus en détail, et d'éliminer, avec justifications, ceux qui n'entraînent pas de risque particulier.

L'étude permet également de choisir dans la panoplie des moyens de démonstration, ceux qui sont les plus appropriés et de justifier les choix.

Enfin les calculs et la conception permettent de dimensionner les pièces et d'en lancer la fabrication.

Les ESSAIS EN SOUFFLERIE DE GIVRAGE sont de deux types principaux :

- les essais de détermination des zônes et étendues de captation, lorsque cette détermination ne peut être assurée par le calcul. Ces essais peuvent être faits sur maquettes à échelle réduite,

- les essais, à échelle 1, d'efficacité d'éléments de protection. Il s'agit en général de tronçons représentatifs. Il est important de noter que les effets thermiques ne peuvent pas être simulés à échelle réduite de façon simple.

Les ESSAIS DE LABORATOIRE permettent de vérifier certaines caractéristiques des éléments de systèmes de protection.

Les ESSAIS EN VOL EN CIEL CLAIR permettent, lorsque c'est nécessaire, de vérifier les qualités thermiques ou aérodynamiques des protections, le ruissellement de fluide ou le comportement de l'aéronef chargé de givre sur ses zônes non protégées.

Enfin deux types d' ESSAIS EN VOL EN GIVRAGE sont pratiqués :

- les essais en givrage artificiel dans des nuages produits par des installations fixes ou mobiles,
- les essais en givrage naturel.

5 - ESSAIS EN VOL EN GIVRAGE -

5. 1. - Essais en givrage artificiel.

Les conditions naturelles de givrage sont difficiles à trouver lorsque l'on recherche des sévérités constantes et intéressantes. Aussi s'est-on ingénié depuis vingt cinq ans à faire voler des rampes munies d'injecteurs d'eau de façon à créer, en ciel clair, des nuages de caractéristiques connues, contrôlées et constantes. On a ainsi constitué des sortes de souffleries givrantes, de dimensions infinies, permettant l'étude du givrage et de ses effets en conditions de vol réelles. L'idée est séduisante. Elle a pourtant ses inconvénients. Les aspects principaux sont résumés ci-dessous :

AVANTAGES.

- possibilités fréquentes de vol en ciel clair, à altitudes et températures déterminées d'avance, d'où rapidité de déroulement des campagnes d'essais,
- constance et reproductibilité des conditions givrantes,
- souplesse d'adaptation aux différentes phases de mise au point, reprise d'essais après modification,
- étendue limitée du nuage en largeur et épaisseur, donc possibilité d'étudier la captation sur une zône particulière de l'aéronef en essai, sans perturber les autres zônes,
- relative facilité des essais,
- nombre contrôlé d'heures d'essai donc coût maîtrisé,
- possibilité d'effectuer les essais pendant de nombreux mois de l'année.

INCONVENIENTS.

- givre produit assez différent du givre naturel, du fait de la pulvérisation des gouttes en air sec, donc loin de la saturation,
- diamètre volumique médian et spectre dimensionnel décalés vers les grosses gouttes du fait de l'évaporation des petites,
- difficultés pour assurer une distance constante entre l'aéronef en essais et la grille, d'où variations des caractéristiques des gouttes et de la teneur en eau,
- vol de l'aéronef en essai en atmosphère turbulente provoquée par le sillage de l'aéronef pulvérisateur,
- faible étendue du nuage en largeur et épaisseur, ne permettant pas de juger le comportement de l'aéronef totalement givré,
- en général, autonomie faible de la pulvérisation d'eau pour les fortes teneurs en eau (excepté avec l'avion KC 135).

Compte tenu de tout ceci, les essais en vol en givrage artificiel sont avant tout réservés aux essais de mise au point et ils peuvent rarement à eux seuls, constituer la démonstration globale et finale d'aptitude au vol en conditions givrantes.

Cela s'est montré particulièrement vrai pour la grille de givrage pour hélicoptère du National Research Council du Canada, à OTTAWA, puisque la grille est fixe, à l'air libre, et que seuls des vols stationnaires dans l'effet de sol peuvent y être effectués.

Aux Etats-Unis, le système HISS (Helicopter Icing Spray System) a été développé pour remédier aux défauts de la grille fixe. Il est mobile puisque constitué d'un hélicoptère CH 47 pulvérisateur, mais il est affecté de la majorité des inconvénients cités précédemment.

En ce qui concerne les avions pulvérisateurs dont les exemples les plus récents sont un KC 135 et un CESSNA ... aux Etats-Unis, et un CANBERRA au Royaume Uni, on trouve ici encore des applications dans la mise au point des sytèmes de dégivrage. Les résultats des essais en givrage artificiel sont rarement acceptés sans discussion lorsqu'ils sont présentés comme des justifications aux exigences des normes, par exemple de la FAR 25-1419 appendice C.

Les essais en vol en givrage artificiel peuvent parfois compléter les essais faits en soufflerie de givrage, mais l'on a vu des cas où le même point d'essai, effectué avec les deux moyens, a donné des résultats franchement différents sans qu'il soit vraiement possible d'incriminer l'un des moyens d'essai plus que l'autre.

5. 2. - Essais en givrage naturel.

La seule façon donc de juger la validité des systèmes de protection contre le givrage d'un aéronef, dans leur ensemble, semble être de les essayer dans les conditions de givrage réelles, naturelles. Malheureusement là encore nous allons trouver, à côté des avantages, un certain nombre d'inconvénients. Si nous regardons les principaux, nous trouvons :

AVANTAGES.

- conditions naturelles, réelles, donc inattaquables sur le plan de la représentativité !
- conditions d'ambiance totale, de grande étendue par rapport aux dimensions de l'aéronef, d'où possibilité de juger le comportement global, mais aussi tous les aspects du problème : conditions de travail de l'équipage, moyens de détection ou d'appréciation, phénomènes associés (bruit des précipitations, turbulence naturelle, foudre, grêle éventuelle ...).

INCONVENIENTS.

- difficulté, sinon quasi impossibilité à l'heure actuelle, de mesurer les paramètres principaux du givrage, à part la température de l'air,
- impossibilité donc de se situer valablement par rapport aux exigences des normes,
- difficulté de réaliser les essais puisque de très nombreuses conditions doivent être réunies : situation météorologique favorable, contrôle aérien sûr et efficace, zônes survolées convenant à des essais parfois dangereux,
- immobilisation parfois longue de l'aéronef pour des "campagnes d'essais" et coût élevé de ces campagnes.

Si nous discutons chaque point plus en détail, nous voyons que certains inconvénients sont communs, en fait, à tous les types d'essais en givrage. Ainsi en est-il de la position du point d'essai effectué par rapport au domaine des normes.

En givrage naturel, il est certain que l'on rencontre des conditions réelles, indiscutables, mais faute de savoir les mesurer, on ne saura pas dire si l'essai aura été sévère ou non, ou du moins comment il l'aura été.

Et puis même saurait-on mesurer avec précision et rapidité la teneur en eau, comment utiliser un tel paramètre, qui peut varier en permanence dans un rapport de 5 ou 10 dans les nuages cumuliformes ? Comment le comparer à des normes rigides, monolithiques ?

Ou comment faire évoluer les normes pour tenir compte de variations incessantes de teneur en eau ? Serait-ce même souhaitable ? représentatif ?

Le principal problème avec les essais en givrage naturel est que l'on ne sait jamais quand s'arrêter. A-t-on rencontré un cas suffisamment représentatif ? A-t-on eu assez de nuages stratiformes, de nuages cumuliformes, de cristaux de glace ... pour bien démontrer que l'avion peut voler sainement dans toutes les conditions givrantes ?

Dans la façon de conduire les essais, il faut savoir être très souple si l'on veut être efficace pour un prix raisonnable.

5. 2. 1. - Essais des avions en givrage naturel.

Ces essais suivent toute la somme des démonstrations, justifications et essais précédents. Ils ont surtout pour but de vérifier que tout se passe bien pour l'ensemble de l'avion, qu'aucun point n'a été oublié et que les clients utilisateurs n'auront pas de surprise. Il ne serait en effet pas honnête de laisser voler les utilisateurs dans des conditions d'environnement hostile sans avoir préalablement volé dans ces conditions.

Les essais étant de comportement en givrage, "de vérification finale" et non de mise au point, il est possible et judicieux de les regrouper en une campagne spécifique, sans terrain de départ bien précis, afin de suivre les conditions givrantes dans une zône étendue, l'Europe et l'Afrique du Nord par exemple, ou l'ensemble du continent Nord-Américain (voir figure n° 1).

Il faut pour cela disposer d'un appareil de série, ou presque, mis en oeuvre par une équipe réduite et souple. Les essais peuvent être conduits en une semaine ou même moins, pour trois ou quatre vols significatifs dans les configurations les plus représentatives.

Un exemple typique est celui de l'Airbus A 300 B avec lequel le givrage a été cherché et trouvé à Hambourg, et quarante huit heures plus tard à Tunis.

Les configurations essayées sont en général,(voir figure n° 2) :

- éventuellement croisière (pour avions non pressurisés),
- attente lisse,
- attente becs et volets sortis,
- approche, suivie de remise des gaz avec rentrée du train puis des dispositifs hypersustentateurs,
- cas de pannes typiques.

La rentrée des éléments mobiles après captation peut poser certains problèmes qu'il est bon d'identifier avant de certifier l'avion.

5. 2. 2. - <u>Essais des hélicoptères en givrage naturel.</u>

Compte tenu de l'état actuel des connaissances en aérodynamique de l'hélicoptère et des moyens d'essais des rotors à l'échelle 1, les calculs et les essais au banc, tout comme les essais en givrage artificiel, sont insuffisants pour mettre au point et justifier le comportement de la machine ou les moyens de protection. Les essais en vol en givrage naturel restent donc le moyen principal et les essais doivent s'étendre sur de longues périodes, à proximité des installations du constructeur ou d'un centre bien équipé. Dans les premiers temps de la mise au point, l'équipe sera nécessairement plus nombreuse et les déplacements difficiles. Il n'est pas possible alors de suivre les conditions météorologiques favorables, il faut les attendre sur place. Le choix du lieu d'attente, du "camp de base" devient très important puisqu'il faut optimiser le mélange des paramètres suivants :

- statistiques météorologiques favorables,
- zône non dangereuse et praticable pour le "Search and Rescue",
- contrôle aérien souple, avec en général contrôle radar positif,
- hangars et moyens techniques sol importants,
- facilité et rapidité de liaison avec l'usine du constructeur,
- laboratoire photo et moyens de dépouillement,
- possibilité d'hébergement et de vie du personnel.

Compte tenu de l'importance des essais, l'installation de mesures doit être assez diversifiée et complète. Elle doit permettre de chiffrer :

- les paramètres habituels de vol, et notamment ceux représentant les performances (régime rotor, régimes moteurs, températures moteurs, couples, vitesse, altitude ...),
- les niveaux des contraintes les plus importantes,
- les niveaux vibratoires,
- les paramètres représentatifs du fonctionnement des systèmes de protection,
- l'importance des dépôts résiduels de givre ,

et bien sûr les paramètres du givrage : vitesse de dépôt, température du nuage, teneur en eau et éventuellement, diamètre des gouttelettes.

Enfin, les essais eux-mêmes doivent s'appliquer à toutes les phases de vol de l'hélicoptère car le domaine de vol altitude-vitesse est tel que l'hélicoptère est susceptible, à la limite, d'effectuer toute la durée d'un vol en conditions givrantes, (voir figure n° 2) :

- roulage au sol en brouillard givrant ,
- attente au sol en brouillard givrant,
- montée,
- palier croisière,
- palier attente,
- descente,
- approche,
- remise de pas, nouvelle attente ... etc,

en n'oubliant pas, bien entendu, les <u>cas de pannes</u> les plus représentatifs.

Le programme d'ensemble "mise au point" plus "certification" s'étale bien sur cinquante à soixante heures de vol en conditions givrantes diverses, soit 150 à 200 heures de vol au total, y compris en cumulus congestus avec risques associés (foudre, grêle, turbulence ...). Il est difficile de réaliser cela en une seule saison et, dans l'état actuel de la technique, il est sage de prévoir des essais étalés sur deux ans.

Encore faut-il être conscient que la protection des hélicoptères contre le givrage est une technique récente et que l'on ne peut lui demander les mêmes qualités qu'à la protection"avions"qui bénéficie de plusieurs dizaines de milliers d'heures de vol d'expérience.

En fait, vu la taille et la technologie actuelle des hélicoptères, il faut comparer ceux-ci aux avions bimoteurs à turbopropulseurs (type aviation générale ou 3ème niveau) et non aux jumbo-jets ou avions de ligne récents !

Ce point, très important, semble être fréquemment ignoré, d'où des malentendus lors des discussions de certification.

6 - CONCLUSIONS -

Les essais en vol d'aéronefs en conditions givrantes sont fortement marqués par la "non disponibilité" et la "non reproductibilité" des conditions atmosphériques.

Contrairement aux essais en vol courants de performances ou de qualités de vol, qui se déroulent en atmosphère standard ou pour lesquels il est possible de corriger l'effet des changements de conditions, les essais qui nous concernent sont tributaires de nombreux paramètres qui nous échappent et que l'on ne sait pas facilement mesurer.

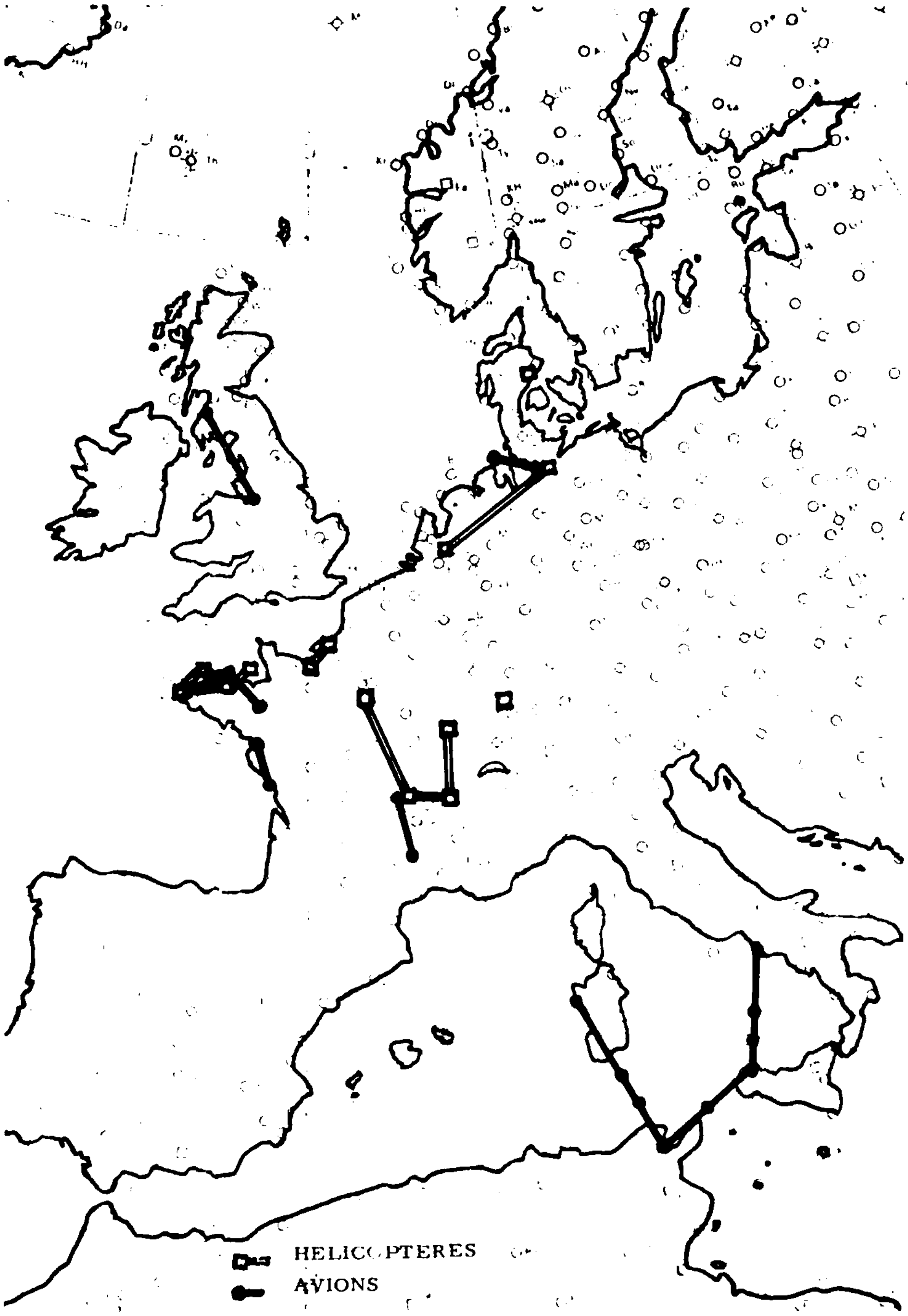
La production de givrage artificiel paraît être la solution miracle, au moins par sa "constance" et sa "reproductibilité", mais les difficultés techniques pour l'assurer et les différences importantes entre givrage artificiel et givrage naturel font qu'il n'est pas possible de tout justifier de cette façon.

Pour les hélicoptères, ce sont encore la mise au point des protections et l'étude du comportement des machines qui sont assurées en givrage naturel.

Pour les avions, c'est principalement la vérification du comportement de l'aéronef, dans ses phases de vol réelles.

Même dans ce dernier cas, les essais ne doivent pas être négligés ; ils apportent une garantie de qualité pour l'ensemble des problèmes du vol en environnement hostile alors que, en général, tous les essais en vol d'un avion se sont précédemment passés en ciel clair, ou presque.

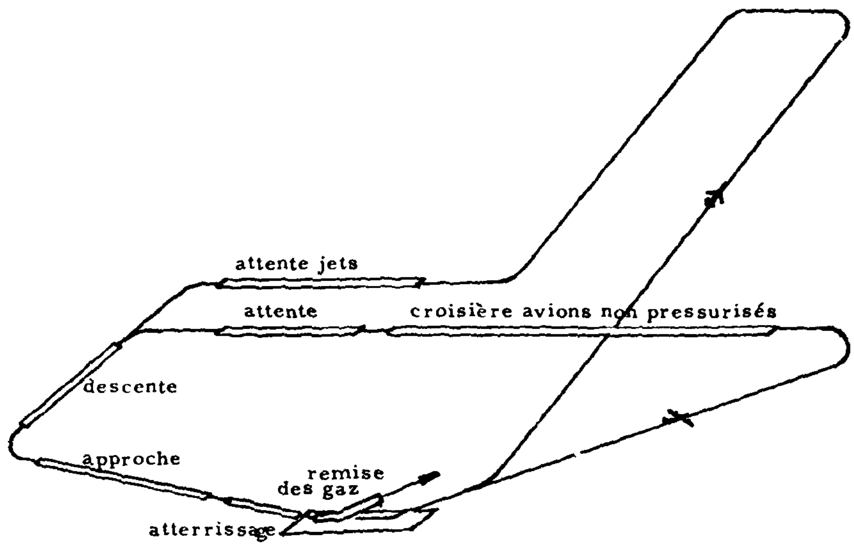
Figure 1



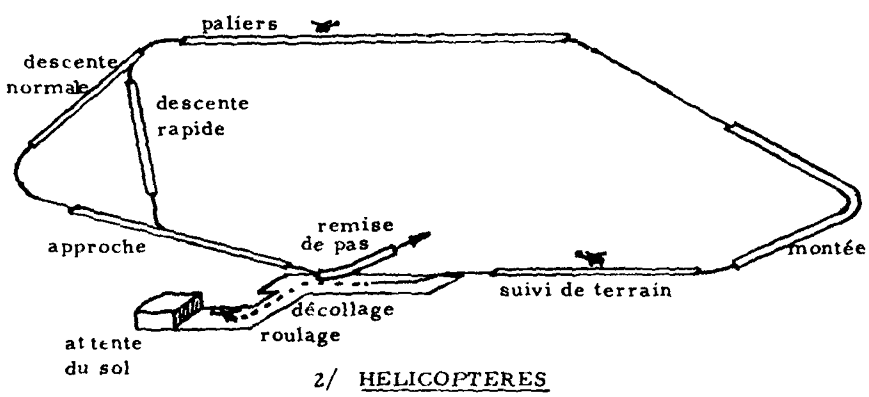
GIVRAGE NATUREL AVIONS ET HELICOPTERES

ZONES DE GIVRAGE UTILISEES LORS DES ESSAIS D'AERONEFS FRANCAIS

Figure 2



1/ AVIONS



2/ HELICOPTERES

ESSAIS EN GIVRAGE NATUREL

CONFIGURATIONS A ESSAYER

Figure 3

	CONFIGURATION	A GIVRE	B NEIGE	C Conditions mixtes	D Pluie verglaçante	E Turbulence Foudre Grêle
VOL NORMAL	1.1. ROULAGE - ATTENTE AU SOL	OUI 10+30 mn a b c d e	OUI 10+15mn a b	NON	OUI 15 mn a b	NON
	1.2. STATIONNAIRE	OUI 30mn HES a b c d e	OUI 30mn DES a b	NON	NON	NON
	1.3. TRANSLATION FAIBLE IV	NON couvert par 2 A	OUI 30 mn a b c	NON	OUI 30 mn a b	NON
	1.4. DECOLLAGE - ACCELERATION	OUI en cond. BG a b c d	OUI a b	NON	OUI a b	NON
	1.5. MONTEE	OUI* a b c d e	OUI* a b	OUI* b c d e	NON	OUI* b c
	PALIERS					
	1.6. Suivi de terrain. Vitesse à définir	OUI 1h00	OUI 1h00	NON	NON couvert par 3 D	NON
	1.7. Croisière long range	OUI 1h00	OUI 1h00	OUI 30 mn	NON	OUI 30 mn
	1.8. Croisière maxi en givrage	OUI 1h00	NON	OUI 10 mn	NON	OUI 10 mn
		a b c d e	a b c	b c d e		b c
	DESCENTE					
	1.9. Normale	OUI*	OUI*	OUI*	NON	OUI*
	1.10. Rapide	OUI* a b c d e	NON	NON	OUI*	OUI*
	1.11. APPROCHE ET DECELERATION	OUI épais. 10mm b c	OUI 6mn	NON	OUI 6mn	NON
	1.12. REMISE DE PAS AVEC RENTREE DE TRAIN	OUI b c	OUI	NON	NON car visibilité suffisante	NON
CAS DE PANNES	2.1. PANNE 1 MOTEUR AU DECOLLAGE	OUI b c	NON	NON	OUI a b	NON
	2.2. PANNE MOTEUR EN CROISIERE	OUI 45mn b c	NON couvert par 9 B	OUI 10mn b c	NON	NON
	2.3. PANNE MOTEUR EN APPROCHE	OUI 10mm sur témoin b c	NON	NON	NON	NON
	2.4. DELAI DEGIVRAGE ROTOR	OUI b c d	NON	OUI c d	OUI a b	NON
	2.5. PANNES PARTIELLES PROTECTION ROTORS	OUI b c d	NON	OUI c d	OUI a b	NON
	2.6. PANNE TOTALE PROTECTION ROTORS - principal - arrière	OUI b c d	NON	OUI c d	OUI a b	OUI b c

* Durée à fixer pour chaque type d'hélicoptère.
Limite de couple à déterminer avant ou pendant les essais.

BG = brouillard givrant

GIVRAGE NATUREL HELICOPTERES

TABLEAU DES CONFIGURATIONS A DEMONTRER

SUMMARY OF SESSION III - ENVIRONMENTAL TESTING

Environmental testing has achieved a considerable maturity and is now regarded an essential feature of an aircraft development program. Outstanding facilities exist in the USAF Climatic Hangar, Canadian National Research Council Helicopter Spray Rig, US Army Helicopter Icing Spray System, to mention but a few - and these facilities are being constantly used for the development and qualification of new systems, aircraft, alternate fuels, etc. Considerable investment by major aircraft and helicopter manufacturing firms and their counterpart government certification agencies attests to the importance of environmental testing for current and future helicopters. Major investments have been made in instrumentation to determine the test environment, results of exposure of test systems to those environments, and monitoring of flight safety parameters. It is agreed that environmental qualification is an incremental process, often starting with design support testing, then full scale simulated environment testing, and finally natural environment testing. It should be mentioned that for the specific case of helicopter icing, an expanded discussion of this subject will appear in an AGARDograph to be issued in Spring 1981 by the FMP Working Group 09.

L'INTERFACE MESURE DES SYSTEMES DIGIBUS

par
C. RAT
Centre d'Essais en Vol/SM
91220 Brétigny-sur-Orge
France

Cet exposé présente le Digibus à travers le programme Mirage 2000, en raison du rôle fondamental qu'il a joué dans la réalisation du système d'armes de cet avion. En effet pour utiliser les grandes capacités opérationnelles de l'avion, un effort important d'intégration a été fait au niveau de la cabine pour réaliser les meilleurs compromis dans la présentation des paramètres et des commandes du système au pilote : visualisations tête haute adaptées à chaque phase de vol, commandes multiplexées, présentations synthétiques de situations tactiques. L'utilisation d'un bus numérique est absolument nécessaire pour mener à bien la réalisation d'un tel programme.

I - Introduction

II - L'avionique et le bus numérique

III - Aspect technique du digibus

IV - Relation bus numérique - visualisation programmable

V - Conclusion.

I - INTRODUCTION.HISTORIQUE

Le Digibus constitue une interface qui permet une transmission de données sous forme numérique multiplexée entre les différents éléments d'un système, en l'occurrence le système de navigation et d'attaque du Mirage 2000.

Il a été développé par l'électronique Marcel Dassault (EMD) en 1972 dans le cadre du programme de l'avion de combat futur (ACF). Par suite de l'évolution des programmes les études faites pour l'ACF ont été utilisées au bénéfice du système d'armes du Mirage 2000.

Le Digibus est utilisé actuellement sur les prototypes du Mirage 2000 et les avions de servitude du programme 2000 qui volent au Centre d'Essais en Vol (CEV). On peut dire que le Digibus a été crée pour distribuer l'information :il assure la transmission des données entre les éléments du système d'armes d'un avion de combat.

Eléments de base du système d'armes du Mirage 2000 - Figure 1

- En rouge :
  . Centrale à inertie UNI 52
  . Centrale aérodynamique
  . Calculateur principal CP 2084
  . Unité secondaire de gestion USG 284
  . Boîtier générateur de symboles BGS
  . Pilote automatique PA
- Encadrés
  . Viseur tête haute
  . Viseur tête basse
  . Visualisation contre-mesure
- En blanc
  . Indicateur de navigation IDN
  . Indicateur attitudes-cap
  . Radar Doppler (RDM ou RDI)
  . Contremesures
  . Poste de commande navigation PCN
  . Poste de commande armement PCA
  . Poste de commande Radar PCR.

.../...

Les six premiers éléments comportent un calculateur numérique capable de traiter les informations avant de les envoyer sur le Digibus, le calculateur principal occupant une place particulière en tant que gestionnaire des échanges. Les trois suivants sont des visualisations programmables dont les images sont calculées par le boîtier générateur de symboles.

II - ETAT ET TENDANCE DE L'AVIONIQUE - LE BUS NUMERIQUE

L'évolution actuelle de l'avionique est considérable et se manifeste principalement par la numérisation. Celle-ci a commencé dans les organes de traitement (les calculateurs) puis a gagné les capteurs (centrales aérodynamiques, centrales à inertie, radars), les pilotes automatiques ainsi que les commandes de vol et le contrôle moteur.

La numérisation des échanges : le bus.

Par les systèmes 2000 actuels les informations sont obtenues directement et transportées sous forme numérique. On voit ainsi apparaître un avantage fondamental des systèmes numériques, il est en effet possible par des méthodes dites de "multiplexage temporel" de faire passer des informations différentes sur un seul fil. On les envoie successivement en les accompagnant d'une information (adresse) permettant de les retrouver à l'arrivée.

Les premiers bus ne comportaient qu'un émetteur et plusieurs récepteurs. L'émetteur est l'équipement qui élabore un groupe d'informations, le récepteur celui qui en a besoin en totalité ou en partie.

Figure 2 - Exemple de messages où la centrale à inertie est l'émetteur.

Depuis le début des années 70 différents types de bus ont été développés pour lesquels chaque équipement connecté peut être tour à tour émetteur ou récepteur. La diminution du volume de câblage et la possibilité de traitement a fait adopter ce type de bus sur de nombreux programmes :

. Mirage 2000

. Mirage F1 nouvelle version

. Atlantic nouvelle génération ANG.

Ils ont également été utilisés sur d'autres systèmes en particulier conçus pour l'aviation civile comme la Caravelle ALIS.

III - ASPECT TECHNIQUE DU DIGIBUS

Sur le Mirage 2000, le Digibus est doublé dans un souci de fiabilité et de maintenance. Il y a donc un Digibus principal appelé Digibus 1 et le Digibus 2 Figure 3.

C'est pour l'unité de gestion en service, qui peut être soit le calculateur principal CP soit l'unité secondaire de gestion USG que s'effectue le transfert de l'information entre les deux Digibus.

. Aspect de la liaison équipement bus.

Structure du support de transmission.

Le branchement des équipements sur le Digibus (périphériques PI) se fait en dérivation ce qui permet de retirer un périphérique PI sans interrompre le fonctionnement du système. Seuls les équipements de bout de ligne qui comportent les résistances électriques d'adaptation doivent être remplacés en cas d'absence par une résistance équivalente Figure 3

La liaison proprement dite entre l'équipement et le bus se présente sous la forme d'un coupleur (CSS) associé à une logique émission - réception.

Figure 4 On observe sur la Figure 4 les trois types de redondances utilisées pour la fiabilité de la liaison :

. Redondance simple sur les interfaces des bus

. Redondance simple sur la logique et sur les interfaces des bus

. Redondance simple sur la logique double sur les interfaces des bus.

.../...

- Type de liaison

Le système d'échange des informations numériques est organisé autour de deux Digibus comportant chacun deux lignes :

. Une ligne de procédure LP indirectionnelle sur laquelle l'unité de gestion met les mots de procédure à destination des équipements

. Une ligne de donnée LD bidirectionnelle sur laquelle sont transmises les données en émission et réception.

- Caractéristique des signaux de transmission

Le rythme des bits est fonction des besoins en échanges d'informations et des possibilités d'extension que l'on désire. Une vitesse de 1 Megabits par seconde a été retenue.

Modulation et type de codage : le code utilisé est un code biphase retour à zéro et la modulation est à trois niveaux. La composante continue est nulle afin de pouvoir utiliser dans de bonnes conditions les liaisons par transformateurs.

Figure 5 - Schéma du code biphase.
Niveau de tension sur les lignes bus pour un niveau d'émission de 6 volts ($\pm$ 6 V + 1 V)

- Gestion des échanges

. Le caractère est l'élément de base de 10 bits indissociables à la transmission commençant par un bit de validité, un octet de huit bits utiles et un bit d'imparité portant sur tout le caractère Figure 5

. Le message est l'ensemble des caractères véhiculés en un seul bloc et dans un seul sens.

. L'échange est l'ensemble des messages véhiculés sur le bus à partir d'une seule initiative de l'unité de gestion.

- Type d'adressage

Deux types d'adressages sont utilisés dans la procédure d'échange.

- L'adressage explicite au physique (direct)

Il est utilisé lorsque les paramètres à véhiculer ne concernent que peu d'équipements récepteurs car il implique un mot de commande par récepteur concerné.

- L'adressage *implicite* au par label (indirect)

Il consiste à adresser de manière explicite l'émetteur du bloc de mots de données et de manière implicite les récepteurs. Il est intéressant pour les échanges ayant pour origine un équipement et pour destinataires de nombreux récepteurs.

IV - LIAISON ENTRE LES VISUALISATIONS PROGRAMMABLES ET LES BUS NUMERIQUES

Le nombre des informations captées, les possibilités de traitement offertes par les équipements et les possibilités d'échanges offertes par les bus numériques pourraient si elles étaient mal utilisées saturer le pilote avec une quantité d'informations toutes utiles mais pas au même instant.

Il a donc été nécessaire de procéder à l'intégration des visualisations et des commandes du système. Ainsi bien que les capacités du Mirage 2000 soient beaucoup plus importantes que celles de très nombreux chasseurs, la cabine pilote Figure 6 tout en restant assez chargée est comparable à celles de ces mêmes chasseurs.

L'ensemble des informations de pilotage, de navigation et de conduite de tir est présenté sur trois visualisations cathodiques programmables (c'est-à-dire dont l'image est déterminée par programme). Le bus fournit les informations traitées à une cadence suffisante pour obtenir une image continue.

Figure 7

- La tête haute - collimatée
- L'écran tête basse
- L'écran contre-mesure

.../...

La fiabilité des tubes cathodiques est maintenant suffisante pour que l'on puisse les considérer comme les instruments principaux de pilotage. C'est donc sur eux que repose la réussite de la mission, mais pas la sécurité car ils sont relayés en cas de panne par des instruments classiques de secours.

La réalisation du système d'armes a permis de ne présenter au cours de la mission que les informations strictement nécessaires à chaque phase de vol. C'est l'unité de gestion en service qui assure la gestion de la liste des informations à présenter à partir de la connaissance qu'elle a :

- de l'état des postes de commande.
- De l'état des fonctions du système de navigation et d'attaque.

Le conception de l'architecture numérique du système a été faite dans le but de simplifier au maximum la tâche du pilote. Celui-ci dispose au niveau des principales commandes du compte rendu par voyants de la sélection effective des fonctions dans les équipements. Les informations en provenance des principaux postes de commandes sont acheminées par le Digibus. Il s'agit des postes de commandes : Figure 8

- Radar
- Armement
- Navigation

Bien que d'accès facile, ces postes de commandes sont utilisés pour des présélections au cours de phases de vol pendant lesquelles la charge de travail du pilote est moins importante. De plus un MIP (module d'insertion de paramètres) introduit dans le PCN (poste de commande de navigation) permet de charger rapidement la totalité du plan de vol de l'avion. La préparation du MIP s'effectue au sol pendant la préparation de la mission.

Maintenance intégrée.

Toute la surveillance et l'analyse de panne d'un tel système n'étant plus possible par le pilote, des circuits d'autotests ont été incorporés dans les équipements numériques. Seuls les résultats sont communiqués au pilote pour lui indiquer s'il peut ou non débuter ou poursuivre sa mission dans les conditions prévues.

Ces moyens peuvent être utilisés pour la maintenance au sol par l'intermédiaire du Digibus :

- La mémorisation en vol des résultats d'autotests mauvais.
- Le traitement logique différé de ces mémorisations et visualisations.
- Le déclenchement de séquences particulières d'autotests.

A titre d'illustration la fonction navigation et approche fournit un exemple de visualisation programmable ; sur le Mirage 2000 elle est autonome et présentée en viseur tête haute. Elle permet :

- La navigation vers un but affiché.
- La navigation suivant une route désirée.
- L'arrivée sur le but à une heure désirée.
- Le recalage de la navigation.

Les informations sont présentées au pilote suivant deux modes :

- Le mode navigation Figure 9
- Le mode approche Figure 10

V - CONCLUSION

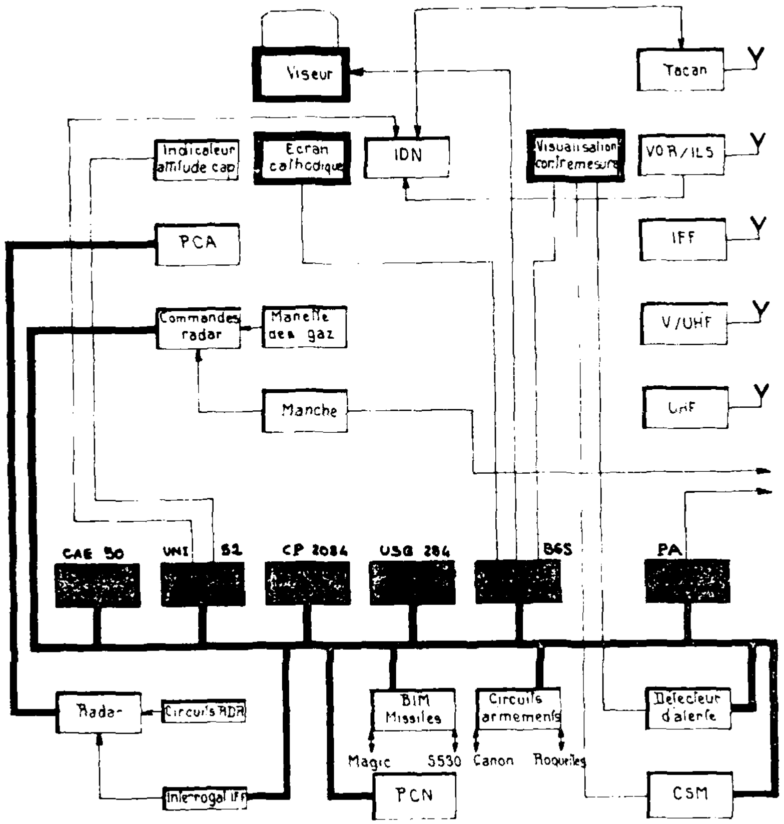
Le programme Mirage 2000 était ambitieux. Le choix de l'avion monoplace a conduit à une étude particulièrement poussée du système dont la complexité n'a pu être envisagée :

- Sur le plan utilisation par un seul membre d'équipage, que par l'optimisation du pilotage grâce aux moyens de visualisation programmable.

- Sur le plan technique, que par la numérisation des équipements et l'utilisation du Digibus pour permettre leur dialogue.

Fig. 1

Synoptique. Système de navigation et d'armement



Digibus Critère de transmission des informations
- Sur fond rouge les équipements comportant un calculateur numérique
- Equipements encadrés. Les trois visualisations programmables dont les images sont calculées par le boîtier générateur de symboles

Fig. 2

Sortie de la centrale à inertie

Outputs

	Platform synchro	Platform resolver	Amplified synchro
Analog	Pitch Roll	Azimuth	Pitch Roll Stabilized magnetic heading True heading
Digital (multiplexed digital bus)	Latitude Longitude Velocity X Velocity Y Vertical velocity Z Acceleration X Acceleration Y Vertical acceleration Z Wind direction Wind magnitude	Altitude Roll Pitch Platform azimuth Wander angle True heading Magnetic heading True air speed Ground speed Inertial slope Potential slope	Track angle Desired track Accross track Track angle error Δ Track angle Distance to go Target bearing Δ time Target number

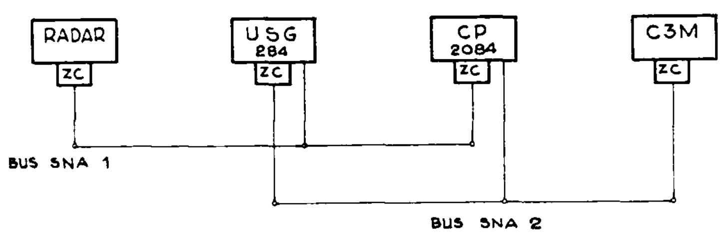
Messages

UNI 1	PHIU TETAU PSIU	ROLL PITCH TRUE HEADING
UNI 2	L G	LATITUDE LONGITUDE
UNI 3	V X V Y V Z VENT X VENT Y VENT Z	VELOCITY X VELOCITY Y VERTICAL VELOCITY Z WIND DIRECTION WIND MAGNITUDE
UNI 4		

Les sorties de l'UNI 52 sont regroupées en messages DIGIBUS (UNI 1 à 10) ayant leur cadence propre d'émission

Fig. 3

LES 2 BUS DU MIRAGE 2000



LE BRANCHEMENT EN DERIVATION

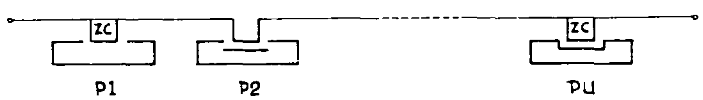
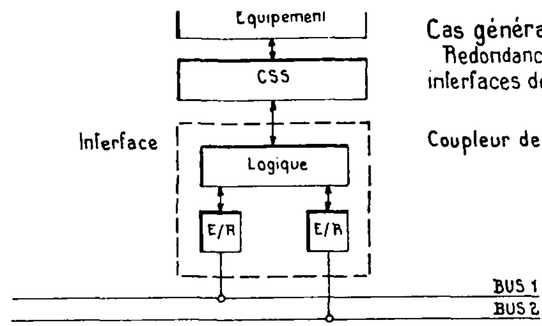


Fig. 4

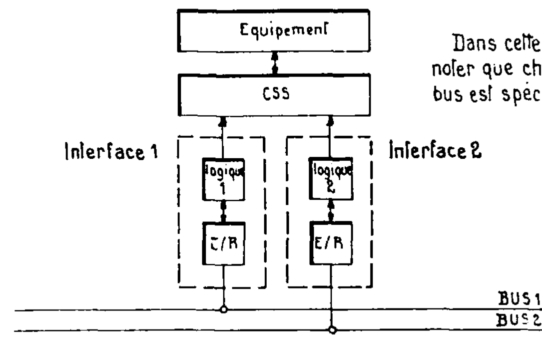
Cas général n°1
Redondance souple sur les interfaces des bus

Coupleur de bus standard



n°2. Redondance simple sur la logique et les interfaces des bus

Dans cette solution, on peut noter que chaque coupleur de bus est spécifique d'un digibus



n°3. Redondance simple sur la logique et double sur les interfaces des bus

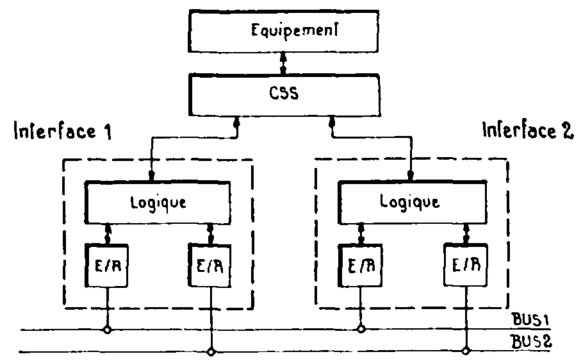


Fig 6

POSTE DE PILOTAGE DU MIRAGE 2000

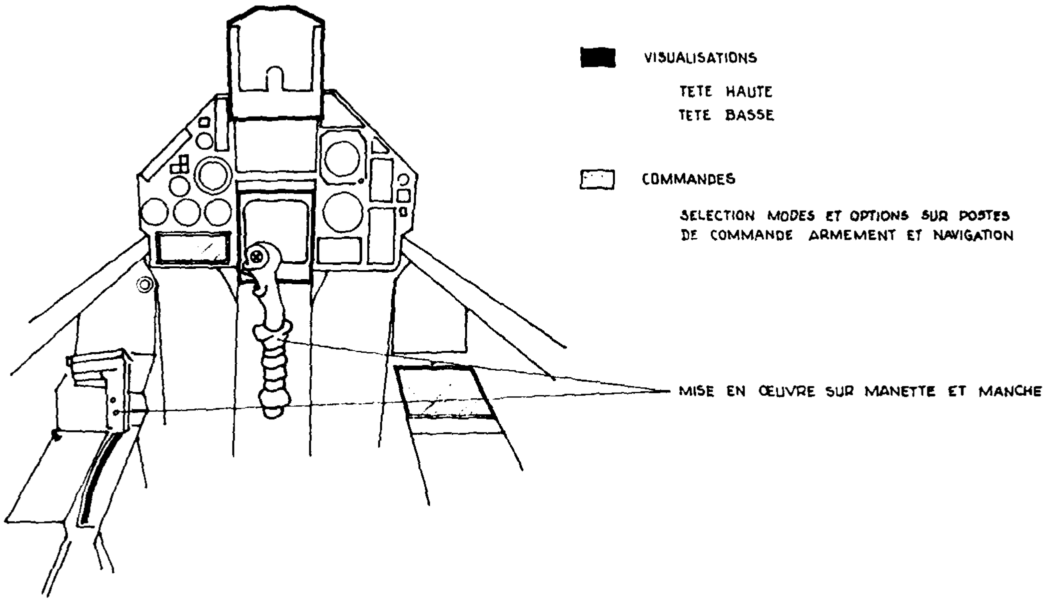


Fig. 5

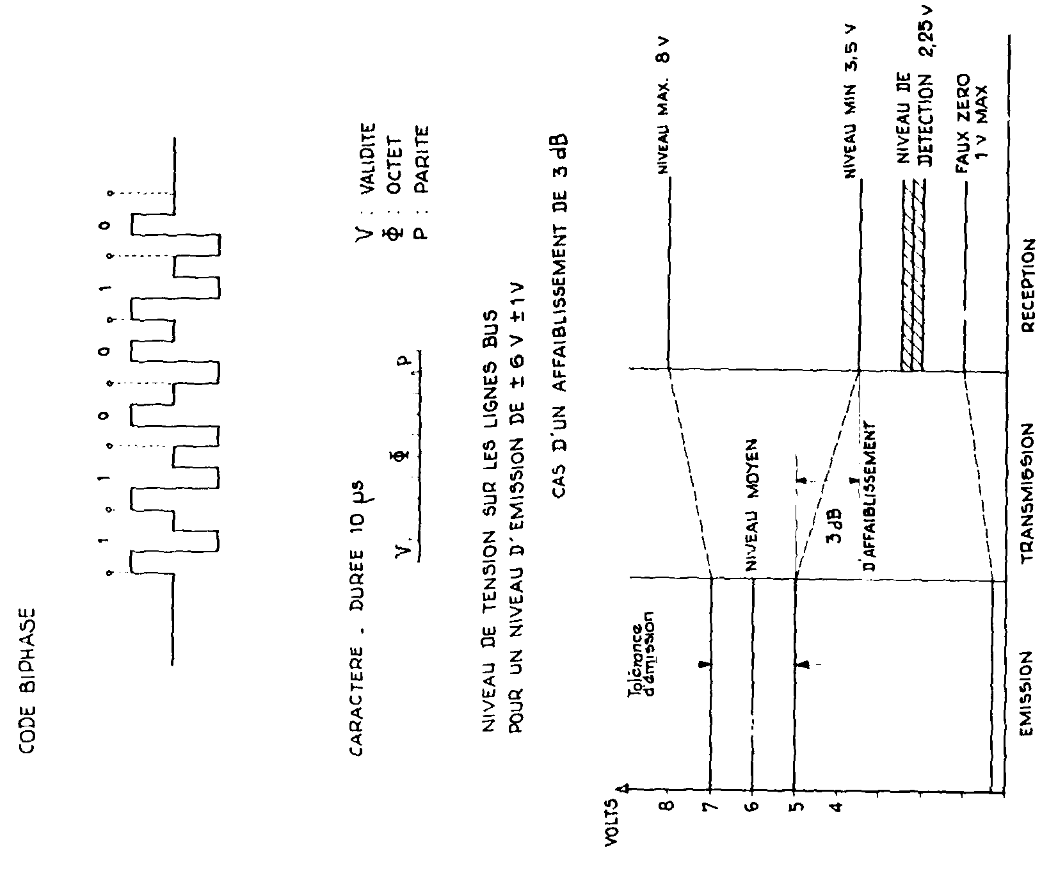


Fig. 7

Tête de visée

VE 130

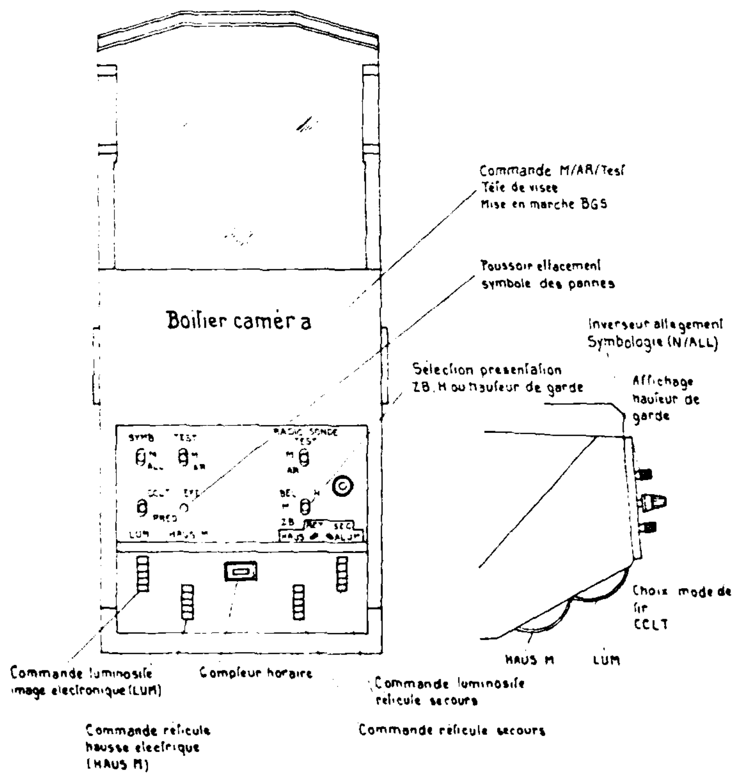
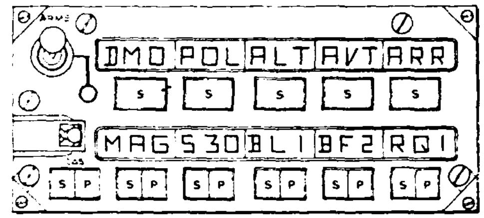
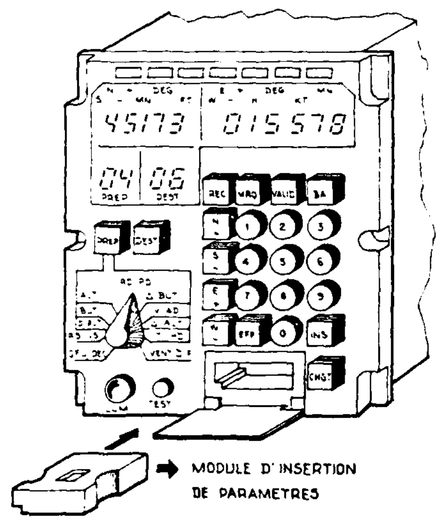


FIG. 8



Poste de commande armement



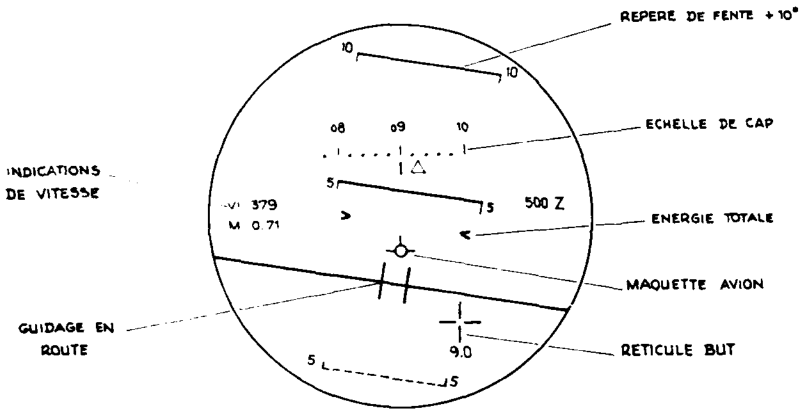
POSTE DE COMMANDE NAVIGATION

FIG. 9

FONCTION NAVIGATION

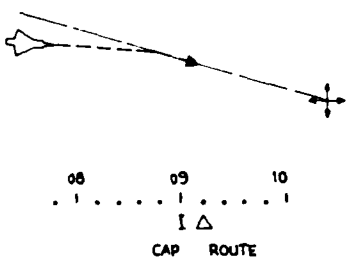
DISTANCE BUT < 10 NM

RETICULE BUT



COMMENTAIRES

LE PILOTE ARRIVERA SUR LE BUT SITUE A 9 NM
A LA ROUTE DESIREE 100 S'IL REALISE
LE GUIDAGE LUI FERA EFFECTUER CETTE TRAJECTOIRE



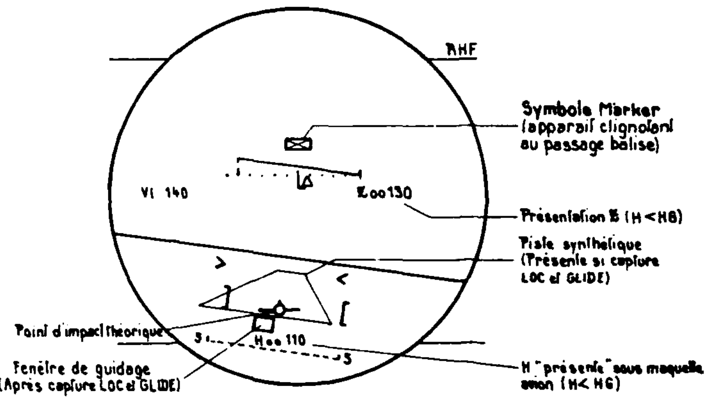
ON NOTE QUE L'AVION EST AU CAP 090 ET A LA ROUTE 092 : ON LIT DONC DIRECTEMENT UNE DERIVE DE 2° DROITE

Fig. 10

Fonction approche

Approche avec ILS

Après capture LOC et Glide
Cas H radio-sonde < HG
et sélection hauteur radio-sonde



Commentaires

Le pilote est averti qu'il doit
Virer à gauche et augmenter la pente en réalisant
L'ensemble maquette et fenêtre se place
alors en début de piste synthétique
Le pilote devra ensuite réaliser (régime)

>] [<

L'avion sera sur l'axe et en configuration normale
d'approche lorsqu'il obtiendra

>] [<

PROGRAMMABLE MULTIPURPOSE FLIGHT TEST INSTRUMENTATION SYSTEM

by

Rüdiger Karmann

Institut für Flugmechanik

Deutsche Forschungs- und Versuchsanstalt
für Luft- und Raumfahrt e.V. (DFVLR)
D 3300 Braunschweig-Flughafen, West Germany

SUMMARY

As a result of the increased use of flight control and stability augmentation systems, higher frequency contents of the control loops together with an increased amount of information provided by additional sensors have placed severer demands on the flight test instrumentation. To achieve a reasonable relationship between costs and benefits, a computer - controlled flight test instrumentation system was developed by the DFVLR. Through the modular construction of this device a wide field of applications is achieved.

1. BACKGROUND OF DFVLR FLIGHT TEST METHODOLOGY

In classical flight mechanics the improvement of flight performance and flying qualities have been the basic objectives. Here only the rigid body modes of flight vehicle dynamics are of importance. The rigid body modes have been satisfactorily described by a frequency bandwidth of 0 to 1 Hz.

As a result of the increased use of flight control and stability augmentation systems [Ref. 1], higher frequency contents of the control loops together with an increased amount of information provided by additional sensors, have placed severer demands on the flight test instrumentation [Ref. 2]. This development is mainly visible in the data rate necessary for flight tests. Fig. 1 shows the data rates for windtunnel and flight testing at the DFVLR Institut für Flugmechanik including an estimate for the next five years.

Until the year 1978 the data rates were not higher than 100 kbit/sec. With the beginning of the dynamic rotor tests (DRT) using the rotor test stand (RTS) and the Bo 105 helicopter tests, there has been a continuous increase of data rates since the year 1978. For future tests in the range of active control technology (ACT) an increase of data rates up to approx. 1 Mbit/sec is expected.

Until 1978 the general range of flight tests could be carried out satisfactorily using flight test instrumentation in analog frequency multiplex technology with channel numbers to max. 36. For future applications this system has been proved to be inadequate as regards bandwidth, channel numbers and accuracy.

In 1978 it therefore was decided to develop a new flight test system, the so-called Mobile Flight Test And Real-Time Data Analysis System.

2. APPLICATION FLEXIBILITY OF MOBILE FLIGHT TEST AND REAL-TIME DATA ANALYSIS SYSTEM

The aspect of cost-effectiveness in flight test techniques is becoming more and more important, as a result of the constantly increasing complexity of modern flight vehicles [Ref. 3]. For future flight tests a reasonable relationship between benefit and costs is only achievable by the consistent consideration of the following points:

- Short installation times of flight test instrumentation
- Automatic testing and monitoring of the on-board and ground-system
- Variable application spectrum by modularity in hard- and software

These requirements necessitate the consistent use of modern computer systems [Ref. 4].

Fig. 2 gives an impression of the Mobile Flight Test and Real-Time Data Analysis System.

At present the application-spectrum of this system covers research tasks of the Rotor Test Stand [Ref. 5], Helicopter Bo 105 [Ref. 6, 7] and the Inflight Simulator HFB 320 [Ref. 8, 9].

The Mobile Flight Test And Real Time Data Analysis System consists of the following components:

- PROGRAMMABLE MULTIPURPOSE FLIGHT TEST INSTRUMENTATION SYSTEM (PMFTIS) for on-board data acquisition and telemetry
- TELEMETRY AND TRACKING SYSTEM for the data-uplink and downlink as well as for the position-measuring of flight vehicles
- GROUND BASED DATA SYSTEM, including the equipment for telemetry, quick-look and data recording
- GROUND BASED DATA ANALYSIS SYSTEM with multi-computer systems for realtime data analysis such as harmonic analysis, spectral analysis, stress analysis, rotor stability analysis, flight performance analysis.

This last system is at present being developed. An account will be given here of the PMFTIS.

3. DESCRIPTION OF PROGRAMMABLE MULTIPURPOSE FLIGTH TEST INSTRUMENTATION SYSTEM (PMFTIS)

For the design of PFMTIS the following requirements were given special consideration:

- Compatibility of available measuring devices
- Modularity
- Suitability for a wide application range
- Expandability for future measuring tasks
- Simple operation
- Computer controlled built-in-test
- Automatic acquisition of the complete test-relevant parameters.

The block diagram shown in fig. 3 is based on the above-mentioned requirements.

In the maximum configuration the acquisition of sensor signals and the remote control of flight vehicles from the ground is possible using a PCM-downlink and uplink. A maximum of flexibility is achieved by the three bus-systems.

The central, interactive control of the whole system is accomplished with help of the Airborne Computer System and the Control and Display Unit.

In the Analog Signal-In Processor Modules (ASIPM) up to eight analog sensor signals are preamplified, filtered, digitized and buffered.

For the processing of the digital data, the so-called Digital Data-In Processor Modules (DDIPM), each with 8 x 16 digital inputs, are available.

The PCM Controller Encoder (PCMCE) reads the buffered data after addressing the single module. The control of the actuators by the PCM Controller Decoder (PCMCD) occurs in the complementary manner.

The total data acquisition (sampling of analog signals, storage of digital data) is controlled either by the Time Code Generator (TCG) or by an External Frame Sync. The Time Code Generator contains external synchronization inputs for automatic synchronization of the internal quartz oscillator by an external mother clock and automatic reading of time and date.

3.1 INTEGRATED SENSOR MODULE (ISM)

In the field of flight test techniques, in many cases it is difficult to transfer small sensor signals (microvolts to millivolts) to the signal processing electronics through long leads.

Therefore one should use Integrated Sensor Modules (ISM), as shown in fig. 4. Such modules contain, integrated in a screening box, the sensor, the preamplifier, and if necessary, a prefilter and isolation amplifiers for potential separation. Through the use of Torquer Sensor Systems the single sensor can also be included in the built-in-test.

Fig. 5 shows a three axis accelerometer-ISM. Here the sensors, the integrated electronics and the DC/DC-converter can be seen.

3.2 ANALOG SIGNAL-IN PROCESSOR MODULE (ASIPM)

Because of the relatively high signal level at the outputs of the ISM's, there is no need for high post amplification in the ASIPM. Fig. 6 shows the block diagram of an ASIPM. The signal path leads from the input switch to the programmable amplifier, the programmable low pass filter, the test multiplexer, sample and hold, multiplexer and 12 bit ADC to the E-Bus-Interface.

The selection of offset voltage, gain and filter cut-off frequency is made under computer control using the C-Bus-Interface and the control logic.

Any test signal can be generated using a special D/A converter. With the help of the input switches and the test D/A, a built-in-test including the sensors, is possible, as well as a test of only the ASIPM. Every single input channel can be selected by the test multiplexer from the control and display unit CDU. After voltage to frequency conversion, the measuring signal of the selected ASIPM-channel is transferred to the computer and can be displayed on the CDU's display.

If necessary, the eighth signal channel can be subcommutated: channels with 1/8 of sample rate are available.

The sample and hold, multiplexer and A/D-card can be changed by a card with 18 voltage controlled oscillators (VCO) for a frequency multiplex operation. This guarantees the compatibility of available analog telemetry devices.

Fig. 7 shows an ASIPM together with a programmable signal conditioning card.

3.3 DIGITAL DATA-IN-PROCESSOR MODULE (DDIPM)

The DDIPM, shown in fig. 8, contains 8 x 12 digital inputs. The transfer of the digital data to the shift register occurs simultaneously for all modules via the sample and hold-clock. To read the digital data, the DDIPM is addressed using a 4 bit address. Subsequently the data are serially clocked out by the PCM-clock.
The E-Bus consists of 7 twisted pair wires.

3.4 PCM-CONTROLLER ENCODER (PCMCE)

The PCM Controller Encoder has four essential functions:

- Central, simultaneous control of signal sampling in the E-Bus-modules
- Addressing of the single modules (random access)
- Reading of serial data from the single modules
- Inserting of sync words.

The "schedule" for the data transfer is written into the memory of the PCMCE once by the computer before beginning data acquisition, as is the information concerning the number of connected modules, the sample rate and the frame composition. The PCM-frame, shown in fig. 9, is very simple in structure as a result of the modular construction of PMFTIS. The beginning of the frame is characterized by two sync word. At the beginning of the frame all analog and digital signals are sampled and written into the buffers. The first eight data words from M frames contain the time, date and status information, such as:

- Type of measuring signal
- Calibration data of sensor
- Gain
- Offset
- Filtering

for each channel

- Test specific data
  (Pilot, test-vehicle etc.)

The first eight data words are followed by eight data words for each of the N-data acquisition modules, connected to the E-Bus.

In fig. 9 channel 8 of module 1 is subcommutated with 1/8 of sampling rate.

With the wordlength of 13 bit (12 bit information + 1 parity bit) and the sample frequency f_s , the bitrate is:

$$R = f_s \cdot [(N+1) \cdot 8 + 2] \cdot 13 \text{ bit.}$$

This bitrate, which is dependent only of the number of modules (N+1) and the sample frequency f_s, must be generated by a special method of clock processing.

3.5 CLOCK PROCESSING

In the field of flight test techniques the data of several flight test vehicles and ground devices, e.g. for position measurement, are often measured in different flight test instrumentation systems. If the data sampling in these instruments is not synchronized, the offline computation of synchronized data is quite difficult.

The time synchronization of PMFTIS was therefore an essential condition for the development of the system. In addition, the data evaluation should be further simplified through simultaneous sampling of all signals. A further requirement was the external synchronization of sampling, e.g. phase-synchronously to helicopter-rotor-cycle.

The clock processing shown in fig. 10 is based on these requirements. The Clock Processing Unit (CPU) is controlled by the computer. The value of the sample frequency f_s and the number of modules N are written into the registers of CPU once before beginning data acquisition.

In the case of time synchronous sampling, f_s will be generated by a 10 KHz clock derived from the 5 MHz main oscillator with the help of the programmable frequency divider. By resetting this divider each second, a phase-locked allocation to the clock is achieved.

In accordance with the formula for PCM-clock frequency R in section 3.4, f_s will first be multiplied by the factor 26 in a Phase-Locked Loop (PLL)-circuit.

By means of a further frequency multiplication in the range of 9 corresponding to N_{min} = 1 up to 65 corresponding to N_{max} = 15 the clock frequency R is generated. By connecting the sample clock input to an external clock generator, the sampling synchronity to an external clock can be achieved.

3.6 TIME CODE GENERATOR (TCG)

The TCG is driven from the main processor clock, which has a stability of better than 5×10^{-9} per day. The phase synchronization and the reading of date and time information occurs automatically after connection to the external mother clock. For FMMPX-operation the IRIG-B-Code is generated. In this case the status information of the modules is transmitted with the help of the 45 information bits of the IRIG-B-Code.

3.7 AIRBORNE COMPUTER SYSTEM (ACS)

The Airborne Computer System is based on microcomputer type SBP 9900. It is placed in an ARINC-box 1/2 ATR (fig. 11) and contains the following components:

- Contral Processor Unit
- Memory (16 K-words)
- Parallel I/O
- C-Bus-Interface
- Terminal Interface

The programming is carried out in program language PASCAL.

3.7.1 CONTROL AND DISPLAY UNIT (CDU)

The Control- and Display Unit (fig. 11) contains an alpha-numerical display with 4 lines of 16 characters each and a 4 x 4 keyboard. Of these 16 keys, 10 serve to enter numbers, 2 for cursor positioning and 4 for the control of special functions.

The programming is carried out block-wise. Each program block is characterized by a special page in the display.

4. PMFTIS APPLICATION

The following example will illustrate the flexibility of PMFTIS. Fig. 12 shows the data acquisition with a rotor-synchronous sampling of measuring signals.

The whole measuring device consists of a special rotor flight test instrumentation system (RFTIS) for measuring strain gauge signals and a PMFTIS for data acquisition of measuring signals on board the helicopter.

The PMFTIS receives angle-synchronous pulses of a rotor shaft angle encoder for generating the sample rate and the information of the zero position of the rotor. The angle-synchronous sample pulses are transferred from PMFTIS to RFTIS by a slip ring. The output data of RFTIS are transferred serially to PMFTIS by a further slip ring.

With the help of this measuring principle a dynamic analysis of rotor- and helicopter dynamics in relation to the rotor angle position is possible.

5. CONCLUSIONS

The Programmable Multipurpose Flight Test Instrumentation System presented here is the answer to the annually increasing complexity of flight test techniques. Through the modular construction of this device a maximum of flexibility for a wide field of applications is achieved.

Important aspects in the development of this device were the compatibility with available analog flight test devices, the simultaneous sampling of all measuring signals, the external synchronization of sample rate and the interactive programming.

The essential data of PFMTIS are summarized in table 1. The high accuracy of the analog section and the high main processor clock stability are especially worth emphasizing. The PMFTIS is based on a total conception for real time data processing. Thus it is guaranteed that the test results are available for the experimentator directly or immediately after flight tests. This leads to a significant reduction in costs of flight tests.

LIST OF SYMBOLS

PMFTIS	Programmable Multipurpose Flight Test Instrumentation System
FMMPX	Frequency Multiplex
PCM	Pulse Code Modulation
RTS	Rotor Test Stand
INS	Inflight Simulator
TIFS	Total Inflight Simulator
PI	Parameter Identification
DRT	Dynamic Rotor Test
OLGA	Open Loop Gust Alleviation System
HQ	Handling Qualities
ACT	Active Control Technology
ASIPM	Analog Signal-In Processor Module
DDIPM	Digital Data In Processor Module
PCMCE	PCM Controller Encoder
PCMCD	PCM Controller Decoder
TCG	Time Code Generator
ISM	Integrated Sensor Module
CDU	Control and Display Unit
VCO	Voltage Controlled Oscillator
CPU	Clock Processing Unit
ACS	Airborne Computer System
RFTIS	Rotor-Flight Test Instrumentation System

REFERENCES

1. Hamel, P.G. — "Aircraft Parameter Identification Methods And Their Applications-Survey and Future Aspects", AGARD-LSP-104, 1979

2. Bartoli, Felo — "An Advanced Airborne Data Acquisition System", AGARD-CP-223, 1976

3. Statler, I.-C. — "Flight Mechanics - A Review of the Activities of the AGARD Flight Mechanics Panel" AGARD Highlights 78/1 (1978)

4. Lunn, K., Hutchinson, C. — "Realization of a Helicopter-Oriented-Real-Time Data System for Research, Experimental, and Prototyp Flight Testing". American Helicopter Society, Reprint No. 79-50, May 1979

5. Wilhelm, K., Gmelin, B. — "DFVLR-Dynamic Model Testing in Wind Tunnels for Activ Controls Research" ICAS, Munich, W-Germany, Oct. 1980

6. Pausder, H.-J., Gmelin, B.L. — "Flight Test Results For Task Orientated Flying Qualities Evaluation" American Helicopter Society, Reprint No. 80-29, May 1980

7. Sanders, K., Pausder, H.-J., Hummes, D. — "Flight Tests And Statistical Data Analysis For Flying Qualities Investigations" Sixth European Rotorcraft And Powered Lift Aircraft Forum, Bristol, England, Sept. 1980

8. Hanke, D., Lange, H.-H. — "In Flight Handling Qualities Investigation Of Various Longitudinal Short Term Dynamics And Direct Lift Control. Combinations For Flight Path Tracking Using DFVLR HFB 320 Variable Stability Aircraft" AGARD-CP-260, Stability and Control, Ottawa, Sept. 1978

9. Lange, H.-H., Hanke, D. — "Use Of DFVLR In-Flight Simulator HFB 320 HANSA For Handling Qualities Investigations" ICAS, Munich, W-Germany, Oct. 1980

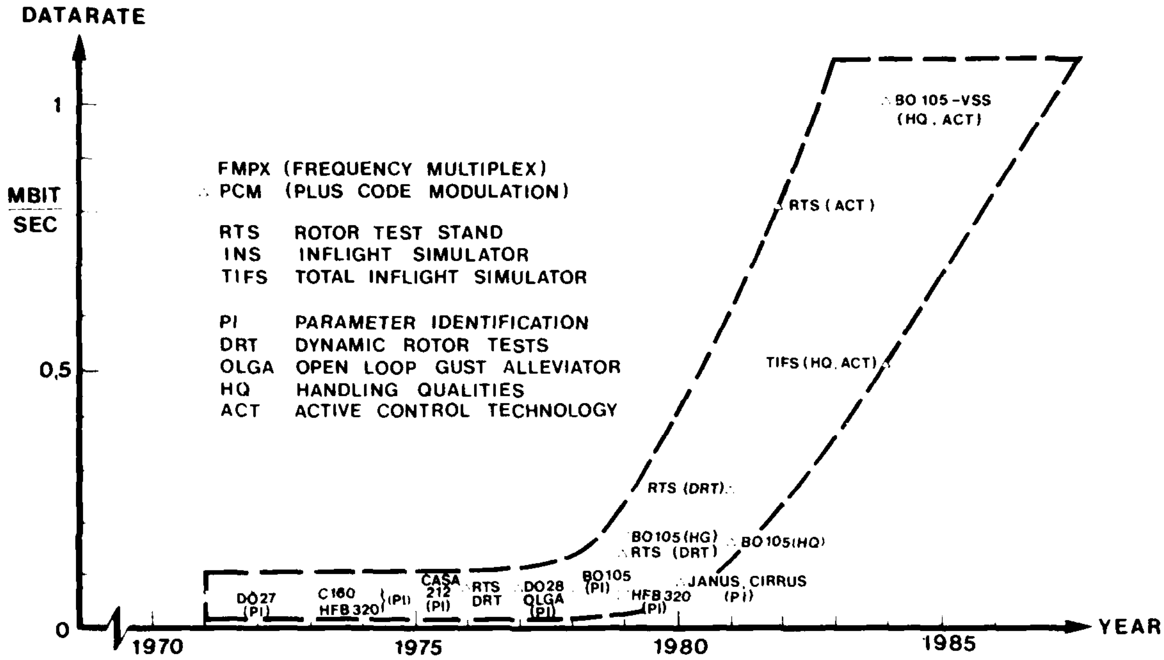


Fig.1 Data rates for windtunnel and flight testing at the DFVLR Institut für Flugmechanik

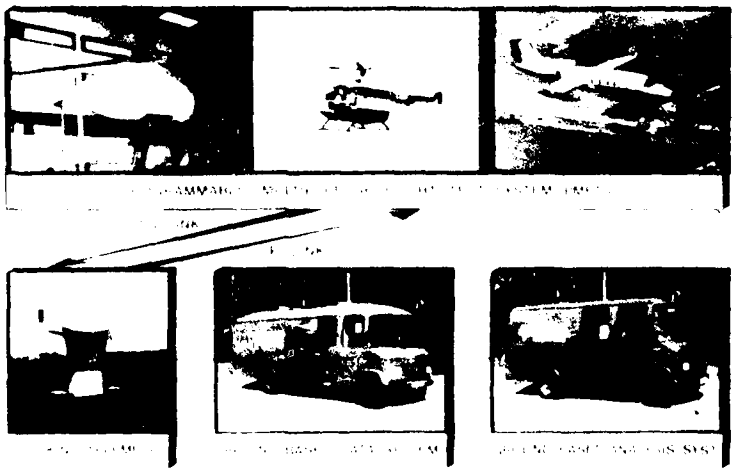


Fig.2 Application flexibility of PMFTIS

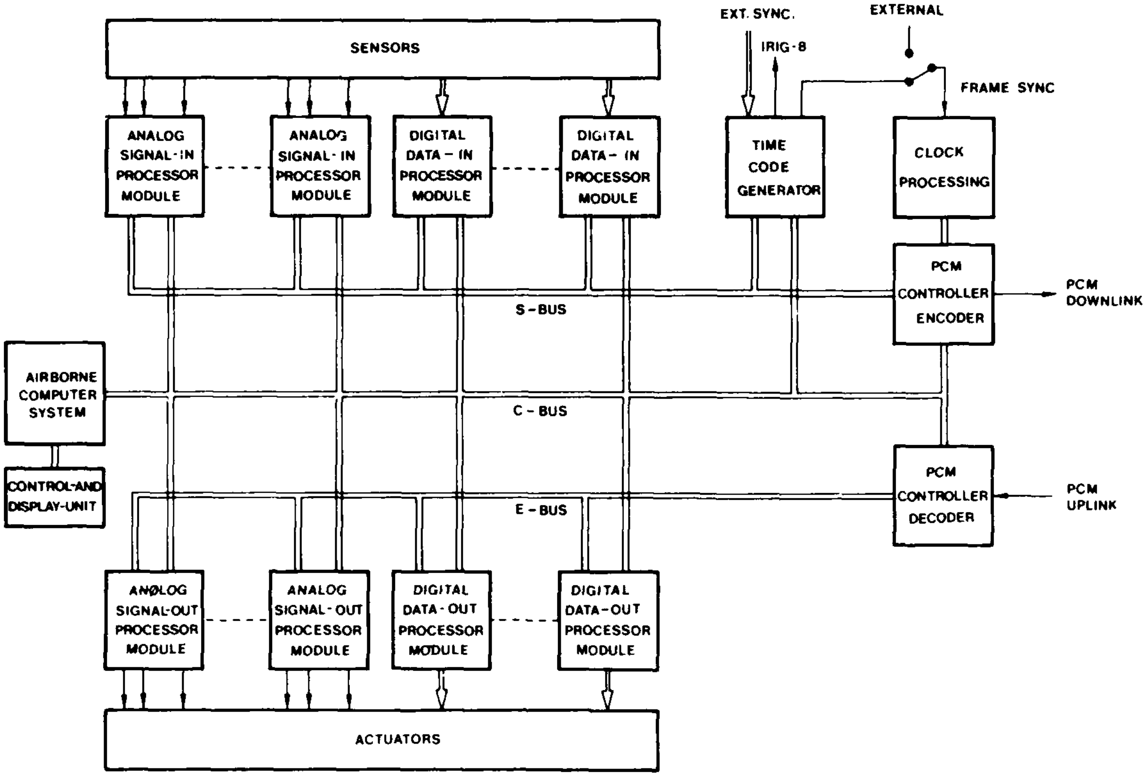


Fig.3 Block diagram of programmable multipurpose flight test instrumentation system (PMFTIS)

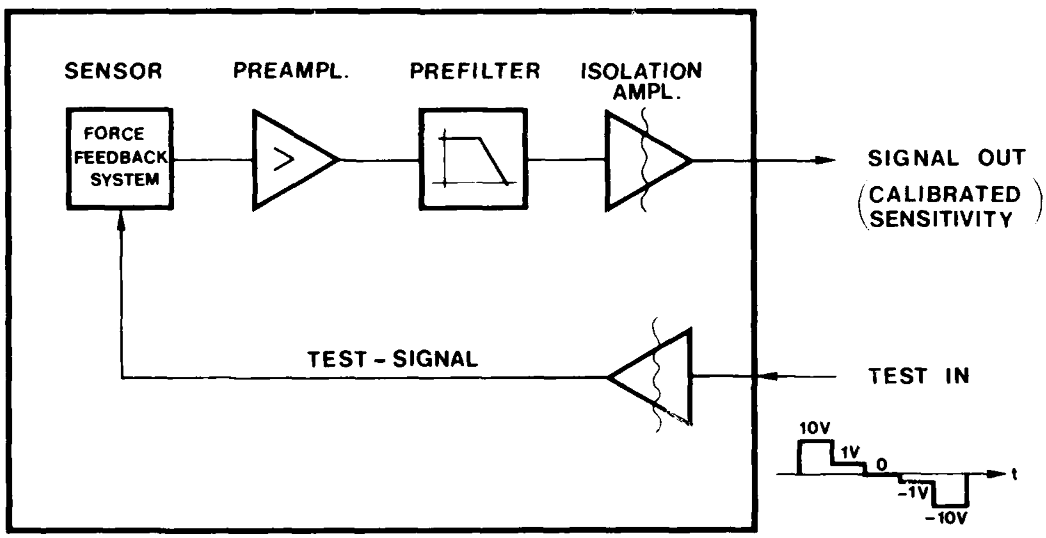


Fig.4 Integrated sensor module (ISM)

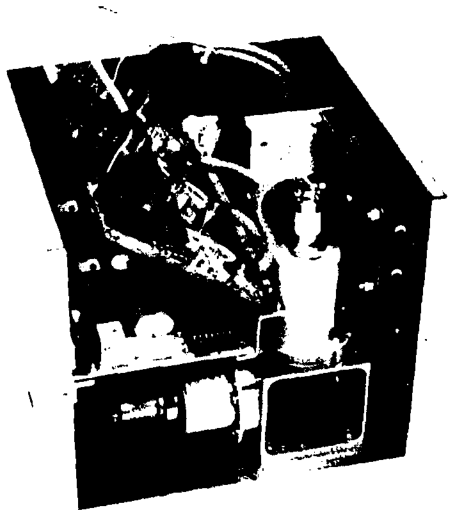


Fig.5 Three axis accelerometer ISM

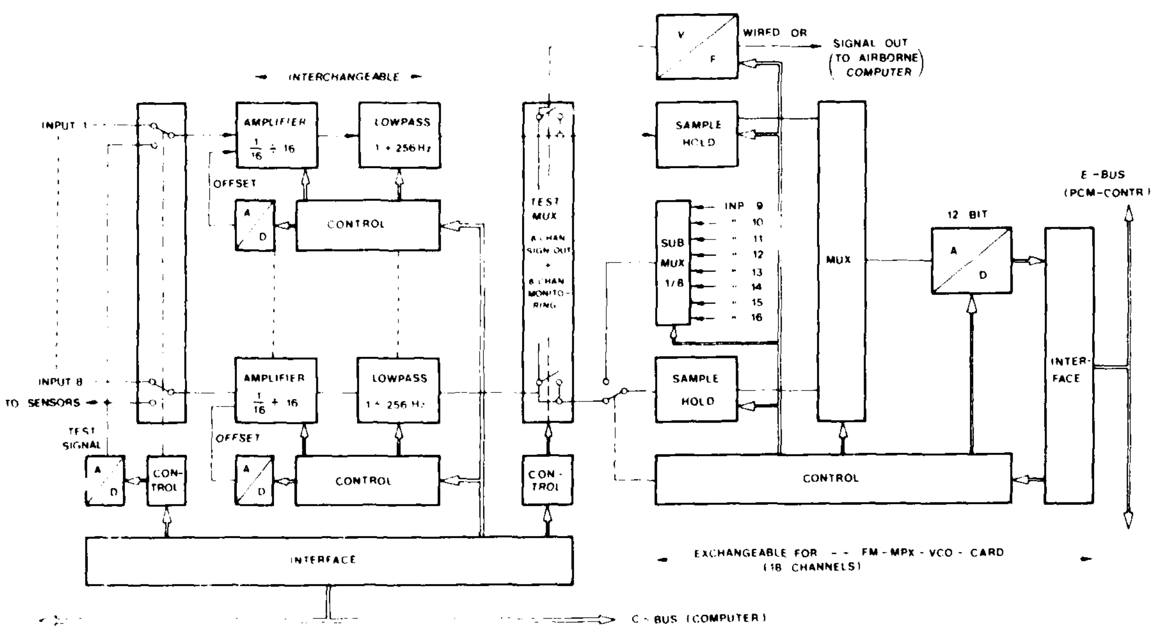


Fig.6 Block diagram of analog signal-in processor module (ASIPM)

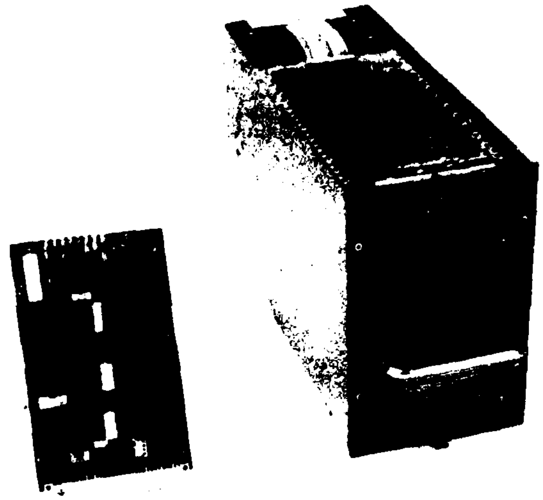


Fig.7 ASIPM and programmable signal conditioning card

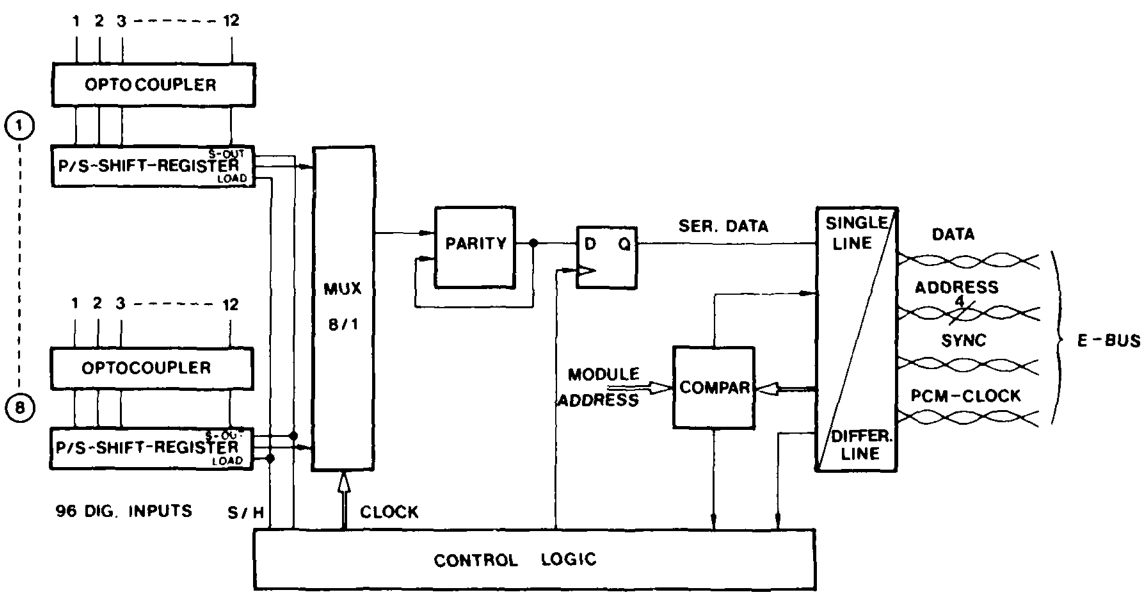


Fig.8 Digital data-in processor module (DDIPM)

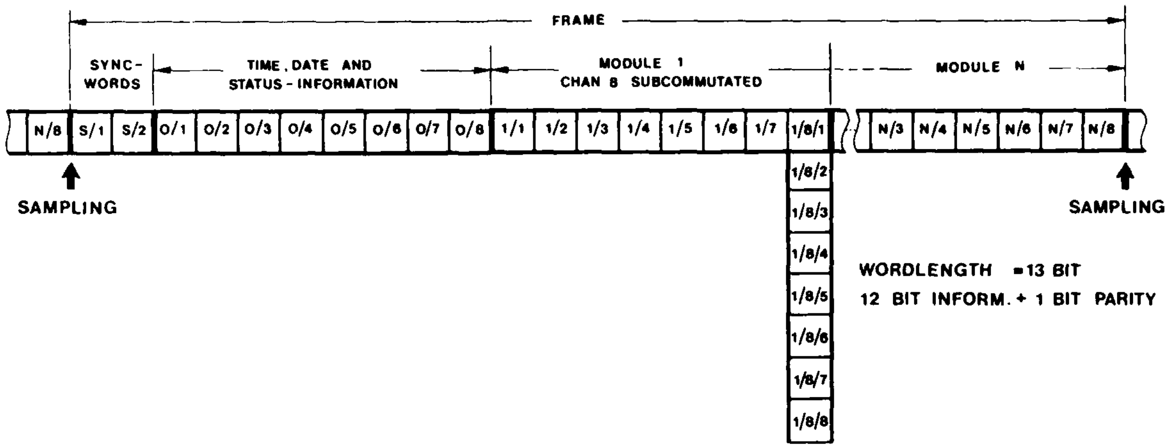


Fig.9 Example of PCM-frame

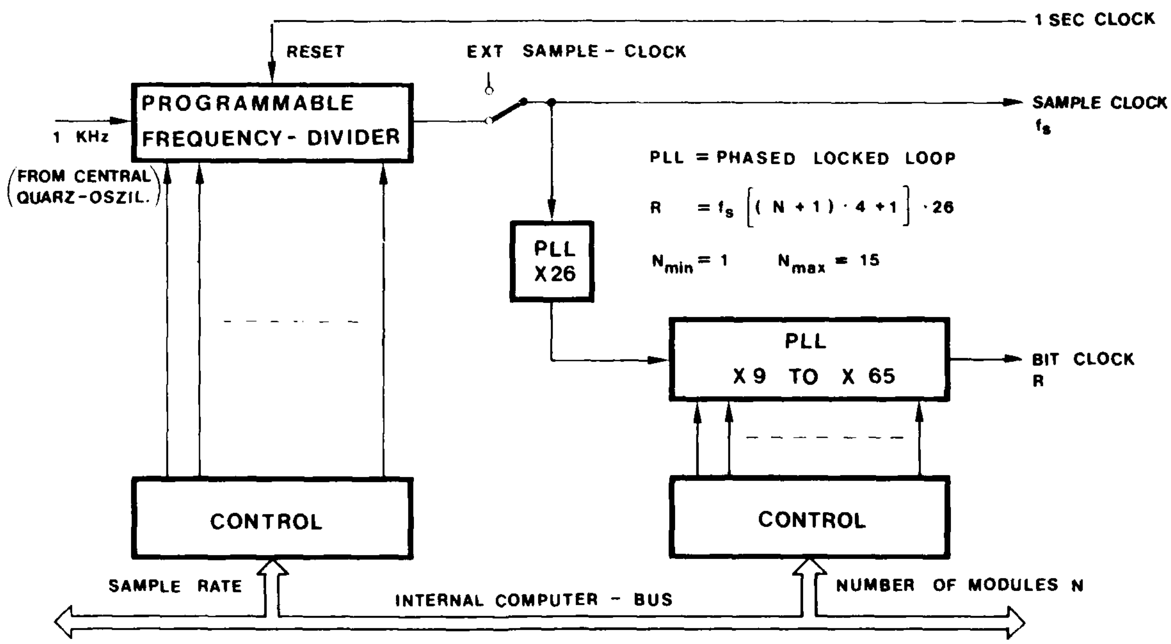


Fig.10 Block diagram clock processing

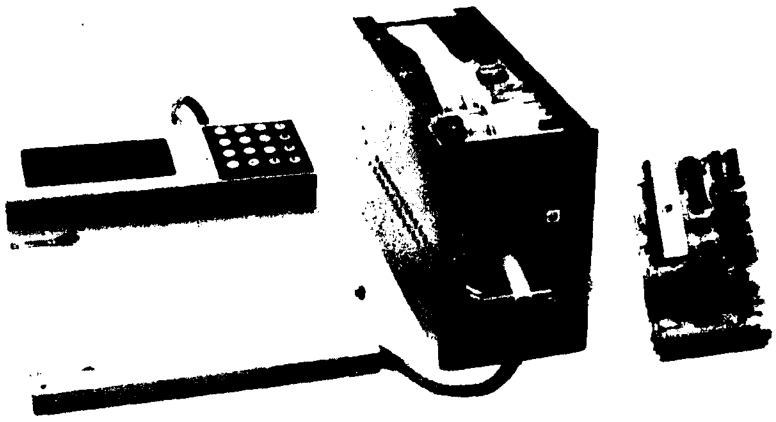


Fig.11 Airborne computer system (ACS) together with control and display unit (CDU)

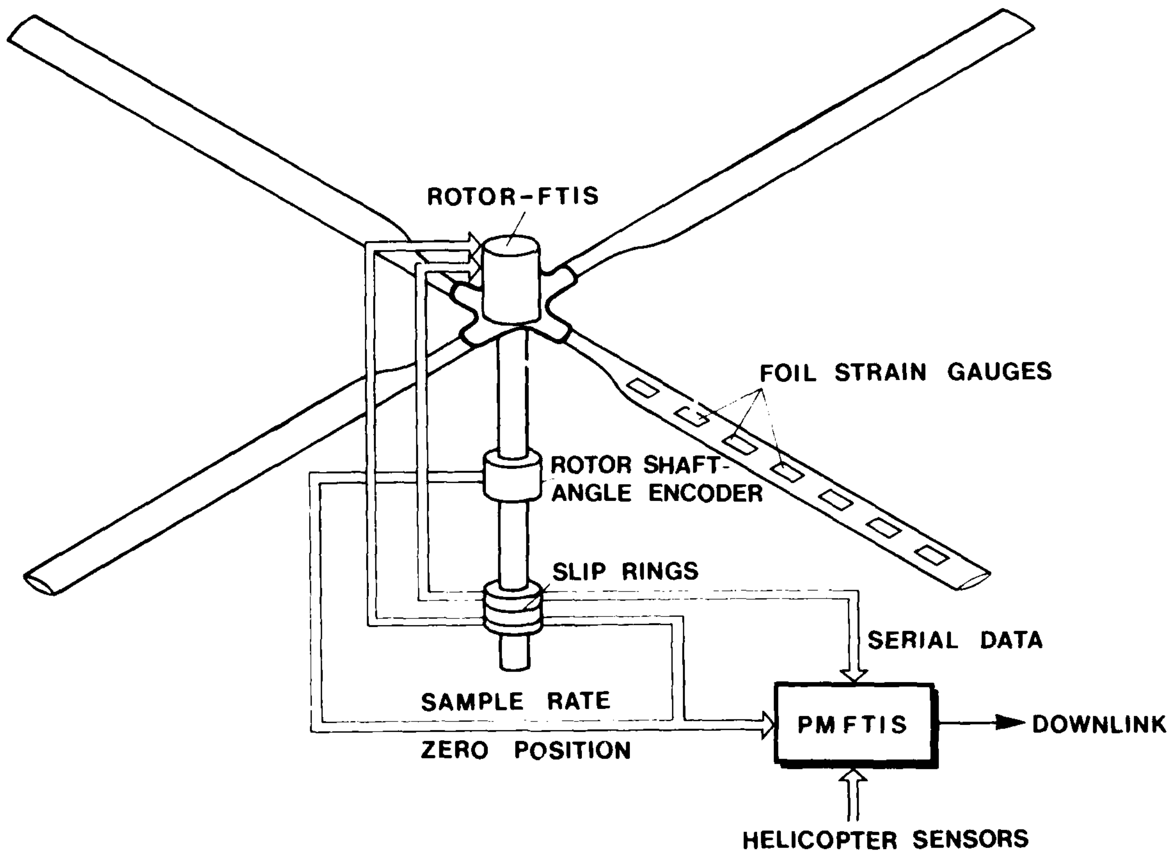


Fig.12 Rotorsynchroneous data sampling

RECENT TRENDS IN FLIGHT TEST SIGNAL CONDITIONING

DONALD W. VEATCH
Aerospace Instrumentation Consultant
Star Route 2, Box 1143, Tehachapi CA 93561, USA

ABSTRACT

Radical changes are pervading the whole data acquisition process. These changes are so dramatic in many areas that the impact on signal conditioning is often not being properly evaluated. This paper will cover some of the changes in techniques being brought about by this new technology as well as some of the more interesting hardware developments for specific signal conditioning tasks.

INTRODUCTION

Many of the recent advances in signal conditioning technology have been the result of the development of the silicon chip integrated circuit. This development has made possible extremely sophisticated systems that require almost negligible space and power. Also in recent times when the trend has been towards rapidly increasing costs, electronic components are the only example that comes readily to mind where costs have dropped dramatically. Further justification for the use of integrated circuitry in signal conditioning is that it is extremely reliable. The characteristics of small size, low power consumption, high reliability and low cost come close to the flight test instrumentation engineers' design constraints of no space, no power drain, no problems, and little money. It therefore should come as no surprise that integrated circuits are changing signal conditioning technology.

DEFINITIONS

Before becoming too involved in recent signal conditioning trends it is desirable to define what flight test signal conditioning encompasses. Flight Test Signal Conditioning is defined in AGARDograph No. 160, Volume 1 (Ref. 1) as "all signal modifications between the transducer and the input stage of the recording or telemetry system, the indicator or the airborne computer." Signal conditioning is also a way to optimize the match of the transducers to the aircraft terminal devices, e.g., tape recorder, transmitter, etc. Properly done, signal conditioning greatly simplifies airborne data acquisition and ground data reduction tasks. For example, Pulse Code Modulation (PCM) signal conditioning makes it possible to place many channels of data on a single tape track or on a single telemetry transmitter channel and also match the data conveniently to modern computerized ground stations.

THE INSTRUMENTATION AMPLIFIER/PRE-SAMPLING FILTER

To illustrate how silicon chip integrated circuits have changed signal conditioning technology a combined instrumentation amplifier and pre-sampling filter will be discussed. What has been done with the amplifier/filter could also be done for signal conditioning circuits such as the charge amplifier, current-to-voltage amplifier, etc.; however, many of the special circuits required by the instrumentation engineer are not generally available at a reasonable cost. Figure 1 compares a transistorized instrumentation amplifier/pre-sampling filter with its counterpart constructed using hybrid thick-film techniques. The transistorized amplifier was a very significant advance over the previous large, inefficient and unreliable electron tube amplifier in terms of cost, power consumption, size, etc. The hybrid thick-film technology has provided another major size reduction. Both amplifiers of Figure 1 include provision for voltage gain and offset adjustments and changing filter cutoff frequency. Figure 2 is the same amplifier using thin-film techniques. Normally a small reduction in size might be expected using this technique; however, in this instance it was decided to considerably enhance the pre-sampling filter section. The larger of the two metallic packages in Figure 2 is the pre-sampling filter and the smaller is the instrumentation amplifier. These packages are mounted on a printed circuit board which provides terminals for adjusting the amplifier's voltage gain and offset and the filter's cutoff frequency. Figure 3 shows the basic hybrid thin-film amplifier with the cover removed.

Eglin Air Force Base, Florida, is using a similar commercially available amplifier/filter which has some very desirable features. Figure 4 is a simplified schematic diagram of this unit. The input amplifier is voltage gain programmable from 0.1 to 1000 and the offset is adjustable to $\pm$ 100 percent of full scale. It has a differential input impedance greater than 100 megohms and a common mode rejection ratio of 100 dB minimum at a gain of 1000. The filter section is a 6 pole Butterworth low pass filter factory set at a cutoff frequency of 5Hz. The cutoff frequency can be adjusted from 5Hz up to 5kHz by means of six external resistors. The size is 0.5 cm high, 2 cm wide and 3.2 cm long. The unit will mount directly to a printed circuit board by means of pins located on its base. This unit and the previous unit are similar in size and capabilities and represent what should soon be readily available at a reasonable cost.

AN ECONOMICAL MICROCIRCUITS LABORATORY

Almost any signal conditioning circuit can be designed into a hybrid microelectronics package; unfortunately, unless the demand is large enough, the particular circuit required may not be generally available, and small quantities of special-order hybrid thin-film microcircuits can be prohibitively expensive. Laboratories which can produce silicon chip circuits and to a lesser extent thick-film and thin-film hybrid circuits are very expensive. If you do not have these types of facilities there is a fairly economical alternative. For example, the NASA Dryden Flight Research Center (DFRC) in California constructed an economical hybrid circuits laboratory for their own use. The laboratory was maintained by one part-time technician. After a short training course the technician and the Center's electronic design engineers could use the facility to construct signal conditioning circuits as required and it was easier than producing similar circuits on printed circuit boards. The total cost for this laboratory was less than the printed circuit facility already in existence at NASA-DFRC. For this economy laboratory, only silicon circuit chips, chip resistors and chip capacitors were used. The desired chips were bonded to a ceramic substrate and the chips were then wired together by a wire bonding machine. Careful planning of the chip placement on the ceramic substrate is critical in order to minimize the number of cross-overs of the minute wire jumpers. Very reliable signal conditioning can be produced this way, but the results look very unsophisticated internally when contrasted to the production unit that was shown in Figure 3.

This technique is only economical for producing a small number of units.

PCM ENCODERS

Silicon chip integrated circuit technology has very significantly impacted the electrically complex encoder signal conditioners such as PCM systems. At the present time the typical airborne PCM encoder used in the United States uses Dual Inline Package (DIP) technology. PCM encoders using medium scale integrated circuits in Dual Inline Packages are adequately sized for most applications. An example of one such unit is shown in Figure 5. A printed circuit card from this system using DIP technology is shown in Figure 6. Presently these types of systems are readily available with subcommutation that will encode a very large number of data channels. Many of these airborne PCM encoders can accept data in more ways than most ground stations have the ability to acquire the data. For example many systems can change bit rates, word lengths, frame lengths, etc., as required by aircraft flight test conditions. This exceeds most of the ground stations' capabilities to maintain synchronization.

Why would any flight test group ever want an even smaller PCM encoder than the present DIP technology type PCM systems? The subscale Remotely Piloted Vehicle (RPV) programs have generated such a requirement at NASA-DFRC. These subscale RPV's which are used to perform high risk flight test missions such as spin research and buffet studies on existing types of aircraft shapes or investigations on new concepts such as skew-winged aircraft often require truly small signal conditioning systems. A miniature PCM used at NASA-DFRC for this purpose is shown in Figure 7 and as shown, can encode 128 input data channels. The end plate has been removed from the encoder to show the ultra-violet Erasable Programmable Read Only Memory (EPROM) which allows the user to easily change formatting and timing programs. The mating connectors are supplied by the manufacturer and include short wiring pigtails of 28 gauge wire. This creates a problem in that in general practice, most aircraft use larger gauge wires. These wires further are often shielded-twisted-pair wires, where every shield usually has a wire connected to the PCM encoder ground. Thus, the final wiring fan-out can be overwhelming. In practice the total signal conditioning module is usually wired in the shops, not on the aircraft.

These small PCM units which use thin-film technology have proven to be very reliable and are becoming economically competitive with their DIP technology counterparts. The larger DIP technology systems at present offer more system variety and flexibility; however, this gap is rapidly being narrowed.

While only the PCM encoders have been discussed, Frequency Modulation (FM) encoders are readily available in thin-film hybrid modules. These FM modules are available that conform to the Inter Range Instrumentation Group (IRIG) specifications for proportional, constant bandwidth and wideband FM encoders.

CHANGING APPLICATION PHILOSOPHIES

The use of large scale integrated silicon chip circuits will continue to impact the signal conditioning art; however, of equal importance is the fact that these advances are changing the philosophy of how the signal conditioning is used. The reason that enhanced reliability and a significant reduction in signal conditioning size necessitates a change in the way signal conditioning is applied is that experience has shown that appreciable amounts of time and money can be saved by such changes.

To illustrate these changes, first consider that traditionally the signal conditioning has been centrally located in one easily accessible area and in this same area typically would also be found most of the other instrumentation equipment such as power supplies, tape recorders, junction panels, etc. The transducers would be located throughout the aircraft where required and then their wiring routed through the aircraft to this central location. In the larger vehicles this central area was between flights a center of activity with the instrumentation people patching in new channels, tracing problems, replacing defective equipment, calibrating data channels, etc. In addition the aircraft mechanics would be working on the aircraft itself. All this activity led to considerable congestion and confusion. In smaller aircraft a single access hatch to the centrally located instrumentation bay was often responsible for long delays between flights. The concept of centrally locating the instrumentation was mandated mainly by the large size and the required maintenance of these types of equipment.

In the DC-10 flight certification program, the Douglas Aircraft Co. (DAC) implemented an improved de-centralized approach. The following example involved a large commercial passenger aircraft which required over 700 input data channels. In this flight test program the signal conditioning including the PCM encoders were located near the centroid of each concentrated instrumentation area: On each engine, in each wing, in the tail, and in the cabin. See Figure 8. To further simplify the system some of the transducer signal conditioning was placed on plug-in cards integral to the PCM Remote Multiplexer and Digitizing Units (RMDU's). This considerably reduced the number of instrumentation boxes with their interfacing connectors and wiring harnesses. Thus, a good percentage of the transducers were terminated directly at the remotely located PCM RMDU's. These RMDU's were connected by means of a simple "data bus" to a centrally located data management unit. The purpose of the data management unit was to request data from the various remote PCM RMDU's and interweave the resultant data into the formats required by the various aircraft data terminal devices such as tape recorders, transmitters, crew displays, etc.

By using a distributed signal conditioning system, as contrasted to the centralized signal conditioning system, the Douglas Aircraft Company estimated (Refs. 2 and 3) that they reduced the length of wiring from 62,500 meters to 7,600 meters and saved over $55,000.00 in wire costs alone. Additionally this distributed type of system saved an estimated 3,900 installation manhours and reduced the instrumentation weight by more than 14,000 Newtons. The money saved on the installation was estimated to have paid for all of the remote multiplexing system hardware. Thus by a change in installation philosophy one company has helped finance their new PCM data system. Similar savings can be expected in a 700 channel data acquisition system used on fighter type aircraft. In this case the wiring runs would be much shorter; but, the de-centralized approach avoids the task of running large bundles of wire in confined spaces which often requires an extensive and costly manpower effort.

In general the distributed type of data acquisition system is desirable mainly for the larger instrumentation tasks, e.g. larger than 100 to 300 data channels. Unfortunately no distributed type data system is presently available which has all the desired features. The major impediment to the development of the distributed systems seems to be that there is no consistent demand for the large instrumentation systems. A solution to this problem may well be for the data users to make the manufacturers aware that a reasonable number of changes in the initial design of a small central type PCM system can make the unit function also as a remote encoder controllable by means of a "data bus" from a central data management unit.

FIBER OPTICS WIRING

An interesting innovation, a by-product of the DAC's distributed encoding system, was the application of fiber optics wiring. The PCM RMDU's and tne data management unit communications took place at a 2 megabit rate. When the wires connecting the RMDU's to the central data management units were over 100 to 200 feet long, special and expensive high frequency wire was required. This wire is being replaced by a readily available fiber optics strand for the follow-on instrumentation tasks. Optical connectors are readily available as are the electrical-to-optical transmitters and the optical-to-electrical receivers. The techniques for using fiber optic "wiring" are well developed and involve no major outlay of money. Fiber optics "wiring" can be very useful where high frequencies have to be transmitted for any distance and where voltage isolation or electromagnetic interference are major problems. As always, care must be taken in introducing new procedures to insure that the problems associated with the new techniques are fully appreciated at all levels of the installation process.

DIGITAL DATA SIGNAL CONDITIONERS

A trend which is a direct result of the progress made in microelectronics is the startling growth of the amount of digital data being processed by flight data acquisition systems. Some instrumentation groups estimate that over 90 percent of the airborne data is either digitized in the data acquisition system or presented to the system as digitized data. This trend has produced a large number of digital signal conditioners which are here classified as formatting, microprocessor, microcomputer and minicomputer signal conditioners. The above order of classification reflects the general order of increasing computer-like capabilities of each system. The boundaries between the bottom of one class and the top of the next class involves considerable overlap. For example, the most powerful of the microcomputers often equal or exceed the capabilities (e.g., speed, flexibility, word size, software libraries, etc.) of the more limited minicomputers. Formatting signal conditioning has no computational capability but can have considerable ability to sort, rearrange and select digital data.

FORMATTING SIGNAL CONDITIONERS

One recent trend which is impacting signal conditioning is the rapid advancement of avionics subsystems instrumentation and the linking of these avionics subsystems by means of a data bus. By definition these avionics systems are aircraft control systems and not as such, a part of the data acquisition system. Two considerations, however, have caused them to heavily impact the data acquisition signal conditioning engineer. The first consideration is that these systems have become extremely complex; therefore, the avionics engineers, especially in the first stages of flight test, often want to record all the information on the data bus to aid them in problem analysis. The second consideration is that systems such as the Inertial Navigation Systems, Air Data Computer Systems, etc., have become highly advanced and the data bus contains much excellent data which the flight test engineer can use.

Figure 9 shows how the F-16 Advanced Fighter Technology Integrator (AFTI) aircraft flight test system meets both types of demands with formatting type signal conditioners. (The F-16 AFTI is a joint U.S. Air Force - NASA program which is being implemented by General Dynamics Corporation). The Avionics Bus Interface 1 satisfies the requirements of the avionics design group to record all the information which occurs on Buses A and B. The data bus used on this aircraft is the Military Standard MIL-STD-1553B data bus (Ref. 4). Three types of twenty-bit words are permissible on the two data buses A and B. These are: Command Words, Data Words and Status Words. A word on one bus can overlap a word on the other bus and there are many times when no words exist on either bus. The Avionics Bus Interface 1 is a simple formatting type signal conditioning system that generates twenty-bit psuedo data words for insertion onto the tape channel until a data bus word is available. When a data bus word arrives it is inserted into the data stream as soon as a twenty-bit pseudo data word is completed. If a word on the alternate bus should overlap another word it is delayed until the end of the first word and then it also is inserted into the data stream: Thus all words on both buses are recorded with only minor time shifts.

The Avionics Bus Interface II is considerably more sophisticated. In this system, the data bus which generates words asynchronously has to be matched to a continuous repetitive PCM data stream functioning at a different bit rate. Figure 10 is a block diagram of this system. The bus receiver takes the words off the two buses and forwards one bit to identify from which bus the word was derived. The Decoder changes the code format from Manchester II bi-phase level to a non-return to zero (NRZ) format, removes the synchronization word which involves three bit times, and also removes the parity bit. The resulting sixteen-bit serial word is passed on to the Buffer Register. The Buffer Register takes the sixteen-bit serial word and constructs a new twenty-bit parallel word which contains a parity bit, one bit to denote the bus from which the word originated, and two bits which identify the type of word involved, i.e., Command Data or Status. Also, this register sends the Command Word to the Word Selector Logic. The Word Selector Logic can be set to recognize requests for up to 256 words and selects which words will be entered into the PCM data stream. The Buffer Register passes all data words on to the Data Register which takes the words in at one rate and clocks them out at a much slower rate since the word is now a twenty-bit parallel word. All words are sent to the Random Access Memory but only the words of interest are written into memory. From the Random Access Memory the data words are selected in the order they are to appear in the PCM data stream and sent to the Buffer Memory, which synchronizes the word rate to that of the PCM system. One interesting feature of the Random Access Memory is that the parity bit is replaced by a bit which notifies the data user if the word in the Random Access Memory has been refreshed since it was last inserted in the data stream. This last Bus Interface takes up one printed circuit card and is a sophisticated example of formatting type signal conditioning.

MICROPROCESSORS AND MICROCOMPUTERS

In the two previous examples there were no requirements for onboard real-time data reduction and thus no real need for the computational capabilities of a computer or microprocessor. However, the DAC flight test instrumentation group has used a microprocessor to accomplish essentially the same bus interface, except the outputs were presented in engineering units and scaled to be recorded on airborne strip chart recorders. In this case where actual data reduction was required, the computational power and flexibility of the microcomputer was desirable.

When should computers be used for airborne signal conditioning? They should simply be used when they can do the task better and cheaper than other approaches. The author's position is that computerized systems should mainly be used when real-time on-board data reduction is required and two-way telemetry, (i.e. raw data on the telemetry down-link and reduced data on the telemetry up-link), is not desirable for whatever reason. An actual example is discussed below, which illustrates how this type of airborne signal conditioning can be used to good advantage.

If a person visits the various flight test facilities they cannot escape the conclusion that the major trend at present in signal conditioning is the rapid increase in the application of microcomputers. Microcomputers are being used for example: To make a small 32 cell pressure manometer which electronically scans and repetitively calibrates each pressure module during flight, to produce a scanning spectrum analyzer, and to change raw data to engineering units for aircraft real-time use (as in the DAC microcomputerized bus interface mentioned above).

The Boeing Commercial Airplane Company (BCAC) has used microcomputers to provide real-time on-board computation of flight test Gross Weight and longitudinal Center-of-Gravity (GW-CG). A block diagram of this system is shown in Figure 11. The Real-Time GW-CG Computing System (Ref. 5), was developed to solve two similar, but different problems. The first problem was that the on-board flight test engineers were often required to know the aircraft's gross weight and/or center-of-gravity to set up a particular flight test routine. Computing GW and CG from the volumetric flow data and other raw data was a difficult and time consuming process involving complex graphic solutions. The present system eliminates this problem entirely and has significantly reduced actual flight hours. The second problem was that before certain data could be reduced in the ground based computers, a complete data tape pass was required to generate the GW-CG profiles into the data base. Now, if data is requested for a certain portion of the flight program the GW-CG information is available with the first data tape pass. This system has made an appreciable reduction in the required ground based computer's data reduction time.

This is a very interesting and well designed system. A complete system uses eleven microcomputers. Each of the five Fuel Weight Flow Encoders and the two Discrete Data Acquisition Units have their own microcomputer. The GW-CG Processor uses three microprocessors. In the GW-CG Processor one microcomputer controls the keyboard interface, video display, message processing, and the GW-CG computing; the two subordinate microcomputers transmit and receive messages via the communications network. The microcomputer in each Fuel Weight Flow Encoder Unit is programmed with a set of very complete calibration equations to provide an accurate computation of Fuel Temperature, Fuel Pressure, Fuel Flow Rate (pounds per hour), Fuel Used (pounds), and several diagnostic parameters. Calibration equations are programmed into the microcomputer for the transducers the Fuel Encoder Unit services, i.e. a turbine flowmeter, a platinum temperature sensor and a pressure sensor. The GW-CG Processor provides the Fuel Encoder Unit with an initialization factor which is derived from a preflight fuel density and temperature measurement. The Discrete Data Acquisition Unit accepts inputs as shown in Figure 11 and outputs information such as the moment changes caused by a change in position of the landing gear and flaps, the composite weight and moment of the aircraft water ballast, and which fuel tank is connected to which engine.

MINICOMPUTERS

BCAC used a powerful minicomputer to save considerable flight time and therefore money during the flight certification program of the 747 SP aircraft (Ref. 6). Figure 12 is a simplified representation of the BCAC computerized flight test data acquisition system. The BCAC standard flight test data acquisition system is shown in Figure 12 above the dashed line. The portion of the system shown below the dashed line represents a very limited version of the BCAC ground monitoring station and also provides pre-flight and post-flight instrumentation analysis capabilities. Since BCAC was testing a large commercial passenger aircraft which had no stringent space or weight restrictions, it was logical for them to move the minicomputer monitor capability onboard the aircraft so that they could use the minicomputer's generous capabilities to monitor and accomplish near real-time analysis of the data in flight. For example BCAC estimates that the on-board minicomputerized monitoring system used on the 747 SP program cut their Flight Load Survey flight time by approximately 20 percent and reduced the Airplane Performance testing by approximately 15 percent to 20 percent. This change was facilitated since BCAC does not fully subscribe to the telemetry principle except for critical flight safety testing, such as flutter flight tests. Their reasoning is summarized as follows: First, BCAC test aircraft are large enough to carry analysis and instrumentation engineers. Second, the Seattle, Washington area, because of surrounding mountains and the prevailing weather patterns, is in a particularly poor telemetry area. Third, BCAC wishes to be flexible in scheduling flight tests at many different geographic locations, which may not have telemetry facilities. BCAC has found this flight testing technique very useful and is using it on all their present test programs. This same approach will be used on the future flight tests of the 767 and 757 aircraft. If other flight test facilities subscribe to BCAC's telemetry philosophies then placing this amount of ground station type computing capabilities in large test aircraft might constitute an emerging trend.

CONCLUSION

In conclusion, it is obvious that miniature reliable electronics are dominating the signal conditioning trends at this time. Complex encoders such as constant bandwidth frequency modulation or PCM systems can be constructed which will fit in the palm of a hand. The data bus which is already linking computerized avionics subsystems together is also providing the flight test engineer with new sources of data. The computer is just beginning to influence new types of airborne signal conditioning systems, and should be the major factor in the changing nature of airborne signal conditioning during the next decade.

REFERENCES

1. Pool, A., and Bosman, D., edited by, "Basic Principles of Flight Test Instrument Engineering, AGARDograh No. 160, page 5.1.

2. Crowley, L.D., "Evolution of the Douglas Flight-Test Data System," International Telemetering Conference Proceedings, Volume X, c., 1976, pp: 186-187.

3. Bartoli, Febo, "Advanced Airborn Data Acquisition System," AGARD Flight Mechanics Panel Symposium on Flight Test Techniques, Porz Wahn, Germany, October 11-14, 1976, page 23-5.

4. Military Standard, Aircraft Internal Time Division Command/Response Multiplex Data Bus, MIL-STD-1553B, 21 September, 1978.

5. Brockman, D.M., "A Flight Test Real-Time GW-CG Computing Systems," International Instrumentation Symposium, ISA, May 5-9, 1980, Seattle, Washington, Volume 26, Part 2, pp: 583-589.

6. Lieberman, William S., "Use of Onboard Real-Time Flight Test Analysis and Monitor System - A Progress Report," 1976 AGARD Conference Proceedings, No. 223, pp: 24-1 to 24-10.

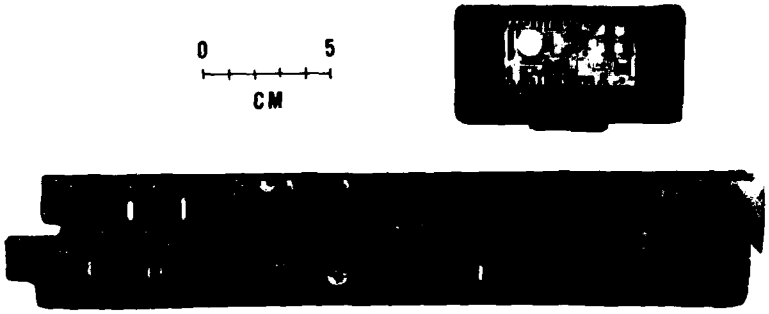


FIGURE 1. TRANSISTORIZED AND HYBRID THICK-FILM AMPLIFIER/FILTER

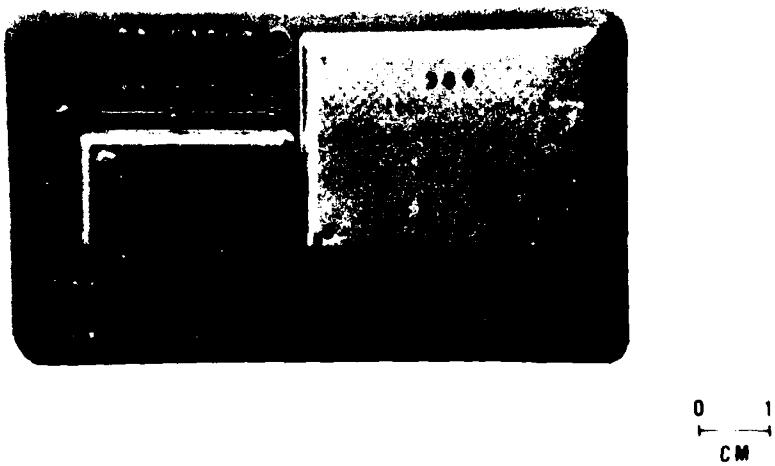


FIGURE 2. HYBRID THIN-FILM AMPLIFIER FILTER

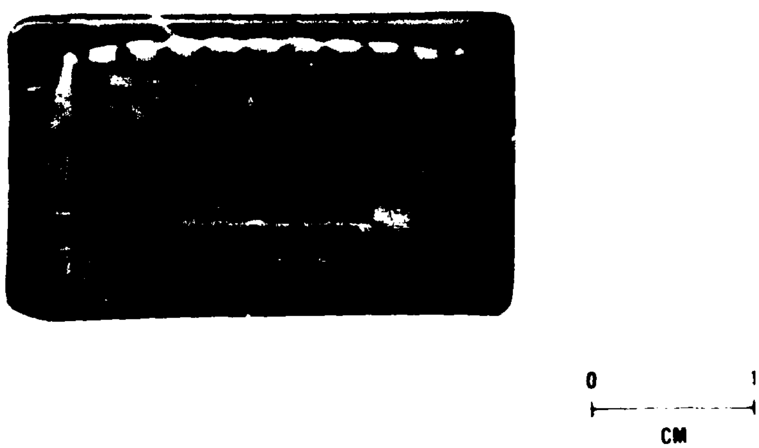


FIGURE 3. HYBRID THIN-FILM AMPLIFIER WITH COVER REMOVED

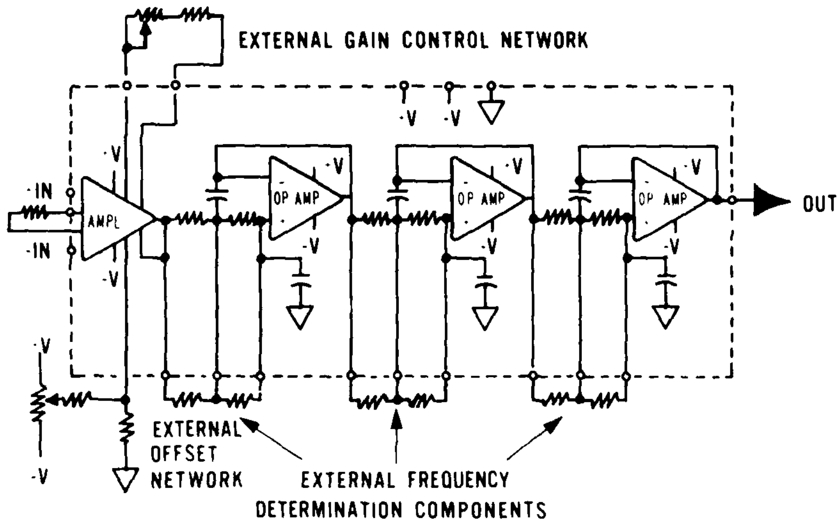


FIGURE 4. BLOCK DIAGRAM OF VOLTAGE SIGNAL CONDITIONING PRE-SAMPLING FILTER

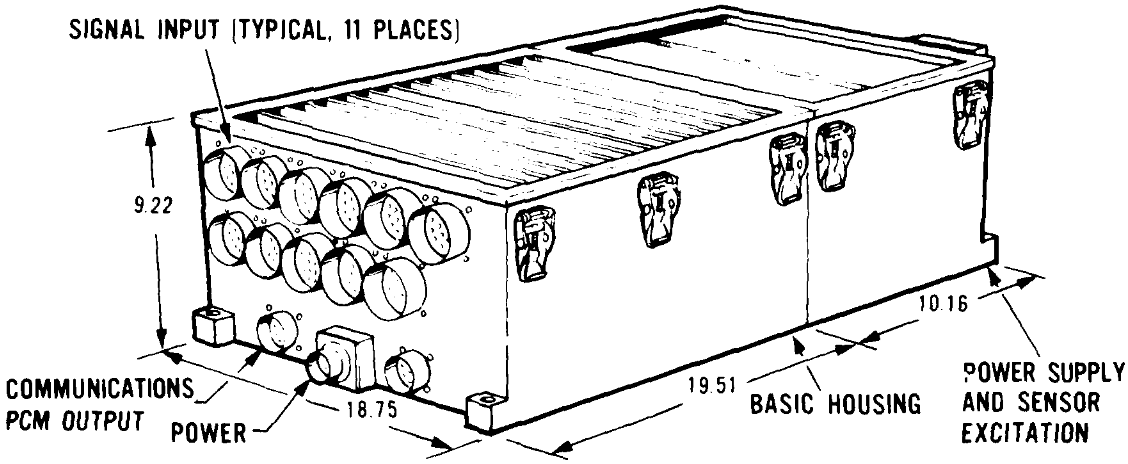


FIGURE 5. PCM MULTIPLEXER USING DIP TECHNOLOGY

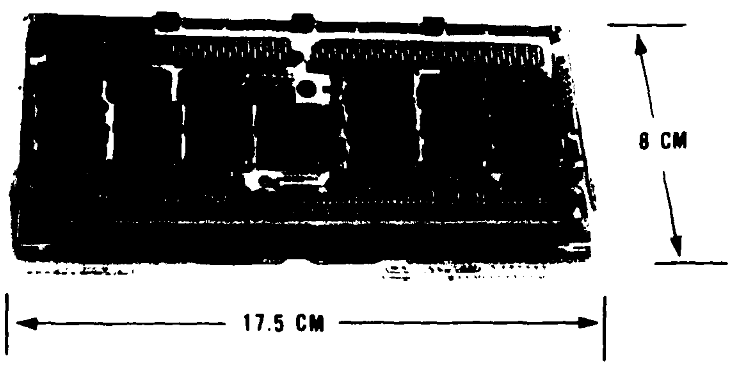


FIGURE 6. DIP TECHNOLOGY PRINTED CIRCUIT BOARD

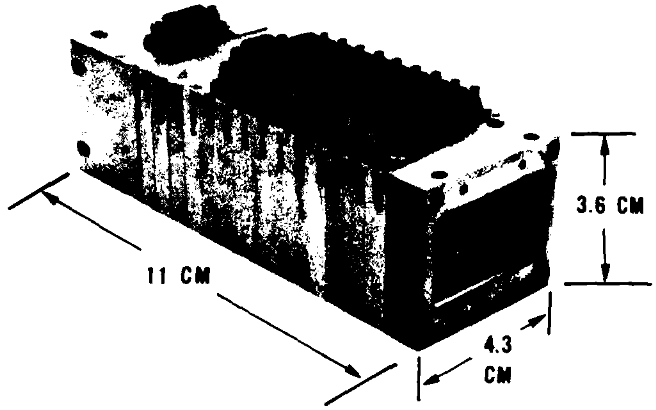


FIGURE 7. PCM SYSTEM USING HYBRID THIN-FILM TECHNOLOGY

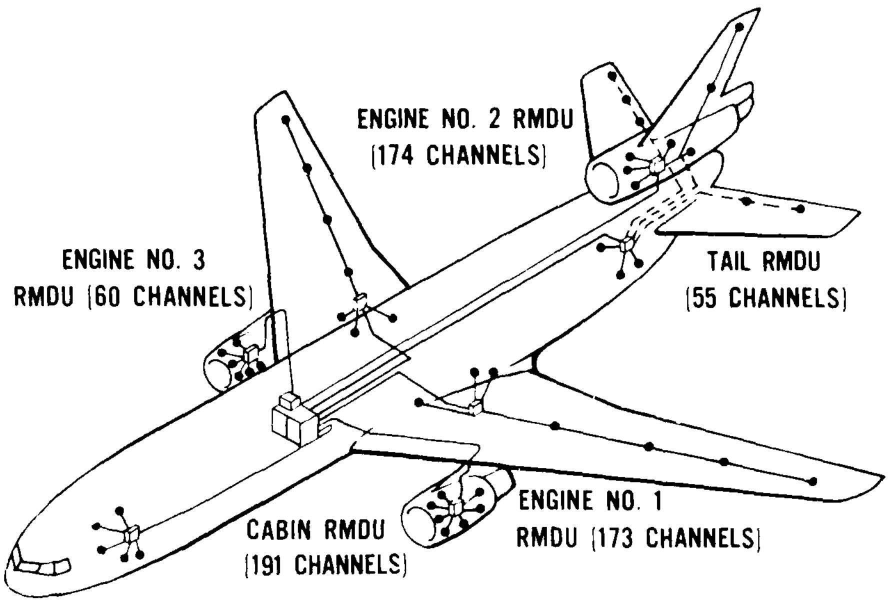


FIGURE 8. DATA ACQUISITION SYSTEM USING REMOTE PCM MULTIPLEXING

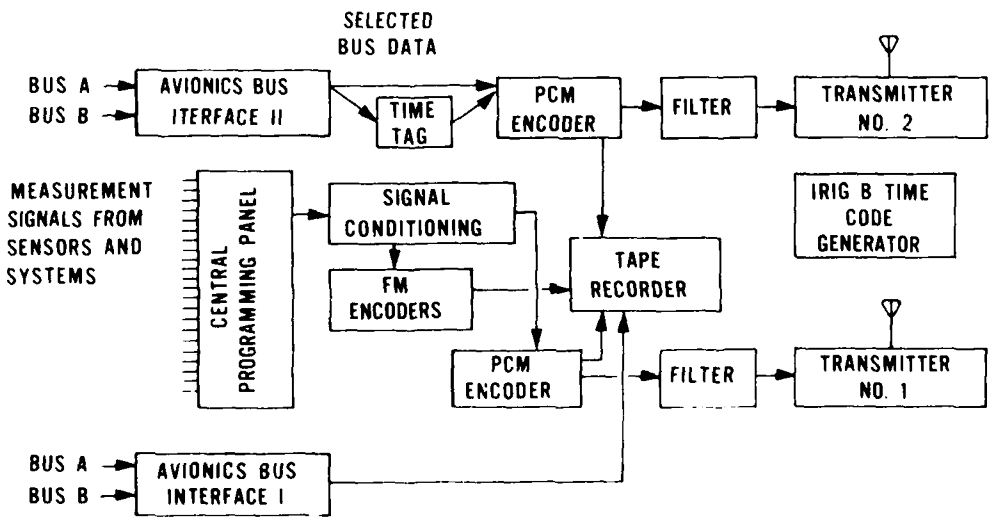


FIGURE 9 SIMPLIFIED INSTRUMENTATION BLOCK DIAGRAM SHOWING AVIONICS INTERFACE BUSES

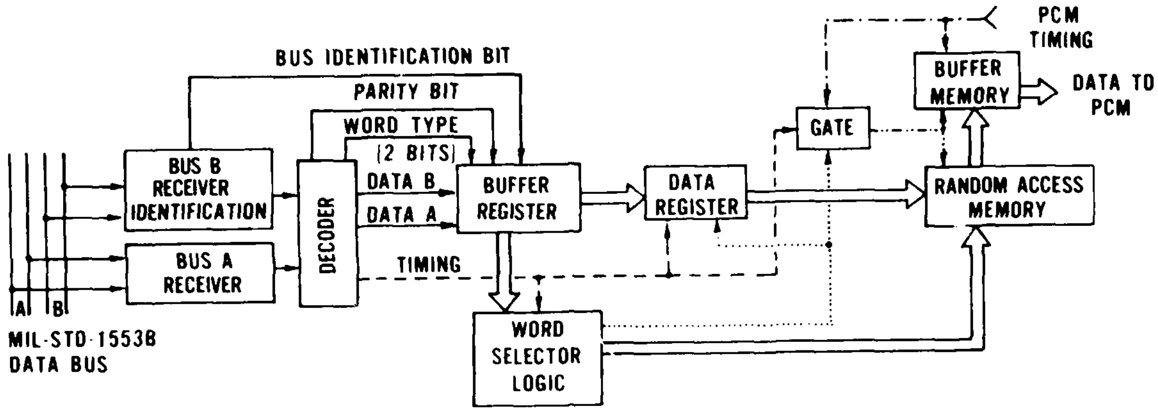


FIGURE 10. AVIONICS BUS INTERFACE II BLOCK DIAGRAM

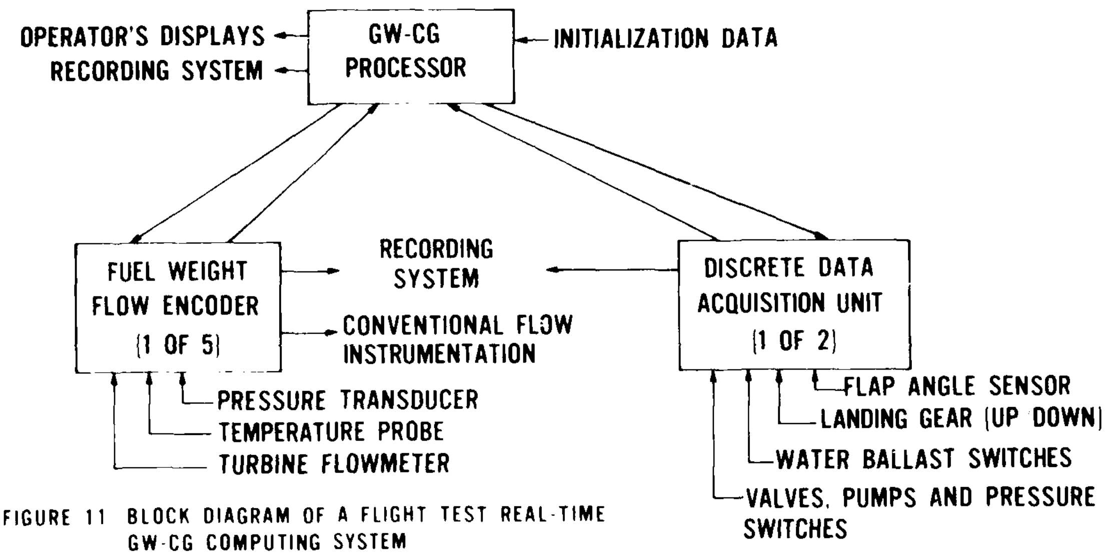


FIGURE 11 BLOCK DIAGRAM OF A FLIGHT TEST REAL-TIME GW-CG COMPUTING SYSTEM

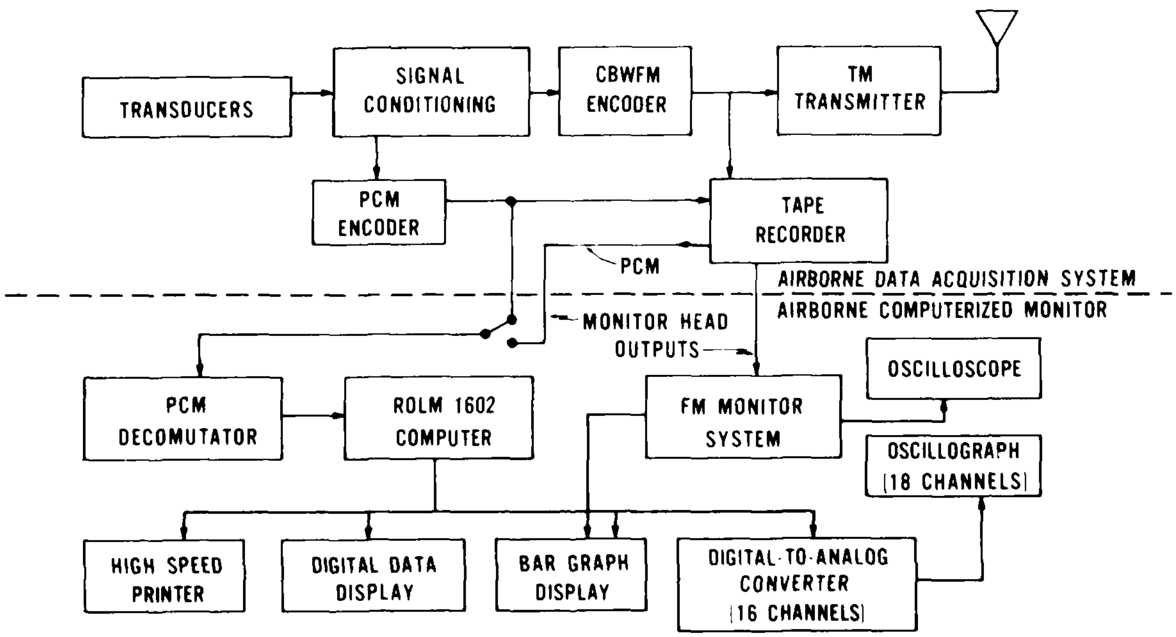


FIGURE 12 AIRBORNE DATA ACQUISITION SYSTEM WITH A COMPUTERIZED MONITOR

AN INVESTIGATION OF THE LINEAR AND ANGULAR VIBRATION ENVIRONMENTS OF TRIALS AIRCRAFT

G. L. Wray and D. J. Flynn
Royal Aircraft Establishment
Farnborough, Hampshire, UK

SUMMARY

This paper describes an investigation of the dynamic environment of trials aircraft based on the measurement of linear acceleration and angular velocity along and about the three principal airframe axes. The results, expressed in the form of power and cross spectral densities, are used to set up an empirical mathematical model of the dynamic environment of an aircraft for use in computer simulations of a strapdown inertial navigator. The application of the model to the simulation of strapdown system alignment in a nominally stationary vehicle is discussed.

LIST OF SYMBOLS

A_p	*maximum amplitude of a peak of a power spectrum*
A_{ij}	maximum amplitude of the cross spectrum between channels i and j
a, a_1, b, g_1	scaling factors or constants
b_1, c, d	
$a(t), a_1(t)$, $b(t), b_1(t)$, $c(t)$	components of motion
B	aircraft body axes or the half power bandwidth of a spectral peak
C_N^B	transformation matrix between navigation axes and body axes
E	earth frame or a constant
$E(x)$	the expected value of a variable x
$\mathcal{F}_\omega\{x(t)\}$	Fourier transform of a signal $x(t)$ evaluated at a frequency ω
$\underline{g}$	apparent acceleration due to earth's gravity
$\underline{g}_m$	the mass attraction gravity vector
$H(s)$	transfer function of a system
h	moment arm
$[\underline{I}]$	matrix of input disturbances
j	square root of -1
k	constant
$\mathcal{L}_s\{x(t)\}$	the Laplace transform of a function $x(t)$
N	navigation or geographic axis system (north, east and down)
$n_i(t_n)$	a Gaussian white disturbance at a time t_n
R	radius of the earth
$S_\theta(\omega)$	power spectrum of an output $\theta(t)$ evaluated at a frequency ω
$S_I(\omega)$	power spectrum of an input $I(t)$ evaluated at a frequency ω
S_{AB}	*cross spectrum between channels A and B*
S_{ij}	elements of a noise scaling matrix
t	time interval
T	time interval or a matrix of transfer functions
t_n	time t_n of a time series t_1, t_2, ...
$[\underline{v}]^N$	a vector v expressed in navigation frame coordinates
$\underline{v}_E$	velocity with respect to the earth
$\|x\|$	the modulus of a quantity x where x may be real or complex
$[\underline{X}]$	a vector representing output motion of a system
X, Y, Z	body axes
α	the real part of a complex number or scaling factor
β	scaling factor
Δt	a time increment
ϵ	error
θ	output displacement or an angle
$\dot{\theta}$	output velocity

LIST OF SYMBOLS (concluded)

$\dot{\nu}$	rate of change of output velocity
$\gamma_{xy}(\omega)$	coherence between channels X and Y at a frequency ω
θ, ϕ	relative phase angle
ϕ_{ij}	elements of a transition matrix
ω_{IE}	the angular velocity of earth axes with respect to inertial space
Ω	the magnitude of the angular velocity of the earth
ω_0	the centre frequency of a peak in a power spectrum
ω	angular frequency
ω_R	constant
λ	scaling factor or latitude
μ	scaling factor
*	complex conjugate
$(dv/dt)_I$	rate of change of a vector with respect to inertial space

1 INTRODUCTION

Early in the development of strapdown inertial navigation systems in the United Kingdom the importance of the dynamic environment of the host vehicle was recognised. The ability to describe the environment in a quantitative way yields valuable data for the design of inertial sensors and the computational algorithms for both the alignment and navigation modes of system operation.

Since vibration data was in short supply it was decided to construct a strapdown measurement package, described in section 2 of this paper and to examine the linear accelerations and angular velocities along and about the principal airframe axes. The aircraft examined so far are a Comet Mk 4 and a Westland Sea King helicopter which are to act as early test beds for strapdown navigation systems. In the near future it is intended also to investigate the environment of a trials British Aerospace (BAe) Buccaneer aircraft.

Several aspects of the flight envelopes have been monitored from the most benign, where the aircraft is parked and nominally stationary, to the most severe, where the aircraft performs manoeuvring flight. The results, analysed in the form of power spectral densities (PSDs) and cross spectral densities (CSDs), are presented in section 3 and discussed in section 4. Mathematical definitions of power spectral densities, cross spectral densities and associated parameters are given in Appendix A.

In order to simulate the dynamic environment of a strapdown inertial navigator in a realistic fashion use can be made of these linear acceleration and angular velocity measurements. However care must be exercised in interpreting the measurements particularly when it is required to separately compute the gravitation and earth's rate signals associated with strapdown alignment. The application of the vibration measurements to strapdown alignment simulation is considered in section 5.

An empirical mathematical model is presented in section 6 which is capable of simulating the linear and angular vibratory motions of the airframe for use in the development of the computational algorithms of a strapdown navigator. The model is set up using the spectral density data obtained from aircraft measurements. A particular advantage of this model is that it can be used to simulate real motion of any required duration without the need to handle large volumes of data. Also it can be employed to generate hypothetical data for any type of vehicle subjected to any type of environmental condition.

One application of the model reported here is the simulation of the environment of a parked and nominally stationary aircraft. The software developed for this application is described in Appendix C.

Finally, in section 7 a programme of future work is outlined. This includes the use of base motion disturbance, simulated by the empirical model, in the development of strapdown alignment algorithms aimed at minimising reaction time in the presence of sensor noise.

2 THE MEASUREMENT PROCESS

2.1 Data requirements

The vibration data requirements for both strapdown sensor design and computational algorithm development correspond to signals which represent strapdown sensor inputs namely specific force and angular velocity with respect to inertial space expressed in airframe or body axis coordinates.

When data is used for algorithm development in system simulations, it is advantageous to extract the component of the vibration sensor measurements which correspond only to disturbance noise. To this noise component can then be added the separately computed gravitation and earth's rate signals ($[g]^B$ and $[\omega_{IE}]^B$) required for the levelling and gyrocompassing of a strapdown navigator. If this approach were not adopted, these signals would have to be obtained directly from the outputs of the vibration measuring sensors and used for simulation. In this case a precise knowledge would be required of the time history of the attitude of the host vehicle over the period for which the vibration measurements were made and as a consequence the performance of the simulation would be limited by the quality of the sensors chosen for vibration measurement.

Adopting the suggested approach of separately computing the levelling and gyro-compassing signal components of the simulated sensor inputs, the noise measurements in the time domain can be transformed into the frequency domain (power spectra) for convenience of data handling and reconstituted in the time domain during simulation. The time domain data so formed can be constrained to have the same spectral characteristics but in general not the same time history as the original data. Clearly this technique could not be employed if the levelling and gyrocompassing signals were not computed separately.

The simplest procedure is to assume that the vibration measurements consist of noise only and can therefore be added to separately computed signals. The errors in signal estimation associated with this approach have been calculated and are summarised in section 5 where the significance of removing biasses from the vibration data is also discussed.

2.2 Accuracy considerations and choice of sensors

The choice of vibration monitoring sensors was dictated by the requirement to quantify the dynamic environment with an accuracy of a few per cent over the frequency bandwidth of interest to the inertial system designer, (0-100 Hz say). A bandwidth of 300 Hz was chosen as an ideal design aim for the sensor package since no significant phase shift is then introduced up to 100 Hz. It will be shown, however, that angular motion sensors could not be found which would satisfy this particular requirement.

It is also required that the vibration sensors have a linear response over the dynamic environment to which they are to be subjected (approximately 100°/s and 6 g in our case).

The choice of linear accelerometer to meet these requirements was straightforward. Sundstrand QA1200 force-feedback accelerometers were chosen since they satisfied the requirements at moderate cost. The relevant parameters of this sensor are compared with the requirements in Table 1.

The choice of the angular rotation sensor however was not so straightforward. The following options were considered:

(i) a pair of linear accelerometers for each axis separated by a moment arm;

(ii) angular accelerometers;

(iii) captured rate integrating gyroscopes with high torque capability; and

(iv) rate gyroscopes.

The first option involves detecting angular motion by sampling the outputs differentially, linear motion being discriminated by common mode rejection. The approach was however rejected because of the need to closely match each pair of transfer functions over the bandwidth of interest in order to achieve adequate common mode rejection.

The relevant parameters for the rotation sensors are summarised in Table 2 where two examples of angular accelerometers (Systron Donner 4591 and Schaevitz ASM), two examples of rate integrating gyroscopes (Honeywell GG1111 and Northrop G1G6) and one rate gyroscope (BAe Dart RG408) are compared with the requirements.

In order to achieve an appreciable bandwidth using angular accelerometers it is necessary to choose devices with a wide dynamic range. This choice, however, leads to a relatively coarse threshold sensitivity. Additionally the uncertainty of the location of the sensitive axis is considerably greater than the required figure of 10 minutes of arc. For these reasons angular accelerometers were rejected.

The BAe Dart RG408 rate gyroscopes were chosen in favour of the rate integrating gyroscopes largely on cost grounds. In addition they were readily available, had been extensively tested at RAE and were felt to be adequate for the present application except for their limited bandwidth. Since none of the angular motion sensors considered were satisfactory in this respect, the design aim for bandwidth had to be relaxed.

The major implication of this limitation is the lack of accurate phase angle information obtained from cross spectral density data at frequencies greater than a few tens of hertz.

The chosen sensors, three Sundstrand QA1200 linear accelerometers and two RG408 rate gyroscopes were mounted orthogonally in a stainless steel block designed to have no structural resonances in the bandwidth of interest. Tolerances of the block are such that orthogonality errors in the mounting surfaces are less than 1 minute of arc which is negligible compared with the axis errors of the devices themselves.

An electronics module was constructed containing the necessary ±15 V dc power supply for the accelerometers and gyroscopes and a 1200 Hz square wave supply for the rate gyroscope rotors. Six amplifiers were included to buffer the sensor outputs and to provide adjustable gain.

It was found necessary to ac couple the linear accelerometer output signals so that sensed components of the earth's gravitational field did not cause saturation of the processing electronics during high sensitivity measurements. A time constant of 10 seconds was chosen for this purpose.

The buffered analogue output signals are fed into an FM interface and the resulting signals recorded at 9.525 cm/s on magnetic tape. The interface gives ±40 per cent modulation for an input of ±2.5 volts.

The sensor block also carries a temperature sensor the output of which is recorded. When required, the temperature signal can be overwritten with a voice commentary using the aircraft intercom system. In this way the magnetic tapes can be annotated to assist the processing. A block diagram of the equipment is given in Fig 1.

The positions within the aircraft for the installation of the vibration monitoring block were chosen to correspond to those for the installation of strapdown navigation equipment. This was done to remove any ambiguity in lever arm effects and the effects of differing structural modes.

Installation in the Comet was on the starboard side of the cabin floor 3.4 metres aft of the nominal centre of gravity. In the Sea King the equipment was installed on the centre line of the rear cabin floor.

Harmonisation of the block (nominal block axes) with respect to aircraft body axes was achieved with uncertainties of 0.1 degree about the x and y axes and 0.5 degree about the z axis.

The contributions to the misalignment between true sensor axes and body axes are summarised in Table 3. Taking RSS values we have:

x or y axis accelerometer	≈	0.53 degree
z axis accelerometer	≈	0.2 degree
x or y axis gyroscopes	≈	0.58 degree
z axis gyroscope	≈	0.32 degree.

It is recognised that the size, weight and mechanical interface of the vibration sensor block are not representative of the inertial reference frame of a strapdown navigator and as such the way in which structural modes of airframe oscillation are modified by installation of the equipment will differ. The only attempt made to alleviate this potential problem has been to select a rigid portion of aircraft structure for installation. In practice mounting has been directly onto seat rails.

2.3 Processing of vibration data

As previously mentioned, for ease of data handling the vibration data is Fourier analysed to produce power spectral densities of the individual channels and cross spectral densities between the channels. This processing has been carried out by the Department of Flight at Cranfield Institute of Technology.

A sample of the processed data from the Comet and Sea King helicopter measurements is presented in section 3 and discussed in section 4.

3 PRESENTATION OF RESULTS

The vibration data obtained in this work corresponds to measurements made on the RAE Comet 4 flying laboratory and the RAE Sea King helicopter. Various aspects of the flight envelopes have been examined as summarised in Tables 4 and 5 for the Comet and Sea King respectively.

The analysis of the linear acceleration and angular velocity data includes the production of:

- (i) power spectra;
- (ii) cumulative power function;
- (iii) envelope ratios;
- (iv) coherence functions;
- (v) phase relationships between channels; and
- (vi) moduli of frequency response functions.

These functions are described briefly in Appendix A. A more detailed discussion can be found in Ref 1.

In all cases the full set of six power spectra have been generated. However, correlated motion between channels has not yet been examined exhaustively.

Examples of each type of analytical treatment of the vibration data are presented in Figs 2 and 3. This represents only a small fraction of the total analysis performed. The results of a comprehensive analysis of the measurements detailed in Tables 4 and 5 will shortly be published in an RAE Technical Report.

4 DISCUSSION OF RESULTS

Fig 2 gives an example of the analysis performed on the Comet data obtained with the aircraft parked. Figs 2.1 and 2.2 correspond to the PSDs of the y gyroscope and x accelerometer signals and Fig 2.3 the coherence analysis between these channels. Figs 2.4 and 2.5 are the envelope ratio and cumulative power respectively of the y gyroscope signal.

Fig 3 illustrates the analysis carried out on measurements made on the Sea King helicopter. These particular measurements were made with the vehicle parked with engines running and rotors engaged. Figs 3.1 and 3.2 are the PSDs of the y gyroscope and x accelerometer measurements respectively and Fig 3.3 the coherence analysis between these channels.

In order to ensure that the spectral data is interpreted in a realistic way it is necessary to establish what contribution is made by noise introduced by the monitoring electronics and the recording process. This can be achieved by mounting the vibration monitoring block on a stable plinth, of the type used for testing inertial grade gyroscopes, and carrying out spectral analysis of the recorded gyroscope and accelerometer outputs. Since there is no real motion of the block the resulting noise spectra can then simply be subtracted from the spectra obtained from measurements of airframe motion.

This procedure has not yet been carried out and therefore it has not yet been established if electronic and recording noise contributes significantly to the results presented here, particularly in Fig 2 which corresponds to a benign environment. The brief discussion which follows is therefore subject to this possible limitation.

In Figs 2.1 and 2.2 consider the peak which occurs at 1 Hz. This may be due to undercarriage resonance[2]. From Fig 2.3 it can be seen that the motion in the two channels is both highly correlated and in phase. The low value of the envelope ratio at this frequency suggests that the motion is non-random (small variance) and the change in the cumulative power between 0.8 and 1.2 Hz indicates that the power present in this peak (approximately 0.005 $(\mathrm{deg/s})^2$) is extremely small.

A 1 Hz peak is also found in the Sea King data (Figs 3.1 and 3.2) which is both highly correlated and in phase. The peaks at 3.4 and 17 Hz correspond to the primary rotor shaft and primary rotor (five blades) frequencies respectively.

The vibration results from the Comet and Sea King aircraft will be presented in full and discussed in more detail in an RAE Technical Report to be published in the near future.

5 APPLICATION OF THE VIBRATION SENSOR DATA TO ALIGNMENT SIMULATION

As stated in the introduction, it is required to develop a computer simulation of aircraft linear and angular vibration for use in the development of strapdown algorithms. Using the measuring device described in section 2, typical aircraft vibrations have been recorded and the data analysed to yield power spectra and cross spectra between channels, as presented in section 3.

Real data could be used for simulation work provided care was exercised in the interpretation of the vibration sensor outputs but the problems involved with handling the required volume of data would be considerable. In addition the results of such a simulation would be limited by the accuracy of the sensors used and a knowledge of the time history of the aircraft's attitude would be required.

The use of a computer model to simulate motion with the correct spectral characteristics has, however, been adopted since it is considered to be a more flexible approach for the following reasons:

(i) the data handling problems are avoided;
(ii) angular and linear vibration data can be simulated which corresponds to any vehicle or environment, real or hypothetical.

In order to remove the necessity to know the time history of the attitude and the dependence of the simulation performance on the quality of the vibration sensors, it is necessary to first separately compute the gravitation $[\underline{g}]^B$ and earth's rate $[\underline{\omega}_{IE}]^B$ signals associated with the alignment. To these are added components representing disturbance noise which are formed from the motion simulation programme. The spectral characteristics associated with these noise components are in turn extracted from the vibration data presented in section 3. The sum of these signal and noise terms then constitute the required accelerometer and gyroscope inputs to the alignment simulation. This technique is now considered in more detail for the gyroscope and accelerometer inputs in turn.

5.1 Simulated gyroscope inputs

The input signal seen by a triad of strapdown rate measuring gyroscopes is the angular velocity of the body frame with respect to inertial space in body frame components $[\underline{\omega}_{IB}]^B$. Now

$$[\underline{\omega}_{IB}]^B = [\underline{\omega}_{NB}]^B + C_N^B[\underline{\omega}_{EN}]^N + C_N^B[\underline{\omega}_{IE}]^N , \qquad (5\text{-}1)$$

where $[\underline{\omega}_{NB}]$ is the angular velocity of the body frame with respect to the local geographic frame. This represents airframe vibration;

$[\underline{\omega}_{EN}]$ is the angular velocity of the geographic frame with respect to the earth frame. This represents bulk airframe motion with respect to the ground;

$[\underline{\omega}_{IE}]$ is the angular velocity of the earth frame with respect to inertial space;

and C_N^B is the transformation matrix between the geographic and body frames.

In constructing the inputs to the gyroscope models in the alignment simulation using Eq.(5-1), it is required to compute separately the gyrocompassing signal (final term on the right hand side of Eq.(5-1) and to add this to the first two terms of Eq.(5-1) which are to be generated by the vibration simulation.

If the vibration simulation is set up with unmodified spectral data from actual rate gyroscope measurements of the vehicle environment, it will generate signals which have the same spectral characteristics as the measurements themselves, *ie* corresponding to Eq.(5-1) plus any rate gyroscope bias. Since the vibration simulation technique generates unbiased outputs, dc components of the measurement (gyroscope bias and the dc component of the final term) will be lost. Thus an unmodified simulator output (S_u) would correspond to:

$$S_u = [\underline{\omega}_{NB}]^B + C_N^B[\underline{\omega}_{EN}]^N + C_N^B[\underline{\omega}_{IE}]^N_{ac} \tag{5-2}$$

Since the last term on the right hand side of Eq.(5-2) is to be generated separately it is ideally required to deduce the power spectrum corresponding to this term and to remove it from the power spectrum of the angular measurement $[\underline{\omega}_{IB}]^B$.

(*Note* that simple subtraction is not appropriate (see Appendix A) since the elements are correlated through C_N^B and $\underline{\omega}_{NB}$.)

The resulting power spectrum corresponding to the first two terms in Eq.(5-2) would be used to prime the vibration simulator which would then generate a modified output (S_m) where:

$$S_m = [\underline{\omega}_{NB}]^B + C_N^B[\underline{\omega}_{EN}]^N \quad . \tag{5-3}$$

The gyrocompassing signal $C_N^B[\underline{\omega}_{IE}]^N$ in both its dc and ac components could then be separately computed in the simulation, and added to the modified vibration simulator output S_m. Thus the required inputs to the sensor models (Eq.(5-1)) are generated without error.

This procedure is, however, rather complex and it is required to establish whether a simple approximation will yield sufficiently accurate results.

Suppose the vibration simulator is primed with the measured angular velocity power spectrum, then it will generate an output corresponding to Eq.(5-2). To this output would be added the separately computed gyrocompassing signal $C_N^B[\underline{\omega}_{IE}]^N$ in both its ac and dc components and the resulting signal (S_A) used as an approximation to Eq.(5-1). Now

$$S_A = [\underline{\omega}_{NB}]^B + C_N^B[\underline{\omega}_{EN}]^N + C_N^B[\underline{\omega}_{IE}]^N_{(ac)} + C_N^B[\underline{\omega}_{IE}]^N_{(ac)} + C_N^B[\underline{\omega}_{IE}]^N_{(dc)} \tag{5-4}$$

which is in error by $C_N^B[\underline{\omega}_{IE}]^N_{(ac)}$ compared with the required signal given by Eq.(5-1). This approach is therefore valid only if the error in approximating Eq.(5-1) *viz*

$$\epsilon = \frac{|C_N^B[\underline{\omega}_{IE}]^N_{(ac)}|_{max}}{|[\underline{\omega}_{IB}]^B|_{max}} \tag{5-5}$$

is acceptably small.

In order to estimate the magnitude of this error assume that the airframe is subjected to both pure linear disturbances along the y axis and angular disturbances about the x axis which give rise to linear components due to moment arm effects.

Assume that the pure linear motion can be described by

$$y = y_0 \sin(\omega_L t)$$

and the angular motion by

$$\theta = \theta_0 \sin(\omega_a t)$$

which gives rise to a linear component

$$y' = h\theta_0 \sin(\omega_a t)$$

where h is the moment arm.

In order to simplify the treatment ignore the fact that these motions are uncorrelated, assume they are in phase and simply add the effects. From data taken on the Comet, we may assume that $\omega_L = \omega_a = 2\pi$, h = 200 cm and apply the constraints:

$$y_0 + h\theta_0 = 0.5 \text{ cm} ,$$

(maximum lateral movement at the mounting point of the sensor pack) and arbitrarily,

$$y_0 = h\theta_0 .$$

Evaluating $[\underline{\omega}_{IB}]$ in Eq.(5-5) gives:

$$[\underline{\omega}_{NB}] \triangleq \theta_0\omega_a = 7.8 \times 10^{-3} \text{ rad/s} ,$$

$$[\underline{\omega}_{EN}] \triangleq \frac{y_0\omega_L + h\theta_0\omega_a}{R} = 5 \times 10^{-9} \text{ rad/s} ,$$

where R is the radius of the earth and

$$[\underline{\omega}_{IE}] \triangleq \Omega \cos \lambda = 4.6 \times 10^{-5} \text{ rad/s} ,$$

which is the horizontal component of earth's rotation rate at a latitude λ of 51 degrees. Hence

$$[\underline{\omega}_{IB}] \triangleq 7.85 \times 10^{-3} \text{ rad/s} .$$

In order to evaluate the maximum value of the numerator in Eq.(5-5) assume that the vehicle is pointing west and is subjected to the motion described. Then

$$C_N^B[\underline{\omega}_{IE}]^N_{(ac)} = \Omega \cos \lambda \cos \theta - \Omega \sin \lambda \sin \theta - \Omega \cos \lambda .$$

Since the maximum value of θ is small (2.5×10^{-3} rad) this approximates to

$$C_N^B[\underline{\omega}_{IE}]^N_{(ac)} = -\Omega \sin \lambda\theta_0 = 7.06 \times 10^{-8} \text{ rad/s} .$$

Substituting these values for $[\underline{\omega}_{IB}]$ and $C_N^B[\underline{\omega}_{IE}]^N$ into Eq.(5-5) yields

$$\epsilon = 9 \times 10^{-4} \text{ per cent}$$

which is acceptable for the disturbance motion assumed.

5.2 Simulated accelerometer inputs

The input signal seen by the strapdown accelerometer triad is the specific force in body axis components $[\underline{A}]^B$. Now

$$[\underline{A}]^B = \left[\left(\frac{d^2\underline{R}}{dt^2}\right)_I\right]^B - [\underline{g}_m]^B \tag{5-6}$$

where the first term on the right hand side represents inertial acceleration and the second represents the mass attraction gravitation vector.

Expressing $\underline{g}_m$ in terms of $\underline{g}$ the apparent acceleration due to the earth's gravitational field, Eq.(5-3) can be written

$$[\underline{A}]^B = \left[\left(\frac{d^2\underline{R}}{dt^2}\right)_I\right]^B + [\underline{g}]^B - [\underline{\omega}_{IE} \wedge (\underline{\omega}_{IE} \wedge \underline{R})]^B . \tag{5-7}$$

Using Coriolis' equation it can be shown that inertial acceleration is related to groundspeed $\underline{v}_E$ by

$$\left[\left(\frac{d^2\underline{R}}{dt^2}\right)_I\right]^B = \left[\left(\frac{d\underline{v}_E}{dt}\right)_B\right]^B + [(\underline{\omega}_{IB} + \underline{\omega}_{IE}) \wedge \underline{v}_E]^B + [\underline{\omega}_{IE} \wedge (\underline{\omega}_{IE} \wedge \underline{R})]^B . \tag{5-8}$$

Therefore Eq.(5-7) becomes

$$[\underline{A}]^B = \left[\left(\frac{d\underline{v}_E}{dt}\right)_B\right]^B + \left[[\underline{\omega}_{IB}]^B + [\underline{\omega}_{IE}]^B\right] \wedge [\underline{v}_E]^B + C_N^B[\underline{g}]^N \quad . \tag{5-9}$$

In constructing the inputs to the accelerometer models in the alignment simulation (Eq.(5-9)) it is required to compute separately the levelling signal (final term) for the reasons discussed earlier.

In order to synthesise the accelerometer input signals in a realistic way the following procedure might be adopted.

With the same disturbance motion as was assumed for the treatment of the gyroscope signals, the cross product term in Eq.(5-9) is small and can be neglected. Separating the pure linear and moment arm contributions the signal for the y accelerometer can be written as:

$$A_y = \left.\frac{dv_y}{dt}\right|_{linear} + h\ddot{\theta}_x - g\theta_x \quad . \tag{5-10}$$

Since the second and third terms are correlated and uncorrelated with the first term, the following procedure is suggested.

The angular velocity ($\dot{\theta}_x$) power spectrum from the x rate gyroscope is converted into a spectrum of angle (θ_x) by dividing the amplitude of each peak by the square of its centre frequency. This power spectrum is then scaled by multiplying by $-g^2$.

Again operate on the x rate gyroscope power spectrum but now generate a power spectrum of angular acceleration ($\ddot{\theta}_x$) by multiplying the amplitude of each peak by the square of its centre frequency. This power spectrum is then scaled by the square of the assumed moment arm h.

The two power spectra so formed correspond to the spectra of $-g\theta_x$ and $h\ddot{\theta}_x$ in Eq.(5-10). These two correlated spectra are then combined (see Appendix A) and the result subtracted from the power spectrum measured by the y accelerometer to produce the spectrum corresponding to the uncorrelated motion (first term in Eq.(5-10)). This is then used to prime the vibration simulator which will yield as its output the acceleration due to the pure linear motion in the time domain.

The basic structure of the strapdown alignment simulation is illustrated in Fig 4. The real world model generates the time history of vehicle attitude (equivalent to θ_x in Eq.(5-10)). It can also be made to generate the time history of angular acceleration (equivalent to $\ddot{\theta}_x$ in Eq.(5-10)). These two parameters can be scaled by $-g$ and h respectively and added to the vibration simulator output to yield the input signals to the accelerometer models, correctly formed with the levelling signal correctly computed.

Once again the procedure is rather complex and it is required to establish whether a simple approximation will be sufficiently accurate.

The simplest procedure is to prime the vibration simulator with the measured linear accelerometer power spectra so that it will generate an output S_u equivalent to the ac component of Eq.(5-10), *ie*

$$S_u = \left.\frac{dv_y}{dt}\right|_{linear} + \left.h\ddot{\theta}_x\right|_{\substack{moment\\arm}} - \left.g\theta_x\right|_{ac} \tag{5-11}$$

for the simple motion being considered. (Accelerometer bias will not be present in the output since ac coupling of the accelerometer signals is adopted and bias is removed in the vibration simulator.)

To this output would be added the separately computed levelling signal $-g\theta_x$ in both its ac and dc components. The resulting signal given by:

$$S_A = \left.\frac{dv_y}{dt}\right|_{linear} + \left.h\ddot{\theta}_x\right|_{\substack{moment\\arm}} - \left.g\theta_x\right|_{ac} + \left.[-g\theta_x]\right|_{ac\&dc} \tag{5-12}$$

would be used as an approximation to Eq.(5-10). This approach will only be valid if the error, given by

$$\epsilon = \frac{\left.g\theta_x\right|_{ac}}{A_y} \tag{5-13}$$

is acceptably small.

Substituting parameters for the assumed motion into Eq.(5-13) for A_y and $\left.g\theta_x\right|_{ac}$ gives:

$$\frac{dv_y}{dt}_{linear} = y_0\omega_L^2 = 9.9 \text{ cm/s}^2 = 10 \text{ mg} ;$$

$$h\ddot{\theta}_x = h\theta_0\omega_a^2 = 9.9 \text{ cm/s}^2 = 10 \text{ mg} ;$$

$$g\theta_x = 1.2 \text{ cm/s}^2 = 1.3 \text{ mg} ;$$

and hence $A_y = 21.3$ mg . Also

$$g\theta_{x_{ac}} = 1.2 \text{ cm/s}^2 = 1.3 \text{ mg} .$$

For the assumed motion then an error of

$$\epsilon = 6 \text{ per cent}$$

results in the input of the accelerometer models which for the present purposes is considered to be adequate.

It should be emphasised, however, that if the motion was predominantly angular and the moment arm small, as might be the case for high frequency structural modes of oscillation, a large error would result from this approximate approach and the more complex procedure should therefore be adopted.

6 THE MODELLING AND SIMULATION PROCESS

In section 6.1 a mathematical approach is developed for modelling linear and angular vibration and in section 6.2 the application of the model to the simulation of real data is described.

6.1 The mathematics of vibration simulation

Two approaches have been considered for the simulation of vehicle frame vibration. They are referred to here as the 'Analytic' approach and the 'Empirical' approach.

The 'Analytic' approach involves modelling the dynamics of a particular vehicle by deriving a set of transfer functions which relate output motions to input disturbance. These transfer functions would describe the uncoupled and coupled motions between the six channels (three linear and three angular). The output motions $[\underline{X}]$ could then be related to the input disturbances $[\underline{I}]$ by the matrix equation:

$$[\underline{X}] = T[\underline{I}] \qquad (6-1)$$

where $[\underline{I}]$ is a 6×1 vector representing the disturbances, $[\underline{X}]$ is a 6×1 vector representing the output motions and T is a 6×6 matrix of transfer functions.

There are two major problems associated with this approach. Firstly it would be extremely difficult to calculate the 36 transfer functions to any degree of accuracy except for the most simple of systems. In particular it would be virtually impossible to account for all the structural modes of oscillation. Additionally it would be potentially difficult to quantify physical input disturbances in terms of the six input channels. In the case of an aircraft for example it would not be trivial to model as an input the effect of a wind gust.

Because of these difficulties, and the relative inflexibility of the approach, it was decided to employ the so-called 'Empirical' approach. In this case the statistics of the motion to be simulated are expressed as a set of power spectra and cross spectra. These spectra are divided into constituent peaks which are quantified by their magnitude, centre frequency and half power bandwidth. For each of these peaks a transfer function is derived which operates on white noise input disturbances. These disturbances are generated with zero mean to ensure that the resulting output motion is unbiassed. The motions corresponding to each of the peaks in the power and cross spectra are appropriately scaled, phase shifted and added into the appropriate channels to yield net motion with the desired spectral characteristics.

The mathematical analysis of this 'Empirical' approach is treated in four stages as follows.

(i) Initially either linear or angular motion of the aircraft along or about one of the three orthogonal body axes is considered. In addition such motion is assumed to have a particularly simple power spectral density as illustrated in Fig 5. The required output from a simulation will result when white noise is passed through a filter with the appropriate transfer function. The required transfer function is derived in section 6.1.1.

(ii) Algorithms are then derived (in section 6.1.2) whereby the above transfer function is used together with a white noise input to produce linear (or angular) motion as a function of time.

(iii) The motion corresponding to more complex power spectra (as in Fig 6) is then simulated by combining the outputs of individual simple systems of the type considered in section 6.1.1. This is discussed in section 6.1.3.

(iv) Finally, the above treatment can be extended to provide simulated motion for several channels simultaneously. In particular section 6.1.4 shows how correlated motion between channels may be simulated.

6.1.1 Theory for a simple system

As previously stated, the vibrational motion to be simulated is assumed initially to have the PSD illustrated in Fig 5. A transfer function H(s) is selected whose modulus squared approximates to this shape. When such a transfer function acts upon appropriately scaled white noise, the output will represent vibrations with the given PSD.

It is shown in Appendix B that the PSD, $S_I(\omega)$ of an input to a transfer function H(s) is related to the PSD of the output $S_\theta(\omega)$ by the expression

$$S_\theta(\omega) = |H(j\omega)|^2 S_I(\omega) \quad . \tag{6-2}$$

Thus, if the input is white noise scaled to have a uniform PSD of height 1 unit^2/Hz then

$$S_I(\omega) \equiv 1 \quad .$$

Eq.(6-2) therefore becomes

$$S_\theta(\omega) = |H(j\omega)|^2 \tag{6-3}$$

This relates the magnitude of $H(j\omega)$ to the PSD of the required output.

The transfer function for an under-damped second order system may be expressed as:

$$H(s) = \frac{\omega_n^2 \sqrt{k}}{s^2 + 2E\omega_n s + \omega_n^2} \quad . \tag{6-4}$$

It is found that $|H(j\omega)|^2$ corresponds to $S_\theta(\omega)$ as required provided the constants k, ω_n and E are related to the parameters ω_0 (centre frequency), B (half power bandwidth) and A_p (maximum power density) as specified in Fig 5 by:

$$\omega_n = \omega_0\left[2 - \left(1 - \frac{B^2}{2\omega_0^2}\right)\right]^{\frac{1}{4}} \qquad \text{for } \frac{B^4}{4\omega_0^4} \text{ small} \tag{6-5a}$$

$$E = \left(\frac{1 - \left(\frac{\omega_0}{\omega_n}\right)^2}{2}\right)^{\frac{1}{2}} \tag{6-5b}$$

$$k = \left(1 - (1 - 2E^2)^2\right)A_p \quad . \tag{6-5c}$$

Having specified the appropriate transfer function it is now necessary to derive the precise equations required to generate the simulated vibrations.

6.1.2 Implementation of the theory

The task of providing the explicit algorithms to generate motion is tackled in two stages. Firstly the equation relating input to output for a continuous system is expressed in terms of the transfer function, and then converted into a differential equation. Secondly, since outputs are required at discrete time intervals, the continuous equation is converted to a discrete time form from which the required outputs can be extracted.

If the input to a continuous second order system I(t) has Fourier transform $\mathcal{F}_\omega\{I(t)\}$ and the output $\theta(t)$ has Fourier transform $\mathcal{F}_\omega\{\theta(t)\}$ then as shown in Appendix B, these are related by

$$\frac{\mathcal{F}_\omega\{\theta(t)\}}{\mathcal{F}_\omega\{I(t)\}} = H(j\omega) = \frac{\omega_n^2 \sqrt{k}}{(j\omega)^2 + 2E\omega_n(j\omega) + \omega_n^2} \quad . \tag{6-6}$$

Cross multiplying, we have

$$\left[(j\omega)^2 + 2E\omega_n(j\omega) + \omega_n^2\right]\mathcal{F}_\omega\{\theta(t)\} = \omega_n^2\sqrt{k}\,\mathcal{F}_\omega\{I(t)\} \quad . \tag{6-7}$$

This may then be converted to a differential equation by taking the inverse Fourier transform of each side. Thus:

$$\ddot{\theta}(t) + 2E\omega_n\dot{\theta}(t) + \omega_n^2\theta(t) = \omega_n^2\sqrt{k}\,I(t) + \text{const.} \tag{6-8}$$

For purposes of computer simulation it is convenient to express this second order differential equation in a single variable as two first order equations in matrix form. This is accomplished by introducing a second variable

$$\psi(t) = \dot{\theta}(t) \quad .$$

θ is subsequently referred to as output displacement and ψ as output velocity.

Eq.(6-8) may then be written as:

$$\dot{\psi}(t) + 2E\omega_n\psi(t) + \omega_n^2\theta(t) = \omega_n^2\sqrt{k}\,I(t) + \text{const.} \tag{6-10}$$

Combining Eq.(6-9) and (6-10) into a single matrix equation we have

$$\begin{pmatrix}\dot{\theta}\\ \dot{\psi}\end{pmatrix} = \begin{pmatrix}0 & 1\\ -\omega_n^2 & -2E\omega_n\end{pmatrix}\begin{pmatrix}\theta\\ \psi\end{pmatrix} + \begin{pmatrix}0\\ \omega_n^2\sqrt{k}\end{pmatrix} I(t) + \text{const.} \tag{6-11}$$

Eq.(6-11) which is applicable to a continuous system is now approximated by a discrete time equation since outputs are required at discrete time intervals. The form of the corresponding discrete time equation is

$$\begin{pmatrix}\theta(t_{n+1})\\ \psi(t_{n+1})\end{pmatrix} = \begin{pmatrix}\phi_{11} & \phi_{12}\\ \phi_{21} & \phi_{22}\end{pmatrix}\begin{pmatrix}\theta(t_n)\\ \psi(t_n)\end{pmatrix} + \begin{pmatrix}s_{11} & s_{12}\\ s_{21} & s_{22}\end{pmatrix}\begin{pmatrix}n_1(t_n)\\ n_2(t_n)\end{pmatrix} \tag{6-12}$$

where $\theta(t_n)$ and $\psi(t_n)$ represent the value of θ and ψ respectively at time t_n, $n_1(t_n)$ and $n_2(t_n)$ represent input disturbances which occur at t_n, and ϕ_{ij} and s_{ij} are constant coefficients to be determined.

From Eq.(6-12) it is seen that the output at time t_{n+1} is a linear combination of the output at time t_n and the disturbances occurring at t_n.

The noise inputs $n_1(t_n)$ and $n_2(t_n)$ are specified as being white Gaussian random numbers with zero mean and unity variance. Here the term 'white' implies that the input disturbances at any time are independent of those of any other time. By imposing the condition that they are 'Gaussian' implies that they take a range of values each of whose likelihood is given by a 'normal' or 'Gaussian' distribution. The condition of unity variance merely acts as an arbitrary scaling factor, the disturbances being rescaled to the correct values by the choice of $s_{11}, \ldots, s_{22}$.

The coefficients ϕ_{11}, ϕ_{12}, ϕ_{21} and ϕ_{22} are determined by solving the continuous equation (6-11) and then expressing the result in the discrete time matrix form of Eq.(6-12). Defining

$$\omega_R = \omega_n(1 - E^2)^{\frac{1}{2}}$$

then it can be shown that:

$$\left.\begin{aligned}
\phi_{11} &= \omega_n E\phi_{12} + \exp[-E\omega_n T]\cos(\omega_R T)\\
\phi_{12} &= \frac{1}{\omega_R}\exp[-E\omega_n T]\sin(\omega_R T)\\
\phi_{21} &= -\omega_n^2\phi_{12}\\
\phi_{22} &= \exp[-E\omega_n T]\cos(\omega_R T) - E\omega_n\phi_{12}
\end{aligned}\right\} \tag{6-13}$$

where $T = t_{n+1} - t_n$.

The coefficients $s_{11}, \ldots, s_{22}$ are next determined by imposing the requirements that the covariances of the outputs of the continuous and discrete cases are equal. This yields the following:

$$\left.\begin{aligned} s_{11}^2 &= \frac{3k\omega_n}{4E}\left(1 - \phi_{11}^2 - \omega_n^2\phi_{12}^2\right) - s_{12}^2 \\ s_{12} &= -\frac{3k\omega_n}{4E}\left(\phi_{11}\phi_{21} + \omega_n^2\phi_{12}\phi_{22}\right) \\ s_{21} &= 0 \\ s_{22}^2 &= \frac{3k\omega_n}{4E}\left(\omega_n^2\left(1 - \phi_{22}^2\right) - \phi_{21}^2\right) \end{aligned}\right\} . \qquad (6\text{-}14)$$

In order to initialise the process, it is necessary to specify the initial values for displacement and velocity. Typical values for these may be generated by random selection from numbers with a Gaussian density distribution. For example the displacement at any time is assumed to have a Gaussian distribution with zero mean and variance E(x²). The explicit values of this variance is equal to the area under the PSD curve and may be shown to be equal to $3k\omega_n/4E$.

In particular this is achieved by generating a Gaussian random number with mean zero and variance 1, and then multiplying the result by a scaling factor of $\sqrt{3k\omega_n/4E}$. In a similar fashion the initial velocity can be generated by multiplying another such Guassian random number (zero mean, variance 1) by $\sqrt{3k\omega_n^3/4E}$.

Having now defined all the constants in Eq.(6-12) and specified the initial conditions, successive values of θ and ψ may be generated for t_1, t_2, ...

It should be noted, however, that Eq.(6-12) is an approximation to Eq.(6-11) and makes the assumption that the motion is linear in each time increment t_n to t_{n+1}. Therefore Eq.(6-12) may only be used to generate simultaneous outputs at frequencies for which the period is large compared with the time increment.

For example, if it is required to simulate motion up to a frequency of 20 Hz an iteration rate of say 200 Hz should be chosen. Higher frequencies can be accommodated by either increasing the iteration rate or adopting a higher order of approximate solution to Eq.(6-11) than that of Eq.(6-12).

6.1.3 The simulation of more complex power spectra

The results and methods detailed in section 6.1.1 and 6.1.2 are now extended to simulate disturbances whose power spectra have more complex forms as indicated in Fig 6b. This is achieved by simply considering the total output as being composed of the sum of several independent components each corresponding to a distinct peak in the power spectrum.

For example the spectrum of Fig 6b is considered to consist of two components as illustrated in Fig 6a. Outputs for each of these two simple constituent systems may then be generated independently as detailed in section 6.1.2 to yield the required result.

Care must be taken, however, to ensure that the noise inputs to the individual peaks are generated by independent random numbers otherwise the resulting motions would be correlated and additional components would appear in the power spectrum (see Appendix A).

6.1.4 Simulating multiple channel systems

It is now required to extend the above techniques to cope with the simulation of several channels of disturbance simultaneously.

If each of the channels vibrates independently of the others, then the task is simple. In this case each channel may be treated separately and the motion simulated according to the techniques detailed in sections 6.1.2 and 6.1.3. However, in order to ensure that the output motion of each channel is independent of the rest, it must be ensured that the random input disturbances which drive each channel are also independent. This entails generating separate and independent random numbers to drive each channel individually.

The problem, however, is a little more complex if the channels are coupled. The method of incorporating correlated motion in the simulation is introduced here by a simple two-channel example.

Suppose it is required to simulate the motion of a system with two channels, A and B which are coupled to some degree. Each channel consists of two components, one representing the independent or uncorrelated motion, and the other a component common between channels.

Thus the output of channel A is:

$$\left.\begin{aligned} a(t) &= a_1(t) + \lambda c(t) \\ \text{and the output of channel B is:} \\ b(t) &= b_1(t) + \mu c(t - \tau) \end{aligned}\right\} . \qquad (6\text{-}15)$$

Here $a_1(t)$ and $b_1(t)$ represent the independent components of motion and $c(t)$ the common component. This common component is scaled by λ and μ before adding to the channels A and B respectively and a time lag τ or relative phase angle introduced. If $a_1(t)$, $b_1(t)$ and $c(t)$ are generated independently it is found (see Appendix A) that the averaged power spectra of the output of each channel is related to the power spectra of the constituent parts by the relationships:

$$\left.\begin{aligned} S_A(\omega) &= S_{A_1}(\omega) + \lambda^2 S_C(\omega) \\ \text{and} \\ S_B(\omega) &= S_{B_1}(\omega) + \mu^2 S_C(\omega) \end{aligned}\right\} . \qquad (6\text{-}16)$$

In addition the cross spectrum between the channels is given by

$$S_{AB}(\omega) = \lambda\mu S_C(\omega) \qquad (6\text{-}17)$$

(*Note*: in the above it is assumed that averaged expressions are used to give meaningful values.)

For convenience λ and μ may be chosen so that $\lambda\mu = 1$. Then Eq.(6-17) becomes

$$S_{AB}(\omega) = S_C(\omega) .$$

To summarise, by defining power spectra S_{A_1}, S_{B_1} and S_C, generating outputs from each and combining according to Eq.(6-15) the result will be equivalent to two channels whose PSDs are $S_{A_1} + \lambda^2 S_C$ and $S_{B_1} + (1/\lambda)^2 S_C$, and whose cross spectrum is S_C.

The relative phase angle will be expressed by the time lag τ of Eq.(6-15) and may be chosen independently for each peak of the power spectra. In this way it is possible to simulate the motion of two channels where the power spectra of the uncorrelated motion, and the cross spectra are given. It should also be noted that the choice of λ may be made from a range of possible values and indeed may be chosen independently for each peak frequency.

Therefore, the outputs in this simple case are regarded as being caused by three types of disturbance, one of which causes correlated motion, and others causing independent motion in each channel.

For a system with many channels, however, any particular type of disturbance may exercise certain combinations of channels. Also, associated with the resulting output for that particular disturbance will be relative phase relationships between those channels. It is therefore possible to build up a total output from each channel by adding together appropriate components which are either correlated with other channels or are independent.

Consider for example a four-channel system with channels A, B, C and D. Suppose a given input disturbance exercises channels A, B and D. Then such a component of motion may be generated, scaled for each of the three channels, and added to the uncorrelated components of motion of those channels incorporating any specified phase relationships.

To summarise, a method has been outlined whereby the motion of a multichannel system can be synthesised by adding together components as desired. The reverse process is, however, more difficult, i.e. it is more difficult to derive the parameters of the individual components from data specifying the total outputs.

6.2 Application of the model to the simulation of real data

Using the model it is possible to reproduce motion which has the same spectral characteristics as real data. However, the available power spectra and cross spectra will never provide a complete specification of the actual motion except in the most trivial of cases. It is shown below that for a given set of spectra there is a range of possible input parameters to the simulation which yield different types of motion, but whose power and cross spectra are equivalent.

At best a full set of power and cross spectra between all of the six channels (linear and angular) will be available. However, this may not always be the case, particularly in view of the current lack of published angular data. This further compounds the problem of priming the simulation to yield the most realistic output.

Consider, for example, the simple two-channel example discussed in section 6.1.4. The only spectral data available is

$$S_A , S_B \text{ and } S_C .$$

In this case the correlated motion scaling factor, λ, may take any positive value provided S_{A_1} and S_{B_1} of Eq.(6-16) remain positive. In addition associated with the available spectra S_A, S_B and S_C there are likely to be a number of peaks and it is feasible that λ may be different for each.

It therefore becomes apparent that whilst it is possible to reproduce motion which has given statistical properties (specified by power spectra and cross spectra) there is considerable latitude in choosing the necessary parameters to prime the simulation. It is possible that the accuracy of this fine detail may be unimportant in certain applications, however, if this is not so the mechanics of the particular system must be studied in order to gain insight into the correlation mechanisms.

Before considering the task of reproducing six-channel motion, it is instructive to first confine our attention to the simpler two-channel case.

6.2.1 Reproducing specified two-channel motion

It is first required to simulate motion in two channels, A and B, which have associated power spectra S_A and S_B and cross spectrum S_{C_1} as illustrated in Fig 7. The first task in determining the input parameters for the simulation is to identify the principal frequencies of these spectra. It is seen that the spectra given by Fig 7 exhibit only one peak with centre frequency ω_0 and that the cross spectrum is identically zero. The motion in channels A and B is therefore uncorrelated. In this case the programme is primed with two independent peaks which happen to have the same frequency and half power bandwidth but different magnitudes A_A and A_B.

Next let it be assumed that the cross spectrum is given by S_{C_2}. Since the maximum magnitude of the peaks of the power spectra are A_A and A_B, and the maximum magnitude of the cross spectrum is the geometric mean of these, it can easily be shown by recourse to fundamental definitions that in this case the two channels are entirely correlated. In this case the coherence between the channels is unity. Therefore the motion of this system may be simulated using motion appropriately scaled, from a single source. The relative phase angle between channels is derived from the real and imaginary parts of the cross spectrum thus:

$$\tan\theta = \frac{\text{imaginary part of cross spectrum}}{\text{real part of cross spectrum}}$$

In both of the examples so far, no correlation and total correlation, there has been no ambiguity in deriving the simulation parameters. However this is not the case in the third example.

Suppose that the magnitude of the cross spectrum is S_{C_3} as in Fig 7. Here correlation is partial and three independent sources of motion must be used to form the resulting output motion for the two channels. Labelling these sources as $a_1(t)$, $b_1(t)$ and $c(t)$ then the output motion of channel A is given by

$$a(t) = \alpha a_1(t) + \lambda c(t)$$

and for channel B is

$$b(t) = \beta b_1(t) + \mu c(t + \tau) \quad .$$

Here $a_1(t)$, $b_1(t)$ and $c(t)$ are constrained to have unity maximum height power spectra,

α, β, λ and μ are scaling factors, and

τ determines the relative phase angle between the channels and cross spectra.

It can be shown that the heights of the power and cross spectra are:

$$\alpha^2 + \lambda^2 = A_A$$

$$\beta^2 + \mu^2 = A_B$$

and

$$\lambda\mu = A_C$$

where α, β, λ and μ are real.

Since there are only three constraints for the four unknowns, recourse to the dynamics of the system must be taken to provide the most realistic solution.

Finally, if peaks at several frequencies appear in the spectra whose motion is to be simulated, then a similar procedure to the above must be adopted for each peak.

6.2.2 Extension of the approach to six channels

The case of simulating motion in six channels is analogous to the simpler two-channel case above. Firstly the power spectra for the six channels are decomposed into their constituent peaks with specified centre frequencies, heights, and half-power bandwidths. The following procedure is then adopted for each peak in all six channels.

A peak is selected in the spectrum of channel 1 say, at a centre frequency ω_0, and the spectra of the remaining channels inspected for peaks at the same frequency. If none exist then this peak corresponds to uncorrelated motion and corresponding motion can be simulated in a straightforward manner. If, however, peaks do occur in the other channels at this centre frequency ω_0, then the degree of correlation must be established by inspection of the appropriate cross spectrum. If the correlation is found to be zero then these peaks can be treated independently for each channel. Alternatively if the correlation is found to be total, then the motion corresponding to those peaks can be simulated from a single source with appropriate scaling and relative phase angle. If, however, the correlation is partial then the following procedure may be adopted.

Consider each of the six channels to have an uncorrelated component of motion $a_i f_i(t)$ ($i = 1, \ldots, 6$), and a component $c_i g_i(t + \tau_i)$ correlated with other channels. As before $f_i(t)$ and $g_i(t)$ are constrained to have unity height power spectra. τ_i ($i = 1, \ldots, 6$) denotes the relative phase angle of each channel with respect to channel 1 where τ_1 is assumed zero. In addition let A_i represent the maximum values of the given spectra at the given centre frequency, and A_{ij} the value of the maximum modulus of the cross spectrum between channels i and j. As in the two-channel case, a_i and c_i must be chosen for $i = 1, \ldots, 6$ in order that the resulting motion has the correct power spectra. Hence the following must hold:

$$a_i^2 + c_i^2 = A_i \qquad \text{for } i=1,\ldots,6 \qquad (6\text{-}21)$$

$$c_i c_j = A_{ij} \qquad \text{for } i=1,\ldots,6;\ j=1,\ldots,6 \qquad (6\text{-}22)$$

where the a_i and c_i are real.

The values for a_i and c_i may then be chosen at will subject to these constraints. Once again recourse to the dynamics of the particular system must be taken to provide the most realistic result. As before the relative phase angles between channels are determined via the cross spectra in the usual manner (see Appendix A). If it is found, however, that the phase angles between the channels are not consistent with each other, then this may indicate that a more complicated set of independent input disturbances is operative. For example, one may envisage the case where correlated motion may exist between channels 1 and 2 and also between 3 and 4 but not between 1 and 3. In this case separate independent disturbances would be giving rise to the two sets of correlated motion.

7 FUTURE WORK

On the basis of the work carried out so far on airframe vibration measurements and simulation, several major areas warrant further investigation.

(i) In the near future it is intended to carry out vibration measurements in the BAe Buccaneer aircraft which will be used to extend the range of dynamic environment for strapdown inertial navigation system evaluation.

(ii) An investigation will be carried out to establish whether noise generated by the monitoring electronics and recording process contributes significantly to the spectral results obtained from aircraft measurements.

(iii) The effect of incorporating more realistic transfer functions to synthesise particular power spectra will be investigated.

(iv) In section 5 it was suggested that under certain circumstances care must be exercised in interpreting vibration data, when attempting to generate inputs to a strapdown alignment simulation. Techniques for treating the vibration data will be examined in detail in an attempt to produce realistic simulation conditions.

(v) The vibration simulator will be added to an existing strapdown alignment simulation (Fig 4) which will be used for algorithm development to minimise reaction time in the presence of inertial sensor noise, e.g. laser gyroscope random walk, and airframe disturbances.

(vi) The motion simulation technique, discussed here in the context of a parked aircraft, will be extended to cope with in-flight conditions by combining it with a flight profile generator.

Acknowledgments

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Table 1

	Linear accelerometer parameters	
	Requirement	Sundstrand QA1200
Dynamic range (g)	±6	±20
Frequency bandwidth (Hz)	300	>800
Scale factor uncertainty (ppm)	10^4	120
Axis uncertainty (arc min)	10	7
Threshold sensitivity (g)	10^{-3}	10^{-6}

Table 2

	Rotation sensor parameters					
	Requirement	Systron Donner 4591	Schaevitz ASM	Honeywell GG1111	Northrop G1G6	BAe Dart RG408
Approximate cost for three axes (at time)	minimum	£6000	£1350	£4000	£6000	£3500
Maximum angular velocity	$100^\circ/s$			$>100^\circ/s$	$100^\circ/s$	$100^\circ/s$
Maximum angular acceleration	$600^\circ/s^2$	$>10^{4\circ}/s^2$	$>8 \times 10^{4\circ}/s^2$			
Bandwidth	300 Hz	150 Hz	130 Hz	60 Hz	60 Hz	100 Hz
Threshold sensitivity	$<0.1^\circ/s$ $0.05^\circ/s^2$	$0.1^\circ/s^2$	$0.9^\circ/s^2$	$0.01^\circ/h$	$0.01^\circ/h$	$<0.01^\circ/s$
Axis uncertainty	10 minutes of arc	60 minutes of arc	60 minutes of arc	3 minutes of arc	3 minutes of arc	12 minutes of arc
Scale factor uncertainty	1000 ppm	1000 ppm	1000 ppm	<1000 ppm	1000 ppm	1000 ppm

Table 3

SUMMARY OF SENSOR MISALIGNMENTS

Misalignment (degrees)	Body axis		
	x	y	z
Body/nominal block	0.1	0.1	0.5
Nominal block/true block	0.02	0.02	0.02
True block/true gyro	0.2	0.2	0.2
True block/true accel	0.1	0.1	0.1

Table 4

Comet profiles
Parked, engines off, light breeze
Parked, engines running
Accelerating to take-off
Climb out
Straight and level cruise

Table 5

Sea King profiles
Parked, engines off, personnel movement
Parked, engines on
Parked, engines on, rotors engaged
Low level manoeuvring flight
40 knot cruise
Hover

REFERENCES

1 J.S. Bendat and A.G. Piersol, Random data: analysis and measurement procedures. Wiley Interscience (1971)

2 A.P. Worgan, Measurement of the motion of parked aircraft by a photographic method. June 1967, RAE Technical Report 67132

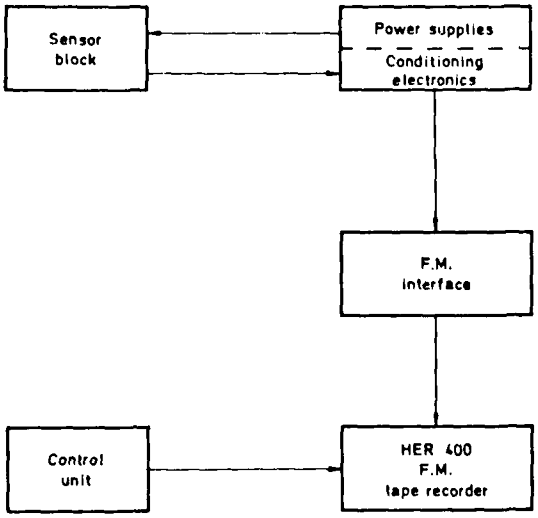
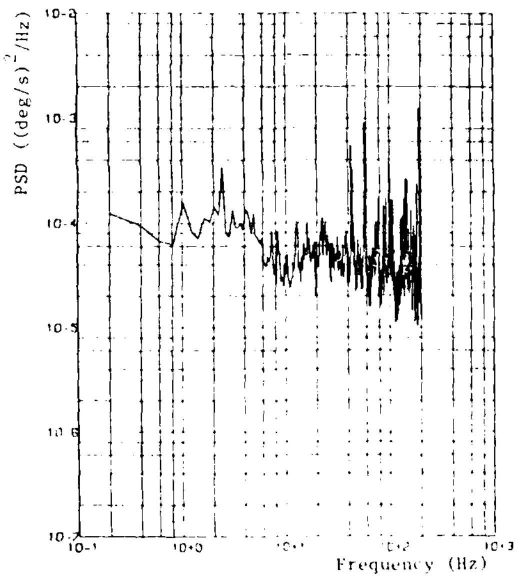
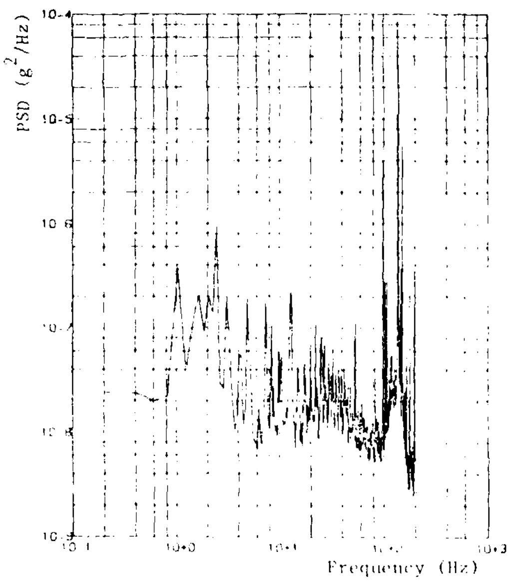


Fig 1 Configuration of measurement apparatus



Bandwidth = 0.2035 Hz
Run time = 235.93 s
rms value = 1.087 E-01 units

Fig 2.1 Comet - parked/engines off - Trk 11 Y-gyro



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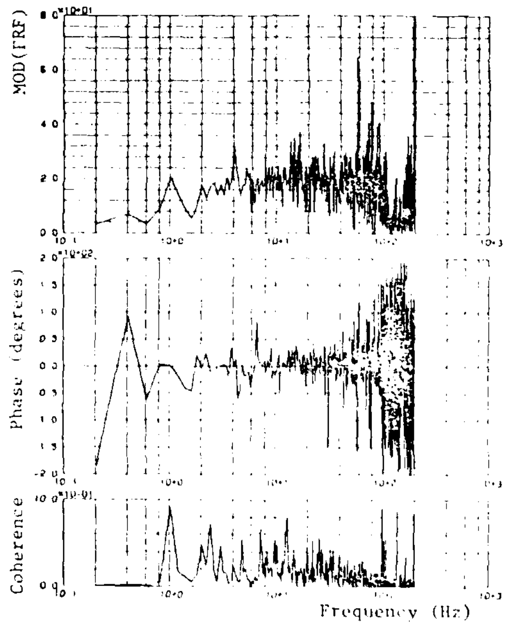
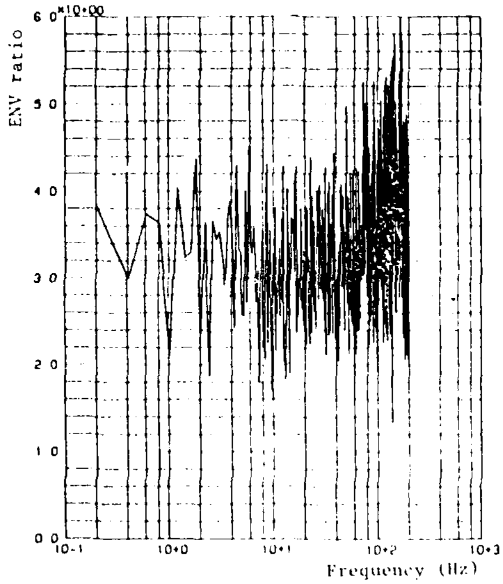
Run time = 235.93 s
rms value = 5.550E-03 units

Fig 2.2 Comet - parked/engines off - Trk 1 X-accn



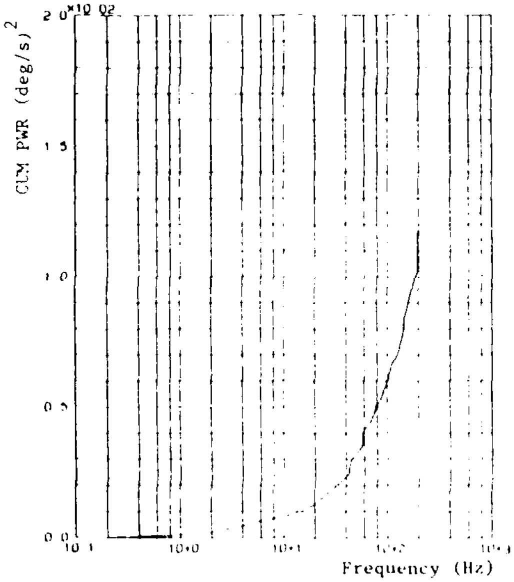
Bandwidth = 0.2035 Hz
Run time = 235.93 s

Fig 2.3 Comet - parked/engines off - Trk 11 Y-gyro Rel Trk 1 X-accn



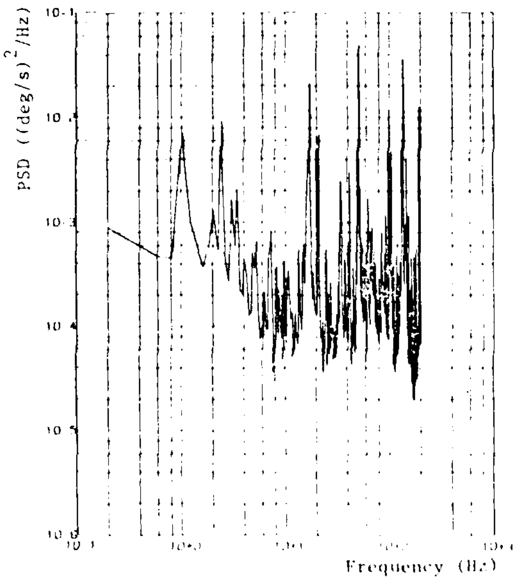
Bandwidth = 0.2035 Hz
Run time = 235.93 s
rms value = 1.087E-01 units

Fig 2.4 Comet - parked/engines off - Trk 11 Y-gyro



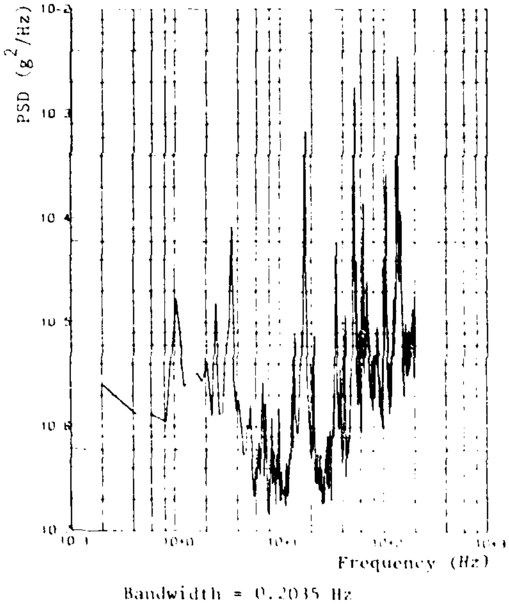
Bandwidth = 0.2035 Hz
Run time = 235.93 s
rms value = 1.087E-01 units

Fig 2.5 Comet - parked/engines off - Trk 11 Y-gyro



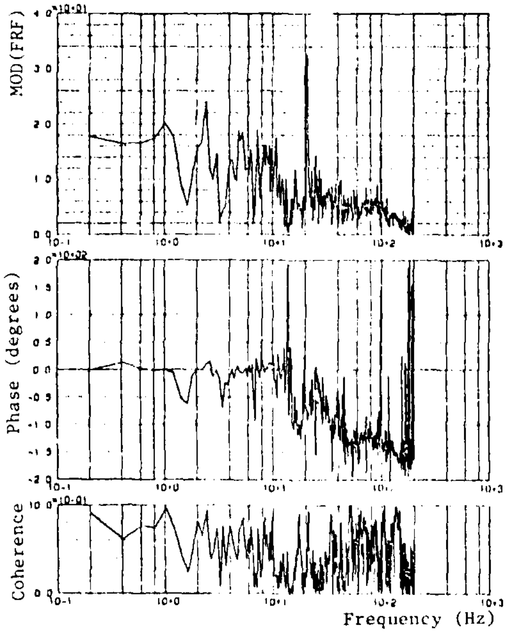
Bandwidth = 0.2035 Hz
Run time = 117.97 s
rms value = 3.991E-01 units

Fig 3.1 Sea King - parked/rotors engaged - Trk 11 Y-gyro



Bandwidth = 0.2035 Hz
Run time = 117.97 s
rms value = 8.744E-02 units

Fig 3.2 Sea King - parked/rotors engaged - Trk 1 X-accn



Bandwidth = 0.2035 Hz
Run time = 117.97 s

Fig 3.3 Sea King - parked/rotors engaged - Trk 11 Y-gyro Rel Trk 1 X-accn

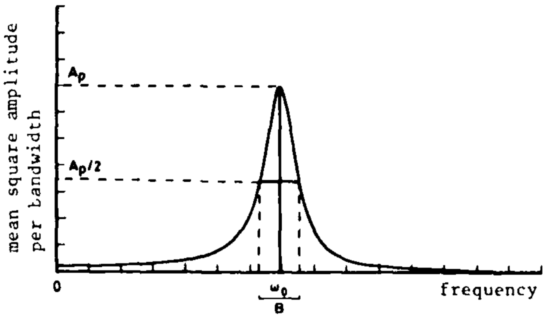


Fig 5 Power spectral density plot of a simple system

ω_0 = centre frequency

A_p = maximum amplitude^2/bandwidth

B = half power bandwidth (range of frequencies over which power is greater than half the maximum)

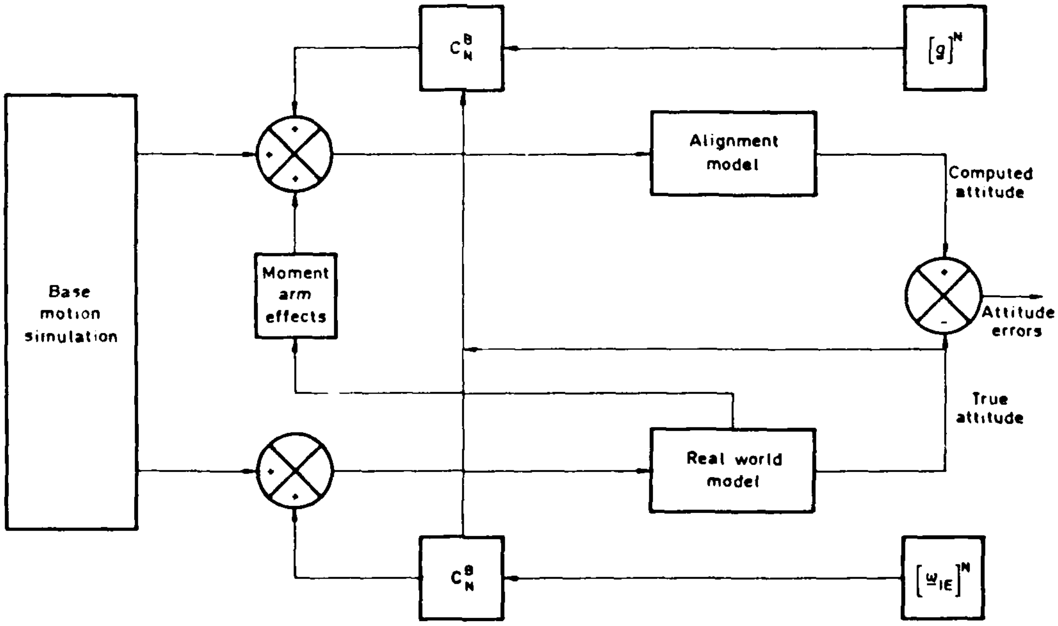


Fig 4 Strapdown alignment simulation

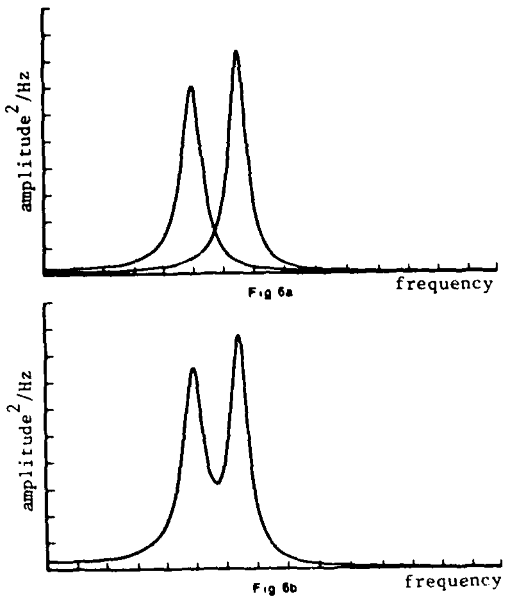


Fig 6 Graphs illustrating how individual power spectra combine to give a total spectrum

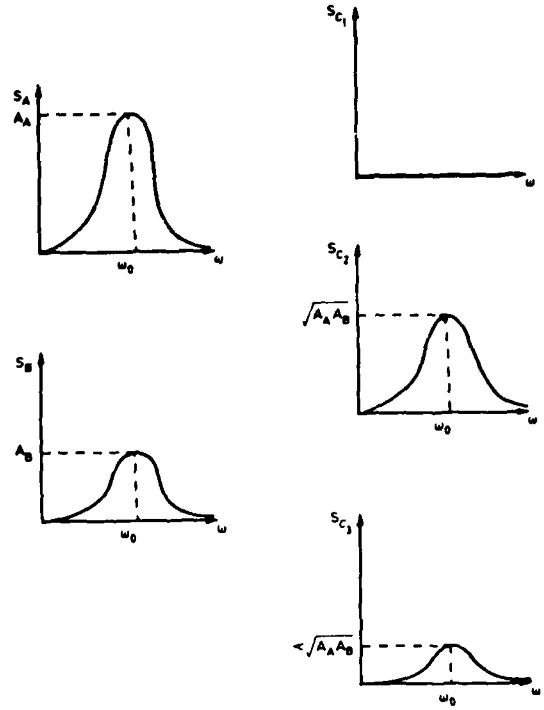


Fig 7 Hypothetical power and cross spectra

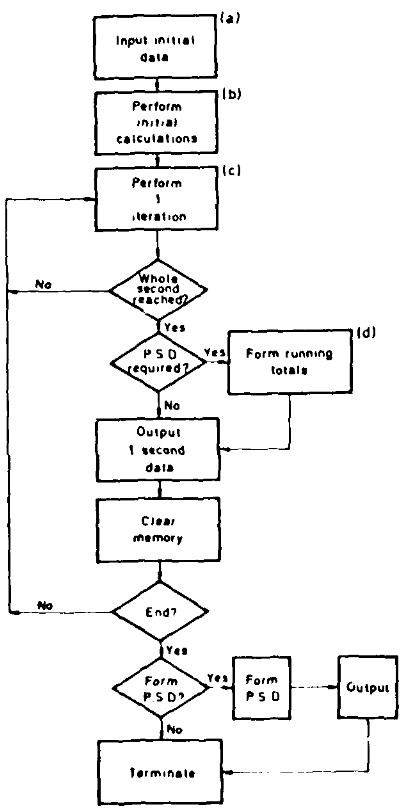


Fig 8 Block diagram of software

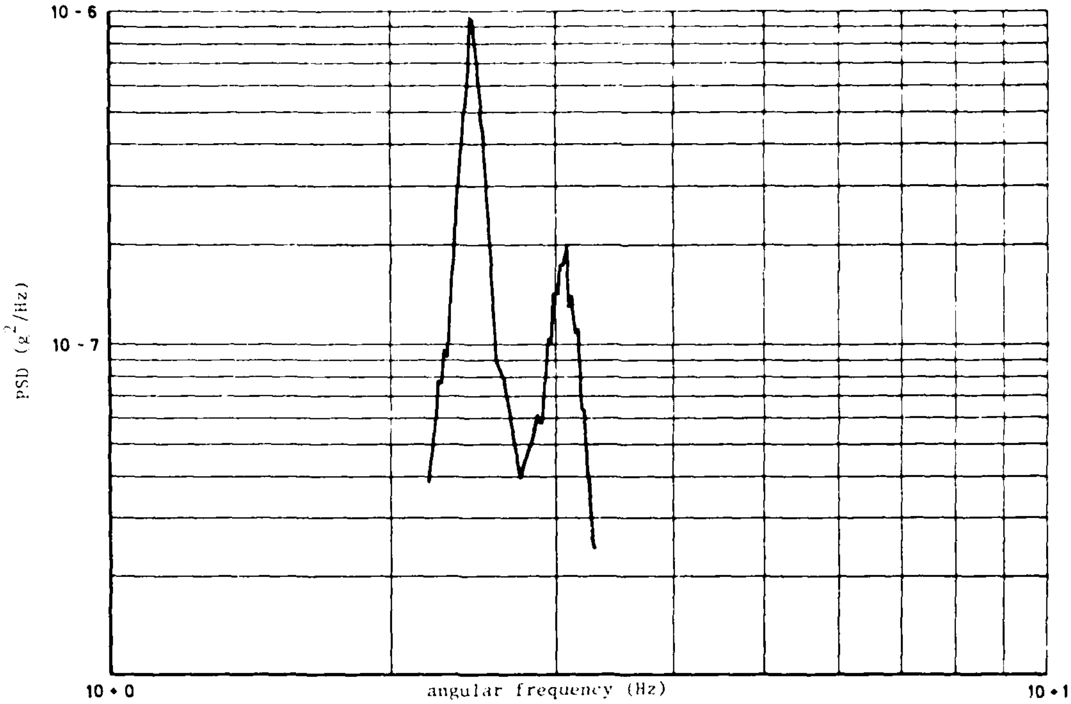


Fig 9 Simulated Comet power spectrum

Appendix A

MATHEMATICAL CONCEPTS USED

A.1 The power spectral density of vibrations

The power spectrum of a mode of oscillation of a vibrating system is defined as a plot of the mean square amplitude, per unit bandwidth, against frequency.

Thus the area under the spectrum between two frequencies ω and $\Delta\omega$ represents the amount of power present in that frequency range.

This may be quantified mathematically as follows.

Suppose the displacement of a system at any time t is given by the continuous function x(t). If the Fourier transform at a frequency of this signal is taken over a time interval 0 to T we have

$$\mathcal{F}_\omega\{x(t)\} = \int_0^T x(t)e^{-j\omega t}dt \quad . \tag{A-1}$$

The result of this is a complex valued function which may be written as $\mathcal{F}_\omega\{x(t)\} = F(\omega)e^{j\theta(\omega)}$. $F(\omega)$ is interpreted as a measure of the 'amount' of signal present at a frequency ω, and $\theta(\omega)$ as the phase angle of that part of the signal. From this the power spectral density function is defined as

$$S_X(\omega) = \frac{1}{T}\left[\mathcal{F}_\omega\{x(t)\}\mathcal{F}^*_\omega\{x(t)\}\right] = \frac{1}{T}\left|\mathcal{F}_\omega\{x(t)\}\right|^2 \tag{A-2}$$

where * denotes the complex conjugate.

This is another way of expressing the amount of signal present at a frequency ω. The value T is introduced to normalise this measure so that an indication of the energy present at that frequency per unit time results.

It should be noted that large fluctuations in the value of the power spectrum can occur from trial to trial when data which is essentially random is analysed. In these cases the spectrum may be averaged over many trials to improve accuracy.

It is shown in Ref 1 that the normalised standard error over q trials is $\sqrt{1/q}$. Thus 100 trials are required for a 10 per cent standard error, and 10000 for a 1 per cent standard error, etc.

The units of the power spectrum are amplitude^2/Hz, and it may be further shown that the area under the power spectrum curve is equal to the variance of the displacement, since the mean value is zero, *ie*

$$\int_0^\infty S(\omega)d\omega = E\left(x(t)^2\right) \tag{A-3}$$

where E denotes expected or mean value.

The above definition, however, applies strictly speaking to functions continuous in the time domain. In this work, however, it is required to form an estimate of the power spectrum of discretely sampled motion. If $x(t_n)$ for $n = 0,1,\dots,m$ represents the series of sampled data, then it can be shown that[1], for positive frequencies, estimates of the power spectrum of x(t) evaluated at ω_k are given by:

$$\frac{2\Delta t^2}{T}\left(\sum_{n=0}^{m} x(t_n)e^{-j\omega_k t_n}\right)\left(\sum_{n=0}^{m} x(t_n)e^{j\omega_k t_n}\right) \tag{A-4}$$

where Δt is the time interval between samples,
T is the duration of the trial where $T = m\Delta t$, and
ω_k is an angular frequency from the set $\{0, 1/T, 2/T, \dots, 1/2, \dots\}$.

Once again accuracy in the estimate of the PSD may be improved by averaging results over many trials.

It should finally be noted that care should be taken in choosing the length of the trial T. This affects the resolution or bandwidth of the filter (expression (A-4)) which is used to evaluate the amount of signal present about a particular frequency. A full treatment of this issue is given in Ref 1. However, for the filter specified by expression (A-4) a bandwidth of about 2/T Hz may serve as a guide to its use. Therefore, when using the above procedure of averaging results over many trials, the duration of each trial T must be chosen so that 2/T, the bandwidth of the filter, is small when compared with the likely half power bandwidth of any peak of interest.

A mathematical concept also used in the presentation of vibration data and closely related to the power spectrum is the cumulative power function. This is defined at a frequency ω_1 as the running integral of a PSD from zero to ω_1, thus:

$$\int_0^{\omega_1} S(\omega)d\omega \quad .$$

It follows that, since a PSD always takes non-negative values, the cumulative power is an increasing function. The use of this variable is illustrated in section 4.

A second function closely related to the power spectrum is the envelope ratio. This is defined at a given frequency to be the ratio of peak to rms values for a set of power spectra evaluated for a set of trials. It indicates the degree of randomness of the signal at each frequency. Again the use of this variable is illustrated in section 4.

A.2 The cross spectral density

This relates the degree to which two vibrations are linked at any given frequency.

If the two sets of vibrations are represented by the continuous functions x(t) and y(t) with Fourier transforms $\mathscr{F}_\omega\{x(t)\}$ and $\mathscr{F}_\omega\{y(t)\}$, then the cross spectral density is defined as

$$S_{xy}(\omega) = \frac{1}{T}\,\mathscr{F}_\omega\{x(t)\}\,\mathscr{F}^*_\omega\{y(t)\} \quad . \tag{A-5}$$

$S_{xy}(\omega)$ is in general a complex valued function and may be represented by

$$S_{xy}(\omega) = \frac{1}{T}\,G^2(\omega)e^{j\phi} \tag{A-6}$$

where $G(\omega)$ is a measure of the component which is common to both vibrations and ϕ the relative phase angle.

Once again the value of the cross spectral density must be averaged to obtain a meaningful result. In so doing the uncorrelated motion is averaged to zero. This is easily seen since uncorrelated motion has a relative phase angle which changes randomly with time. It is these random fluctuations which cause the average for the uncorrelated motion to tend to zero. Whilst the actual phase angle of the correlated motion may also vary randomly, the relative value between channels will be fixed and consequently the result will not average zero.

In an analogous fashion to the definition to the power spectrum considered above, a corresponding set of discrete time equations may be formed for the application of the cross spectral density concept to discrete time series.

A.3 The coherence function

The coherence function $\eta_{xy}(\omega)$ at a frequency ω relates the cross spectrum between two channels X and Y to their power spectra by the expression

$$\eta_{xy}(\omega) = \frac{S_{xy}(\omega)S^*_{xy}(\omega)}{S_x(\omega)S_y(\omega)} \quad . \tag{A-7}$$

Thus it expresses the ratio of the square of the magnitude of the cross spectrum to the product of the individual spectra of the two channels. The coherence function is further seen to take values in the range 0 to 1, zero being taken when the two channels are uncorrelated, and unity when they are totally correlated.

The data presented in this paper on correlated motion is expressed in terms of this variable rather than the magnitude of the cross spectrum itself.

However, given the power spectra and the coherence function the magnitude of the cross spectrum may be obtained from the relationship:

$$|S_{xy}(\omega)| = \sqrt{\eta_{xy}(\omega)S_x(\omega)S_y(\omega)} \quad . \tag{A-8}$$

A.4 The frequency response function

The modulus of the frequency response function (FRF), referred to in the analysis of correlated motion is defined as

$$|FRF| = \frac{|\text{cross spectrum}|}{\text{PSD of reference channel}} \quad . \tag{A-9}$$

For example, if analysis is performed for channel Y relative to channel X, then

$$|FRF| = \frac{|S_{xy}|}{S_x} \quad . \tag{A-10}$$

This parameter, though generated in the data analysis and shown in Figs 2 and 3, is *not* employed in the vibration simulator reported here.

A.5 The combination of power spectra

Two signals a(t) and b(t) with corresponding power spectra S_A and S_B are to be combined and it is required to calculate the resulting power spectrum S_{A+B} of the resulting signal.

By definition:

$$\left.\begin{aligned} S_A &= \frac{1}{T}\mathcal{F}_\omega\{a(t)\}\,\mathcal{F}^*_\omega\{a(t)\} \\ \text{and} \qquad S_B &= \frac{1}{T}\mathcal{F}_\omega\{b(t)\}\,\mathcal{F}^*_\omega\{b(t)\} \end{aligned}\right\} . \tag{A-11}$$

Then

$$\begin{aligned} S_{A+B} &= \frac{1}{T}\mathcal{F}_\omega\{a(t) + b(t)\}\,\mathcal{F}^*_\omega\{a(t) + b(t)\} \\ &= \frac{1}{T}\Big[\mathcal{F}_\omega\{a(t)\}\,\mathcal{F}^*_\omega\{a(t)\} + \mathcal{F}_\omega\{b(t)\}\,\mathcal{F}^*_\omega\{b(t)\} \\ &\qquad + \mathcal{F}_\omega\{a(t)\}\,\mathcal{F}^*_\omega\{b(t)\} + \mathcal{F}_\omega\{b(t)\}\,\mathcal{F}^*_\omega\{a(t)\}\Big] \end{aligned}$$

therefore

$$S_{A+B} = S_A + S_B + S_{AB} + S_{BA} \tag{A-12}$$

If the power spectra S_A and S_B are correlated in general all of the terms in Eq.(A-12) will be non-zero.

For uncorrelated spectra, however, $S_{AB} = S_{BA} = 0$ and Eq.(A-12) reduces to

$$S_{A+B} = S_A + S_B \tag{A-13}$$

Appendix B

DERIVATION OF EQUATION (6-2)

In this Appendix Eq.(6-2) is derived, i.e. the relationship between the power spectrum of the output $S_\theta(\omega)$ and the input $S_I(\omega)$ of a filter with transfer function H(s).

Let the input and output functions be I(t) and $\theta(t)$ respectively.

The Laplace transform $\mathcal{L}_s\{\}$ of the input function is defined as

$$\mathcal{L}_s\{I(t)\} = \int_0^\infty I(t)e^{-st}dt \tag{B-1}$$

where $s = \alpha + j\omega$.

For a sampling interval from 0 to T we can write:

$$\mathcal{L}_s\{I(t)\} = \int_0^T I(t)e^{-(\alpha+j\omega)t}dt \tag{B-2}$$

or

$$= \int_{-\infty}^{\infty} I(t)e^{-(\alpha+j\omega)t}dt \quad . \tag{B-3}$$

By definition the Fourier transform of the input function $\mathcal{F}_\omega\{I(t)\}$ is given by

$$\mathcal{F}_\omega\{I(t)\} = \int_{-\infty}^{\infty} I(t)e^{-j\omega t}dt \tag{B-4}$$

hence

$$\mathcal{L}_s\{I(t)\} = \mathcal{F}_\omega\{I(t)\} \, , \tag{B-5}$$

when α the real part of the complex frequency is zero.

Now the input and output Laplace transforms are related by the expression

$$\mathcal{L}_s\{\theta(t)\} = H(s)\,\mathcal{L}_s\{I(t)\} \quad . \tag{B-6}$$

For a sampling interval from 0 to T and for $\alpha = 0$ we have

$$\mathcal{F}_\omega\{\theta(t)\} = H(j\omega)\,\mathcal{F}_\omega\{I(t)\} \quad . \tag{B-7}$$

Taking complex conjugates gives

$$\mathcal{F}^*_\omega\{\theta(t)\} = H^*(j\omega)\,\mathcal{F}^*_\omega\{I(t)\} \quad . \tag{B-8}$$

Multiplying Eq.(B-7) by (B-8) and dividing by T gives:

$$\frac{1}{T}\,\mathcal{F}_\omega\{\theta(t)\}\,\mathcal{F}^*_\omega\{\theta(t)\} = |H(j\omega)|^2\,\frac{1}{T}\,\mathcal{F}_\omega\{I(t)\}\,\mathcal{F}^*_\omega\{I(t)\}$$

or

$$S_\theta(\omega) = |H(j\omega)|^2 S_I(\omega) \tag{B-9}$$

which is the required result.

Appendix C

SIMULATION SOFTWARE

C.1 Software description

The algorithms described in section 6 have been implemented in FORTRAN IV on a Prime 750 computer. The main program, 'BMS', generates output motion in six channels with specified spectral characteristics. The units of the output motion are mean square linear acceleration/Hz and mean square angular velocity/Hz for the three linear accelerometer and rate gyroscope channels respectively.

The program incorporates an optional test facility, described later in this Appendix, which enables the following parameters to be calculated:

(i) the time averaged output for each channel;

(ii) the standard deviation of the output for each channel;

(iii) the moduli of the power spectra of the channel outputs and cross spectra between the channel outputs. The program is currently restricted to a maximum of 20 frequencies for these parameters;

(iv) the phase angles of the output motions at up to 20 frequencies.

A separate program (not presented here) is used to average these results over many trials of the program to ensure statistical validity.

The basic structure of the 'BMS' program is illustrated in Fig 8. The important aspects of the program operation are described below.

Block (a)

Input data is fed in both directly by the operator and also via an input file. The input data required directly consists of:

(i) input file name;
(ii) run duration in minutes;
(iii) required iteration rate (maximum 200 Hz); and
(iv) whether the program test option is required.

If the test option is required the following additional information is required:

(a) the number of frequencies for which power spectra, cross spectra and phase angle calculations are to be performed; and

(b) the output file name.

The input data file referred to above contains the following information:

(i) the number of peaks to be simulated;
(ii) their centre frequency in Hz;
(iii) their half power bandwidths in Hz; and
(iv) the scale factors and phase angles (in degrees) relative to channel 1 for the motions corresponding to each peak.

Block (b)

Initial calculations are performed for each peak in order to evaluate:

(i) the normalised initial output position and velocity. (The peaks are then appropriately scaled and phased before adding in to each channel);

(ii) the transfer function; and

(iii) the transition and scaling matrices Φ and S (see section 6.1.2).

Block (c)

Random input disturbances are generated and Eq.(6-12) is iteratively updated. As a result successive values of output velocity and acceleration are obtained which are scaled, phased and added into a 12 × 800 element array whose columns represent the velocity and acceleration of each of the six channels at successive points in time. A set of pointers specified by the relative phase angles determines where the velocities and accelerations, appropriate to each peak, are added into each channel. The pointers are incremented each iteration until their values reach 800 when they are reset to zero. In this way the matrix cyclically stores the net motion of the system as it is built up from each peak over 4 seconds.

After every 200 iterations, a block of 200 columns of the matrix representing the system motion over 1 second of simulated time is output. These elements of the array are then set to zero ready to accumulate a new set of values. It can be shown from the above that the program can accommodate a maximum relative phase angle with respect to channel 1 which is equivalent to a 1.5 second time delay.

Block (d)

If the program test option is required, running totals of certain parameters are updated after every second of elapsed iteration time. The final values of these parameters are then used, on termination of the main program, to evaluate the required power spectra, cross spectra, and phase angles according to the algorithms described in Appendix A.

C.2 Verification of the simulation program

In order to verify that the simulated motion has the same spectral characteristics as the data used to set up the model, three simple procedures have been adopted.

(i) The time history of the simulated motion is observed and the centre frequencies of the principal spectral components deduced.

(ii) The simulation program is made to estimate the mean values of the simulated motion which should approximate closely to zero. In addition the variances of the simulated displacements and velocities are determined. These can be shown to satisfy the relationships:

$$E\left(x(t)^2\right) = \frac{3k\omega_n}{4E} \qquad \text{(C-1)}$$

and

$$E\left(\dot{x}(t)^2\right) = \frac{3k\omega_n^3}{4E} \ . \qquad \text{(C-2)}$$

The ratio of these two quantities is used to verify ω_n.

Having estimated ω_0 (in (i) above) and ω_n, Eqs.(6-5) are used to estimate B, C and A_p (using also Eq.(C-1) or Eq.(C-2) above).

Thus relevant parameters can be checked for each spectral peak produced. The procedure, however, becomes unmanagable when numerous peaks are present in the spectra. In this case the third approach is adopted.

(iii) The simulated motion is Fourier analysed and power and cross spectra generated for comparison with the real data. It must be stressed that it is essential to generate these spectra averaged over many trials in order to obtain statistically valid results. An application of this approach is described below.

C.3 Application of the program to the simulation of Comet data

Two simple examples of the application of the simulation to the environment of the parked Comet aircraft are presented as follows.

(1) The motion corresponding to two peaks in the x accelerometer power spectrum (Fig 2.2) have been simulated. The characteristics of the two peaks in question are:

(a) centre frequency 2.45 Hz, 0.1 Hz half-power bandwidth, and maximum height $10^{-6}\ g^2/\mathrm{Hz}$

(b) centre frequency 3.055 Hz, 0.2 Hz half-power bandwidth, and maximum height $2 \times 10^{-7}\ g^2/\mathrm{Hz}$.

The simulation results have been averaged over 100 trials to yield a normalised standard error of 10 per cent in the resulting power spectra. In addition each trial represents 60 seconds of simulated motion. This duration is chosen to ensure adequate resolution of the digital filter used to estimate the spectrum (see Appendix A). The averaged power spectrum of the simulated motion is presented in Fig 9. It is seen that the above parameters are simulated correctly.

(2) In the second example the 2.45 Hz peak mentioned above has been simulated in the x accelerometer and the y gyroscope channels according to the spectra and coherence between channels as detailed in Fig 2. Thus the maximum height of the x accelerometer and y gyroscope spectra are seen to be $10^{-6}\ g^2/\mathrm{Hz}$ and $3.5 \times 10^{-4}\ {}^\circ/\mathrm{s}^2/\mathrm{Hz}$ respectively. Since the coherence is 0.7 then correlation is not total and three independent sources f(t), g(t) and c(t) are used to construct the simulated motion. Each of these components has centre frequency 2.45 Hz, half-power bandwidth 0.1 Hz, and unity maximum height in its associated power spectrum. The x accelerometer motion is then formed as $a_1 f(t) + b_1 c(t)$, and the y goroscope motion as $a_2 g(t) + b_2 c(t)$.

By employing the techniques detailed in section 6.2.2, one possible set of values for the constants a_1, b_1, a_2 and b_2 are chosen as:

$$a_1 = 4.36 \times 10^{-4} \qquad a_2 = 9.0 \times 10^{-4}$$
$$a_3 = 6.87 \times 10^{-3} \qquad a_4 = 1.74 \times 10^{-2} \ .$$

Employing these parameters, the value for the coherence between the channels for the simulated motion is found to be 0.693 which is in good agreement with Fig 2.3.

A METHOD FOR MEASURING TAKE-OFF AND LANDING PERFORMANCE OF AIRCRAFT, USING AN INERTIAL SENSING SYSTEM*

by

A. Pool, J.L. Simons, G.J.H. Wensink and A.J.L. Willekens
National Aerospace Laboratory NLR
Anthony Fokkerweg 2
1059 CM AMSTERDAM
The Netherlands

SUMMARY

The STALINS method for measuring take-off and landing trajectories is briefly described and results of flight tests made in 1978-80 are discussed. The method now meets the requirements to which it was designed, and a few improvements in the hardware and software are being finalized. The method is expected to be ready for operational use in the course of 1981.

1 INTRODUCTION

When supplying instrumentation for take-off and landing measurements to Fokker and other users, the NLR has for the last two decades relied on a method using on-board cameras. In this method (Ref. 1), called the ROBOT method after the make of the cameras, these nose-mounted cameras take pictures of the landing lights along the runway at a rate of one per second. The data processing requires the reading, on each picture, of the positions of 3 pairs of landing lights. A computer programme then calculates the position of the aircraft relative to the runway in 3 co-ordinates, and the 3 attitude angles. These data are combined with the outputs of longitudinal and vertical accelerometers to obtain the ground speed and the forward and vertical accelerations of the aircraft. The method has been very reliable, but it has important disadvantages which weigh heavily in modern flight testing: the data processing (especially the film reading) is very time consuming and requires much manual work, and the accuracy is marginal.

For these reasons the NLR has closely followed all developments in the field of take-off and landing methods and has itself done extensive investigations on one of these: the use of inertial sensing systems. This method is especially suitable for tests which are conducted at several different airfields because the most important part of the instrumentation is on-board the aircraft. The ground instrumentation that is needed can usually be carried in the aircraft to its destination.

The earlier investigations of the NLR were reported in references 2 and 3. They indicated that the method was feasible. In this paper recent work done in 1978-1980 is reported. The results, which were obtained with a computer programme which (when finalized) can provide trajectory data within 24 hours, provide strong evidence that the method meets the requirements to which it was designed. A few improvements of the equipment and the finalization of the computer programme are now in discussion. It is expected that operational application with the required accuracy will be possible in the course of 1981. The method has received the acronym STALINS (STArt and Landing with an INertial System).

The tests discussed in this paper have all been conducted in the Fokker F [illegible] prototype by the Fokker [illegible]. The data [illegible] and interpretation have been done by the NLR, with advice from Fokker. The work has been carried out [illegible] with the Netherlands Agency for Aerospace Programs (NIVR).

2 REQUIREMENTS

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[illegible]

[illegible]

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[illegible]

* [illegible]

- In the LTN-58 system spurious pulses can occur, which can significantly affect the accuracy of the calculated height.
- Although the reference system against which the STALINS results were compared (ROBOT nose camera) did not fully meet the accuracy requirements, these tests indicated that the STALINS results were at least as accurate as the ROBOT results if no spurious pulses were present.

These results were discussed with Litton, who provided modifications to the vertical channel. These resulted in the provision of an additional ground connection and in a new quantizer card, which reduced the measuring range to 0.4 g (which seemed sufficient on the basis of earlier measurements) with a corresponding reduction in the quantizing step.

In 1978 a new series of flight tests was made at Brétigny (France) with the new vertical quantizer circuit. There the STRADA laser-theodolite of CEV was used as a reference system. The results of these tests are discussed below. Two problems were encountered:
- The modified vertical channel of the Litton platform was overloaded during all landings and all RT 's. As a result, the STALINS calculations of the height during the landings did not give usable results (for RT's height is not required).
- The STRADA laser-theodolite showed large errors (up to 5 m) in its horizontal measurements, due to a fault in a circuit which was only detected after the measurements. An analysis of the data showed that the error could be corrected to a certain extent, but the accuracy then was poorer than the 0.5 m specified. The height measurement of STRADA was not impaired.

In order to fill this gap, STALINS recordings were made during operational landing measurements by the Fokker F28 aircraft at Ypenburg Air Base in 1980, where the ROBOT nose-camera method was used as a reference. In the Litton system part of the modifications were eliminated, so that the measuring range was increased and the quantization step increased again to 0.04 m/s.

This paper reports the results of the tests at Brétigny and Ypenburg, and presents the conclusions that can be drawn from them.

4 DESCRIPTION OF THE STALINS CALCULATIONS

4.1 General lay-out of the programme

In view of the time limit of 24 hours on the data processing time, automatic data processing would provide an ideal solution. It was recognized at an early stage, however, that full automation would require extremely complex programmes and that for at least a few of the final corrections human intervention would be much more effective. The computer programme has, therefore, been set up in two stages:
1. A fully automatic stage which completes the calculation of the trajectory without any human intervention, but does not apply some of the corrections that are necessary to obtain the highest possible accuracy. This automatic programme includes correction or interpolation of data in simple cases of data corruption, determination of periods of standstill of the aircraft and of the intervals over which the data must be filtered and integrated.
2. A manual stage, in which a human operator can check the results of the automatic stage at a computer terminal with graphic display and can call up data from several intermediate stages of the automatic calculation. The operator can interactively apply corrections and check the results of these interventions.

In the next few sections the principles of the computer programme are briefly described.

4.2 Basic principles of the calculations

The basic principle of the method is very simple. In the inertial system there is a platform, which is maintained in a horizontal attitude (the ... [illegible] ... towards the North Pole ... [illegible]

[illegible]

4.3 [illegible]

4.3.1 [illegible]

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aircraft, mainly because of the tilt of the platform due to the Schuler motion. The errors can be quite large: values of up to 0.5 m/s have been found during the tests, which would result in errors of up to 50 m for the maximum integration times of 100 s. These are corrected by subtracting from all values of VNS and VEW the velocity recorded during standstill immediately before a take-off or immediately after a landing. The automatic integration is based on these corrected values of the measured velocities. The detection of the periods of standstill and the determination of the values of the speed corrections are done automatically.

For the exact determination of the speed correction three problems are of importance:

- The rate-of-change of the velocity error may reach values of 7×10^{-4} m/s^2, which could cause errors of 3.5 m in an integration over 100 seconds.
- In the Litton system the speeds are presented in discrete steps of about 0.04 m/s, so that for the long integration times of 100 s an error of 4 m in the distance could result.
- If the aircraft does not come to a complete standstill, the programme may not find a value of the velocity at standstill and will then produce no trajectory at all or, if the aircraft velocity stabilizes at a constant low value for a few seconds, it may use that value and produce a completely wrong trajectory.

These points are checked in the second, interactive stage of the computation (see section 4.4).

Using the automatically corrected values of the horizontal velocities, the automatic programme then calculates the average direction of motion of the aircraft during the ground run. The corrected horizontal velocities are then transformed to velocities along that direction and perpendicular to it, and these are integrated to give the distances along the runway and perpendicular to it. During the manual second stage of the computation the calculated average direction of motion is checked against the geographical direction of the runway.

4.3.2 Height

The basic problem with height measurement using an inertial system is that the vertical kinematic velocity of the aircraft must be determined from a very small difference between two large quantities: the measured vertical acceleration and the acceleration of gravity. In order to achieve the required accuracy over a long integration period (100 seconds), this difference must be determined to an accuracy of the order of 3×10^{-5} m/s^2 (3×10^{-6} g). Such an accuracy is difficult to achieve: it is better than can be expected from the specification of the accelerometer and may in many cases be better than the accuracy with which the local acceleration of gravity can be obtained.

In the STALINS method a different approach is therefore used. The value of "zero vertical kinematic velocity" is determined separately during each ground run. On a completely horizontal runway this can be done by averaging the measured vertical acceleration over the longest possible period during which the aircraft is on the ground and the lift is small, i.e. during standstill and the ground run until pitch up. For the Litton system this is done as follows: the vertical velocity output (IVA = integral of vertical acceleration) is integrated over the above-mentioned part of the ground run and then the best quadratic curve is fitted through the data points by a least-squares method. If this parabola is subtracted from the integral of IVA for the ground run, the result will be that the calculated height during the ground becomes zero on the average. The same correction is applied during the airborne part of the trajectory.

If the runway is not completely horizontal, a correction is applied during the automatic processing, using a height profile of the runway that must have been stored in the computer beforehand. In order to be able to apply that correction, the position of the aircraft along the runway must be known. This is not given by the calculation described in section 4.3.1, which gives only the relative distance from the point of standstill. The relation between the runway co-ordinate and the relative distance obtained by integration is established by recording on-board the aircraft the time at which the aircraft passes a fixed point on the runway. This is done by the RASP system developed by the NLR, which radiographically defines a plane surface perdendicular to the runway centre line. The moment at which the aircraft passes this plane, produced by an antenna on the ground, is recorded in the aircraft. A description of an older version of the RASP system is given in reference 4; the present RASP system uses the same principle but has appreciably smaller ground antennas.

Before the height integration process is started, a Coriolis correction must be applied to IVA.

4.4 Calculation - manual interactive stage

4.4.1 Basic principle

It has been found that the automatic data processing described above gives reasonably good results in a majority of cases. It has been found, however, that considerable improvement of the results can be achieved if further checks are done and corrections are applied. These have not been incorporated in the first stage of the calculation because they would increase the complexity of the programme very much and/or because a qualified human operator can reach the, often complex, decisions much more quickly. It does introduce a more or less arbitrary element in the otherwise automatic processing, but it is possible to give specific rules to the operator for those cases which have been encountered before. In case of unknown errors, it is of great advantage to have a human operator in the line. In the computer programme, measures have been taken to make all necessary information readily available to the operator at a computer terminal, and to make it possible for the operator to intervene and introduce new values at all necessary points in the computation.

The final rules for this manual stage have not yet been established completely, and they may have to be adapted as further experience accumulates. Below only the main correction procedures are discussed.

4.4.2 Horizontal distances

The main checks on the horizontal distances are derived from:
- A comparison of the calculated "runway direction" with the actual runway direction,
- "Schuler curves" which are obtained by plotting the velocity values measured during standstill for the whole flight (see figures 1 and 2),
- A comparison between the distance between two RASP antennas as calculated by the programme with the actual distance between these two antennas. For this purpose a second RASP antenna must be used, in addition to the one mentioned in section 4.3.2.

For the measurement of the distance along the runway the comparison between the runway directions is relatively unimportant, as the effect of a small error in the angle is small. It may, however, give an indication that the computer has chosen a wrong fit interval (which is also used in the height calculation). In the rare cases where distances perpendicular to the runway must be determined, it is a very important check. Moreover, it provides a check on possible drift in the North direction of the platform.

The main check on the distance along the runway is the comparison of the RASP distance. The tests have shown that the calculated RASP positions are accurate to about ± 0.5 m if the aircraft is on or near the ground when it passes the antenna. In order to obtain the highest possible accuracy from this check, the antennas should be as far apart as possible.

If a difference between calculated and actual RASP distance occurs, it will usually be due to the error sources mentioned in section 4.3.1. The first error type listed there will cause a velocity error that changes at about a constant rate (acceleration error), the other two will cause a constant velocity error. With a single pair of RASPs it cannot be determined which type of error occurs. The acceleration error can be determined from the "Schuler curves" mentioned above. The "Schuler curves" for the Brétigny tests are shown in figures 1 and 2. It will be seen that the curve does not show the smooth shape obtained during bench tests and has a rather large amplitude. This is partly due to the quantization error (± 0.04 m/s) of the individual velocity values, but also to the effect of aircraft manoeuvres during the tests. As the acceleration is relatively small (order of 7×10^{-4} m/s maximum), the correction is only significant during long measuring runs and even there only in those parts of the Schuler curve where high rates of change occur.

If the acceleration error has been corrected or is insignificant, the remaining difference in the comparison must be due to a constant-velocity error or to the inaccuracy (± 0.5 m) of the RASP measurements themselves.

4.4.2 Height measurement

The main checks on the height measurement are derived from:
- A comparison between successive values of the calculated "acceleration of gravity", i.e. the coefficients of the square term of the fitted parabola.
- A comparison between the calculated height and the height according to a radio altimeter which is used as part of the standard STALINS equipment.

The analysis has shown that in the majority of cases the automatic processing of the data provides a height error which is well within the requirements of section 2. It also shows that the calculated acceleration of gravity is very sensitive to imperfections in the calculated heights. A graph of the successive values of the acceleration of gravity will, therefore, indicate errors, e.g. caused by spurious pulses during the ground run (where they most affect accuracy) or by incorrect choice of the fit interval by the computer.

When it was decided to use a radio altimeter for an overall check on the height measurement, it was thought that this would provide a very firm basis of comparison, and that a mathematical procedure could be developed to automatically determine the most probable height profile. To a certain extent this expectation was not fulfilled, because the radio altimeter showed a number of errors which make its interpretation difficult:
- Before and behind the runway it follows the ground profile, which is usually not flat and may be different for successive runs of similar nature because of small differences in ground track.
- This problem is aggravated by the fact that different surfaces (e.g. concrete and grass) have a significant effect on the calibration of the radio altimeter.
- When the roll angle of the aircraft differs significantly from zero, the radio altimeter does not give the height above the ground because of the limited width of the antenna cone. This may again be aggravated by changes from concrete to grass. Similar errors occur when the pitch angle changes.
- The radio altimeter used showed brief periods in which is went into "memory" and brief spikes. These complicate automatic processing, but are reasonably simply recognized by a human observer.
- The calibration near the ground up to a few metres is nonlinear and is sensitive to pitch effects. This is being investigated in more detail.

These effects cause problems in an automatic data processing, but can be well understood by a human operator, especially if he handles a series of similar runs. When used in this way, the radio altimeter can provide a very useful check on the height calculation.

5 DISCUSSION OF THE RESULTS

5.1 Horizontal distances

For the tests at Brétigny the distances along the runway calculated by STALINS were compared with those measured by the STRADA laser theodolite. As mentioned above, STRADA showed errors of up to 5 m over roughly 1200 m in the centre of the runway. This error reproduced roughly, but not exactly, as a function of the distance along the runway. It was found to be possible to construct a "most probable error" line along the runway. A "most probable correction" was constructed, which is thought to be correct within ± 1 m.

The STALINS values were first calculated by the automatic programme and the differences with the "corrected" STRADA were plotted. In many cases the differences were small, but in some cases differences of up to 10 metres occurred. After application of the acceleration correction derived from the "Schuler curves", the speed errors calculated from the RASP distance were all within the quantization step in the speed (0.04 m/s). When the speed correction based on the two RASPs was also applied, the differences between the STALINS distances and the corrected STRADA distances were all within 1 metre. This showed that the requirements of section 2 were met.

The landing measurements at Ypenburg were operational measurements made with the ROBOT cameras. The STALINS equipment was added to obtain additional experimental information. The results have not yet been fully analyzed, but they seem to confirm the conclusions drawn from the Brétigny measurements. These operational tests provided a few new insights which can be important for future applications:

- The tests were made in several groups of a few measuring runs with relatively long periods (15 minutes or more) between successive runs in the same group. These long periods between runs were partly caused by the fact that the measurements were high-energy landings, after which the brakes had to be cooled in the air, partly by interference from other traffic. It was found that it was not possible to reconstruct the "Schuler curves". This was, in this case, not a large drawback because the time period from 50 ft height to standstill was relatively short (less than 50 seconds) so that the "acceleration" correction was small. Corrections only based on a constant-speed error gave reasonably good results.
- The distance between the RASP antennas was relatively short (300 m, against 1000 m at Brétigny). This resulted in less accuracy in the RASP check. In future measurements the distance between the RASP antennas should be as large as possible.
- During several landings the aircraft did not come to a complete standstill. In some such cases the automatic programme did not find a sufficiently long period of standstill and did not give results at all. In other cases, however, the aircraft speed stabilized long enough at a low value for the automatic programme to detect a "standstill". Then wrong horizontal distances were calculated which were, however, detected by the RASP check.

5.2 Height

The tests at Brétigny were conducted with the vertical channel modified by Litton. During the data processing it was found that in all landings and RTOs the acceleration had exceeded the new range of 0-2 g and that no trajectory could be calculated. Further investigation showed that the exceedance had occurred when the aircraft was already on the ground and strong braking was applied. The cause was a high-frequency (18 Hz) fuselage vibration which exceeded the measuring range, in general only once during a very brief time period. But this provided a completely wrong height trajectory, which could not be corrected.

Typical results of the take-off measurements at Brétigny are shown in figures 3 and 4. Figure 3 shows the difference between the height according to the radio altimeter and the height calculated from STALINS (transformed to the same point on the aircraft) against time; figure 4 shows the difference in heights according to STRADA and STALINS. The oscillation just before the aircraft reached a height of 11 m in figure 3 does not occur in figure 4. It occurs in a similar way for all flights and must be interpreted as an error of the radio altimeter, caused by a non-linearity in its calibration at low heights that was not taken into account during the data processing and by the sudden pitch-up of the aircraft. If this part is smoothed, both figures show that the error in the STALINS calculation is well within the requirements of section 2 in the critical range up to 11 m altitude. Beyond that both figures show a slight increase in the difference to about 0.5 m at an altitude of 65 m (1420 seconds in the graph). The steeper increase in figure 3 beyond that point is due to turning of the aircraft roughly at the end of the runway.

The results of the Ypenburg tests are still being investigated. For the majority of the runs figures similar to those obtained at Brétigny are found. For a few runs the differences of the STALINS altitude with both the radio altimeter and the ROBOT altitude are larger. These are still being investigated, but it seems likely that spurious pulses may have occurred there. They are clearly detected by the second-stage criteria (comparison of accelerations of gravity and comparison with radio altimeter) and can be corrected on the basis of that information.

6 CONCLUDING REMARKS

Although the reference methods against which STALINS was compared did not have the surplus in accuracy that one would have liked, the results of the tests described in this report provide strong evidence that the STALINS method described in this paper meets the requirements specified in section 2.

Negotiations with the manufacturer of the inertial system are now under way to find optimal performance of the system even when high vibrations are present. It seems likely that some improvement can be achieved.

The computer programme is being finalized, so that it will provide the information required for the second stage of data processing in an optimum way.

The system is expected to be fully operational, with delivery of trajectory data within 24 hours, in the course of 1981.

7 REFERENCES

1. Vleghert, J.P.K.: Determination of take-off and landing data with an airborne forward looking camera combined with accelerometers. NLR TN V-1933 (1965).
2. Breeman, J.: Start- en landingprestaties met het LN-3 traagheidsnavigatiesysteem. NLR TR 75027 U.
3. Willekens, A.J.L. and Pool, A.: Evaluation of a method for measuring take-off and landing performance of aircraft using INS. NLR TR 77050 L.
4. Pietersen, O.B.M. and Klinker, F.: A slotted wave guide antenna as a marker beacon for aircraft. NLR MP 75027 U.

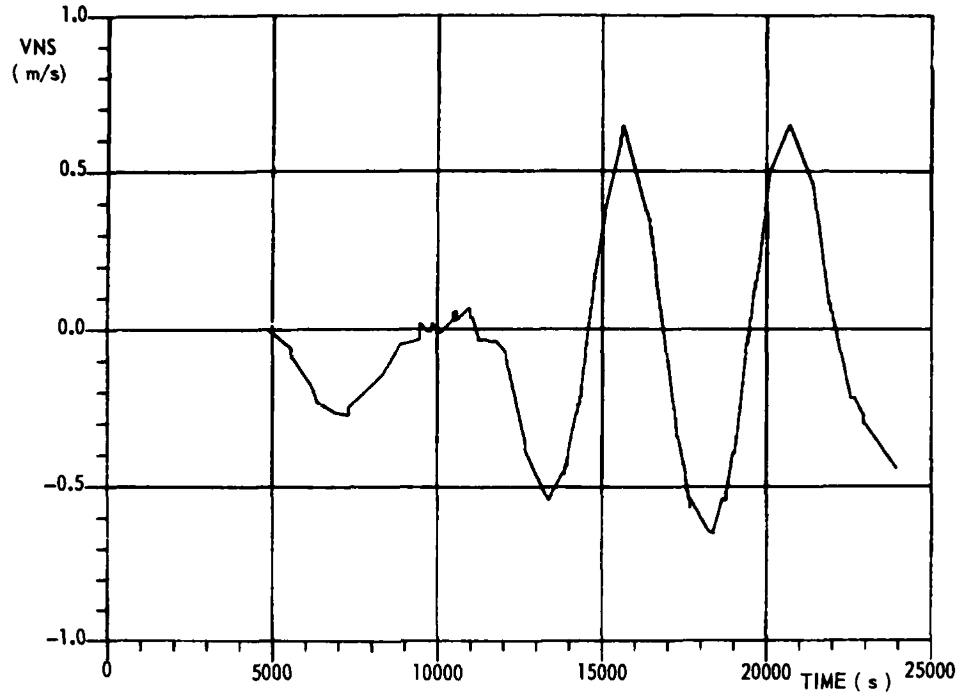


Fig. 1 The values of the north-south velocity **VNS** measured during standstill for the Brétigny tests

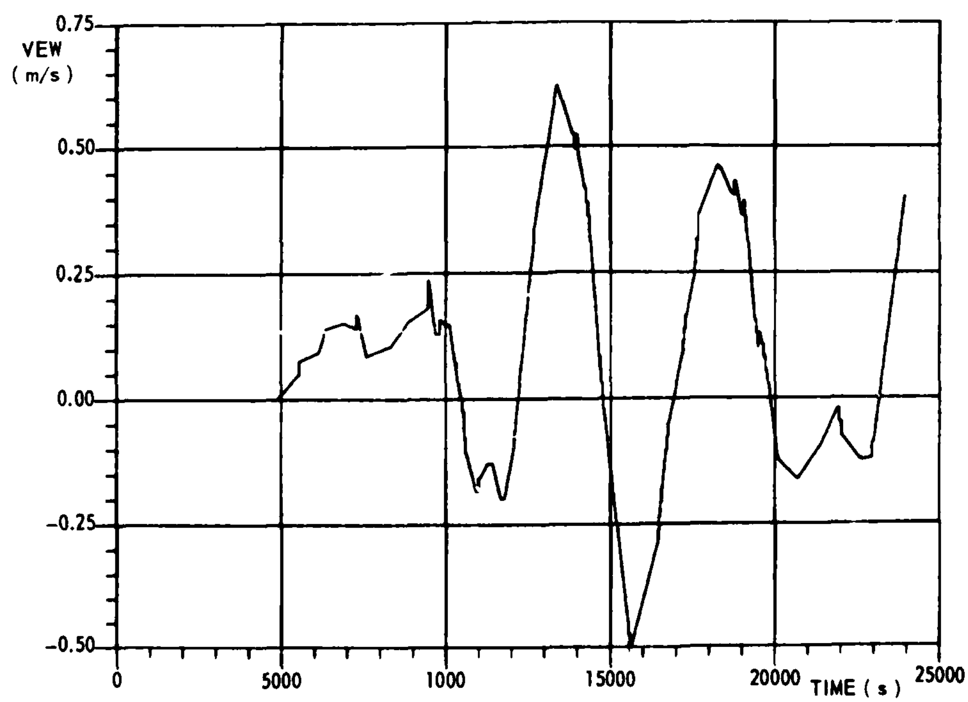


Fig. 2 The values of the east-west velocity **VEW** measured during standstill for the Brétigny tests

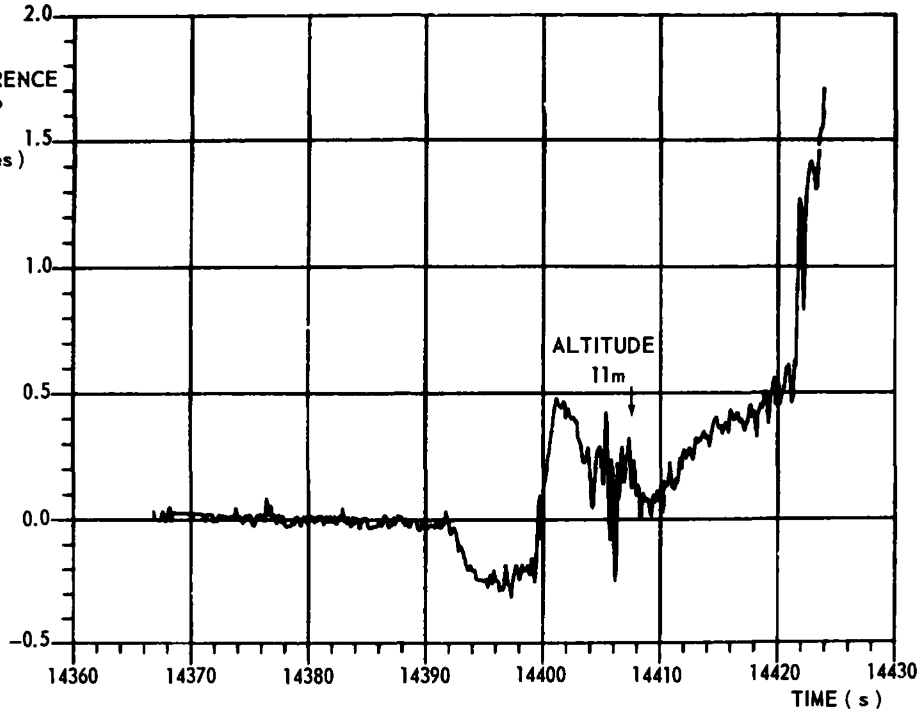


Fig. 3 Difference between height according to the radio altimeter and **STALINS** for a take-off at Brétigny

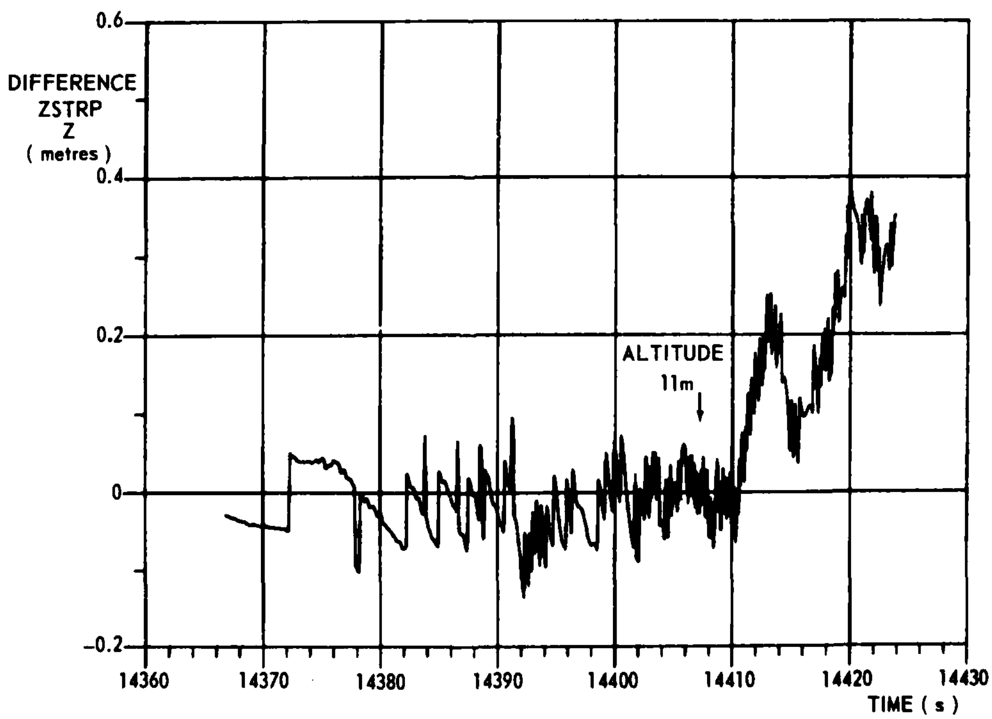


Fig. 4 Difference between height according to **STRADA** and **STALINS** for a take-off at Brétigny

MICROWAVE SYSTEM FOR
REAL TIME SPACE POSITION MEASUREMENT

W. J. Irwin
The Boeing Company
Seattle, Washington USA

SUMMARY

This paper discusses a Microwave Airplane Position System (MAPS) for measuring the space position of a flight test airplane and time-correlating this position with other varying test parameters. Operational range is normally within 10 km of the test range or runway.

A number of microwave transmitter/receiver (T/R) units are located at surveyed coordinates in an optimum ground pattern. Airborne equipment includes an interrogator, digital processor, data storage units, pilot's guidance indicators and a quick look engineering station.

The airborne system interrogates each ground T/R in serial fashion and computes slant range and range rate from the response. The computer performs position calculations in real time using a high speed Kalman filtering algorithm. Data are tape recorded, displayed to test engineers, and used to drive panel instruments which allow the pilot to follow a specific flight profile.

INTRODUCTION

Background

In flight testing there are many conditions where the three-dimensional position of the airplane and its derivatives with respect to time are important parameters. About a dozen tests of this nature, performed at low speed and altitude and close to an airport, are routine in any major test program.

For many years we have determined position by using tracking phototheodolites and specialized airborne camera systems. In particular, The Boeing Company's forward looking and down-looking camera systems achieve 2σ accuracies from about ± 3 meters down to $\pm 1/3$ of a meter, depending upon the test conditions. Accurately surveyed ground targets are required; three or four must be visible in each film frame before the camera position can be computed. Sophisticated calibration, reading and corrective techniques are necessary to obtain accurate data. Weather and ambient lighting often hinder testing. The data processing and analysis are slow, and the waiting time is costly, particularly if data turns out to be unsatisfactory.

There is a great need for (1) real time guidance information to assist the pilot in performing his tasks; (2) quick look engineering information onboard the test aircraft to determine if the test objectives are being met; and (3) rapid turn-around on final data. These requirements can be met only with an electronic ranging and data processing system operating in real time during the test.

Methods investigated as possible candidates for an upgraded position measuring capability included the use of inertial systems, radar and laser trackers with air to ground telemetry, and several forms of microwave distance measuring equipment. The conclusion was that a microwave system, using a combination of ground-based and airborne components, including an airborne minicomputer, would best meet the particular requirements.

Today, noise testing is a significant factor in programs to upgrade current engines and aircraft. It will be very important in certifying the 767 and 757 models. Noise testing has been the stimulus for allocating Boeing funds and manpower to the development of an advanced space positioning system. Initially the emphasis is on meeting the requirements for noise testing under Federal Air Regulation Part 36 (FAR 36). However, during design of the new system, the higher accuracy requirements and special problems of performance testing have been anticipated.

General Requirements

Boeing Flight Test wanted the following in a system.

1) Air transportable on short notice in the test airplane to any of several distant locations.

2) Quick deployment without special alignment or calibration.

3) Independence from ambient lighting or crew visibility.

4) No modification of airplane structure or production systems which belong to a customer.

5) No externally mounted components which affect aircraft performance.

6) Data provided within a 10 km radius and a 3 km height from a test site (which need not necessarily be an airport runway).

7) Digital data output recorded on magnetic tape for post-flight analysis.

8) Real-time guidance displays for the cockpit crew.

9) Real-time engineering data in the aircraft for onboard decision making during testing.

10) A summary display immediately after completing a test condition, showing time-correlated space position and other aircraft and engine performance parameters.

11) Telemetry of selected summary data to a ground station for merging with ground-based measurements.

12) Accuracy at least sufficient to meet the position tolerances for noise testing per FAR 36.

13) System to be fully evaluated and operational before the 767 aircraft certification program, scheduled to begin in the Fall of 1981.

14) Hardware calibration and maintenance, and data base maintenance, to be performed in the Seattle Flight Test Center, without dependence on remote base facilities.

Noise Testing

FAR 36 specifies how compliance with aircraft noise limits will be demonstrated. Without going into detail, the conditions can be summarized as follows (see Figure 1):

1) Take-off overhead noise at a microphone 6500 meters from brake release on the extended runway centerline. Simulated take-offs are made from low level passes; after rotation at a predetermined point, climb is at constant airspeed.

2) Sideline noise at four or more microphone stations 450 meters on either side of the runway in the region of greatest noise. Level fly-by's are used.

3) Approach overhead noise at a microphone 2000 meters from the threshold on the extended runway centerline. A 3 deg. $\pm 1/2$ deg. approach is flown at constant thrust and airspeed.

Weather is restricted per FAR 36 with respect to temperature, humidity, precipitation, wind and turbulence. At test sites from 300 to 3000 miles from home base, the weather window will be open only for a limited time; perhaps for several days, or sometimes for less than one hour every day. This is why a highly mobile system is needed; ground transportation of tracking equipment to remote sites is impractical in this kind of testing.

Interim System

To gain experience with DME equipment, Boeing acquired an interim microwave radar system in 1978 for evaluation. This is a pulsed system using five ground transponders, with an interrogator and distance measuring unit (DMU) in the test airplane.

The master unit in the airplane interrogates each transponder in fixed sequence and determines the slant distance from the time delay of the return pulse. Ranges are tape recorded and also processed in an airborne computer using a three-sphere solution algorithm to compute airplane, x, y, and z space coordinates. This experience has produced the following results.

1) Evaluation of the inherent hardware limitations of magnetron pulsed radar.

2) Verification of computer studies of accuracy requirements.

3) Evaluation of various transponder arrays.

4) Development of interface equipment for the recording system.

5) Development of application programs for processing data in flight.

6) Demonstration of the need for more sophisticated methods of real time calibration and wave velocity correction, and for faster algorithms for computing velocity and position.

7) Pilot's experience with a display of computed distance-to-go. Requirements were established for vertical and lateral guidance instruments.

The interim equipment did not have enough accuracy or growth potential for further development, so a specification was written for a more advanced system. The Cubic Corporation was selected, based on competitive bids, to develop a new system called MAPS. The system has not been delivered yet, so this paper can only discuss how it is intended to work. After experience is gained in testing over the next year or so, perhaps there will be a subject for another paper.

SYSTEM DESCRIPTION

Airborne Data System

MAPS has a stand-alone capability, but as normally used it will interface with a High Speed PCM System (HSPCM) and with an Airborne Data Analysis/Monitor System (ADAMS), as shown in Figure 2. To the rest of the data system, MAPS looks like any group of digital transducers. Its timing is controlled by the HSPCM system, its output is recorded on magnetic tape and is immediately available to ADAMS for further processing and display. Its pilot's indicators provide position information in real time for controlling the aircraft's flight path.

The HSPCM system is a serial, ten-bit system; two adjacent words can be used for data requiring more than ten bits. The sampling rates and sequence are programmable in the Remote Multiplexer Digitizing Unit (RMDU). A data cycle is 200 msec; during this time all channels are sampled sequentially and data is serially output and recorded on magnetic tape. At the end of each cycle the RMDU outputs an end-of-cycle (EOC) pulse. This is transmitted to MAPS for timing purposes, along with inhibit and clock pulses to enable the transfer of data out of the MAPS buffer for recording.

The ADAM System obtains its inputs either directly from the PCM data stream or from a read-head on the flight tape transport. ADAMS has its own Rolm 1666 computer and applications programs for various purposes. With respect to MAPS, its task is to interpolate between adjacent time-correlated space position samples to calculate the exact time at a specified target, such as over a microphone, to merge this data with related engine and airplane parameters, and to produce summary data for decision making immediately after a test condition. Output devices are CRT screens and a high speed line printer. Summary data can be edited and telemetered to the ground.

MAPS System

Battery powered transponders with omni-directional antennas are placed at surveyed ground points in an optimized array. A set of Cartesian coordinates with a designated origin is defined with respect to the array. The airborne equipment consists of an interrogator, omni-directional antenna, distance measuring unit (DMU), Rolm 1666 computer and accessories, floppy disk, keyboard/CRT, high speed printer, and guidance instruments in the pilot's panel.

The RF tracking equipment is based on Cubic's CR-100 precision ranging system, which has been produced for more than a decade. Modifications have optimized the hardware for the MAPS application. It is a frequency modulated carrier wave system which uses phase shift on the return of four harmonically related modulation tones to measure range, and the Doppler shift of the carrier frequency over the round trip path to determine range rate as each transponder is serially interrogated. A small cross-dipole antenna in a 7 cm high fiberglass radome on the bottom of the fuselage handles the two-way communication.

Each transponder has a unique identification code and responds only when queried. Under computer control the interrogator calls one transponder at a time at intervals of 25 milliseconds. This provides 40 samples per second, or typically 5 samples per second from each of eight transponders. The PCM data format can handle up to 19 transponders, and only three to six transponders per data cycle are required to obtain tracking solutions, so there is considerable flexibility and growth in the system.

The interrogator, after sending a specific transponder identifier, superimposes a composite of four modulating tones on the carrier during part of the 25 millisecond interval. Each transponder, upon recognizing its ID, turns on and retransmits the same tones on a shifted carrier. Back in the airplane, the returned signals are demodulated and frequency components are compared. After a settling time, the slant range is measured as a function of phase lag in the returned modulation tones. Range rate is averaged by clocking the carrier frequencies over a short time interval to determine the Doppler shift. The half wave length of the fine tone is 51.2 meters; dividing this with a ten bit digital phasemeter gives a range resolution of ± 5 cm. Similar treatment of the other tones, all related in frequency by a factor of $\overline{16}$, provides a continuous encoded range out to 210 km without ambiguities. Overlapping bits provide data quality checks and the effect of bit errors is small, since only the six least significant bits out of the 22 bit range word are not redundant or subject to limit checking.

The digital output of range rate has a resolution of ±9 mm/second and a maximum value of 288 meters/second.

For each interrogation, the output to the Rolm computer is a block of seven parallel 16-bit words containing transponder ID, data quality indicators, raw range and range-rate, time since EOC, and other parameters. MAPS timing is normally based on receiving an EOC pulses at 200 millisecond intervals; however, for stand-alone operation in the absence of these pulses, it has the capability of generating its own timing signals. MAPS calibrates itself at set intervals by dropping one interrogation and using that 25 milliseconds to measure its internal time delays. The total delay is converted to an equivalent range correction and applied to all range measurements until the next calibration cycle. This procedure eliminates errors from drifts in the airborne transmit/receive circuits.

The airborne Rolm 1666 computer in MAPS is identical to the one in ADAMS. Interchangeable components simplify spares, maintenance and allow hardware to be used on other aircraft when MAPS is not in use. Equipment mounts in standard 19-inch racks. The processor has 64K words of memory. The computer operates under Data General Corporation's Real Time Operating System (RTOS). Peripherals include a Boeing-designed keyboard, 17-inch CRT, Miltope floppy disk, and Datametrics line printer. Under system control, the software consists of routines for initialization, space position solutions, flight path processing, equipment failure reporting and interrupt handling. Figure 3 illustrates the interfacing of the computer with other equipment.

The pilot's display consists of a digital indicator showing distance to go in meters to a designated point on the programmed flight path, plus an analog instrument, similar to an Instrument Landing System (ILS) indicator, where moving bars show lateral and vertical deviations from the intended path. Sensitivity or scale of both indicators can be changed by keyboard command. The update rate of the digital display also is a keyboard variable. These instruments allow the pilot to fly the airplane on a predetermined 3-D flight profile, knowing his position within a small volume at all times. Figure 4 shows these indicators.

The transponders are rugged, low power devices operating from two 12-volt automobile or motorcycle batteries. They are placed at surveyed points and left unattended. Low power drain during stand-by, and the short "ON" time when interrogated, allow several days between battery charges. The coordinates of each transponder are entered in the data base. Transponders are located to give maximum space position accuracy for a particular test. For example, z or height above the ground is most accurate when the airplane is directly above a transponder; x and y coordinates of the airplane are most accurate when ranges from two transponders at 90 degrees are measured. Thus, a common array for noise testing is a line of transponders underneath the flight path with one outrigger well off to one side, as shown in Figures 5A and 6. Other configurations have also been tested. The shape of the z error curve is illustrated in Figure 7.

The primary data outputs from MAPS are the computed x, y, z coordinates of the airplane antenna as a function of time. Raw range and range rate have to be corrected for calibration factors and atmospheric index of refraction. Then the corrected values are processed to obtain space position coordinates which are displayed in real time to the pilot and test engineers. They also are recorded on the flight tape along with data quality bits, raw range and range rate words, and time since EOC. Since EOC pulses and IRIG time also are recorded, MAPS data can be time correlated with other measurements, and it also can be reconstructed later by processing raw data in a ground-based computer if desired. During a data cycle, as data from each transponder is processed, it is stored in the output buffer as seven ten-bit words. At the end of the cycle the latest airplane x, y, z coordinates and the corresponding time from EOC are recorded in a similar block. Space position coordinates are output once per data cycle, or five times per second.

MAPS software has four modes of operation. Mode switching is by keyboard command or in some cases is automatic.

1) Initialization mode includes start up, loading the data base into the CPU from the floppy disk, making modifications to the data base, and confirming system operability.

2) Preflight mode allows ground testing before flight, including exercising the pilot's instruments.

3) Flight Operational mode is the normal mode for calculating airplane position. Certain criteria determine the algorithm to be used. Normally in flight a Kalman filter routine is used to perform a position solution every 25 milliseconds, updating the solution after each transponder is interrogated. A Kalman filter uses weighted values of previous state parameters (position and velocity components) to predict new state values at the next sample time. Then at that time the measured values are compared to the prediction, and the estimate is updated. The state parameters are carried along at 40 updates per second with only small corrections and small errors. However, the Kalman filter requires reasonable

initial data from which to begin predicting. To initialize the filter, the software first selects the best transponders based on availability, quality of their response, and best geometry. It then performs a crude estimate of aircraft position using a least square algorithm involving three transponder ranges extrapolated to a common time (see Figure 8). After a number of iterations to form an improving estimate of position, a linear fit is constructed through the settled data to give an initial position and velocity. During this process initial values for terms of the position and velocity error covarience matrices are determined. Then the Kalman filter is started and the solution converges rapidly to minimize the errors; the whole process takes only a few seconds. This procedure allows MAPS to start or restart the position algorithm from any position on the range with no prior knowledge of the aircraft position. It also is automatically followed if the airplane is on the ground and the filter algorithm cannot obtain a solution until a sufficient altitude is reached to give the proper geometry.

4) Ground tracking is a fourth mode which allows the Kalman filter to be used with fixed z and zero vertical velocity. The filter then estimates and updates only in x and y. This mode allows precise tracking of the aircraft during take-off, refused take-off and other ground operations.

SYSTEM EVALUATION

MAPS should be delivered to Boeing with static testing completed by the end of this year. Dynamic testing on an airplane will begin early next year. There is no method of determining the true position of an airplane in flight, and MAPS should be as good as, or better than, any other instrumentation with which it can be compared. Therefore, evaluation will have to be statistical, comparing position, velocity and acceleration data derived from MAPS, INS and IRS, photo theodolite, radio altimeter, Doppler radar, accelerometers or other available sources. Before the 767 flight test program begins, MAPS should be fully evaluated, and accepted for noise certification.

Error analyses predict root sum square errors (1σ) of ± 0.16 meter for range and ± 0.008 meter/second for range rate. The accuracy of the computed x, y, z coordinates depends upon how the geometry of the array amplifies these errors. Using two properly located, well separated transponders, x and y accuracy of this magnitude can be achieved. The same is true for z when the airplane is directly over a transponder. The error in z is a strong function of the elevation angle of the slant range vector; however, with proper spacing of transponders, and switching based on optimum geometry, z error can be controlled with acceptable limits for a given test.

Error analyses include the estimated effects of resolution, stability, phase jitter, velocity, acceleration, multipaths, timing, signal strength, temperature, atmospheric index of refraction, and array geometry. They indicate that the accuracy requirements for noise testing, shown in Figure 9, should be met with no difficulty at altitudes above 60 meters.

SYSTEM APPLICATION

The first application of MAPS will be for noise testing. The goal is to make it the sole source of space position data, eliminating the present photo theodolite equipment from these tests.

Application to certain performance tests will be investigated as time allows. The ground operating mode designed into MAPS will allow it to provide data during take-off roll, refused take-offs, landing roll, thrust reverser tests, minimum ground control speed tests, rollout drag tests, etc. Airborne applications can include low altitude static source calibrations, and approaches including multi-segmented profiles. Special techniques and arrays will aid in expanding its capability as experience is gained.

CONCLUSION

In conclusion, MAPS is expected to provide real time position information which will facilitate the completion of noise testing. It also has potential applications in performance testing. It will aid the crew in flying the desired test conditions, and before the aircraft lands they should know if the test objectives have been met. By reducing reflights, higher utilization of test airplanes should be possible, and costs should be reduced. If the expected accuracy of MAPS is achieved, it will provide a significant improvement over the camera systems now in use.

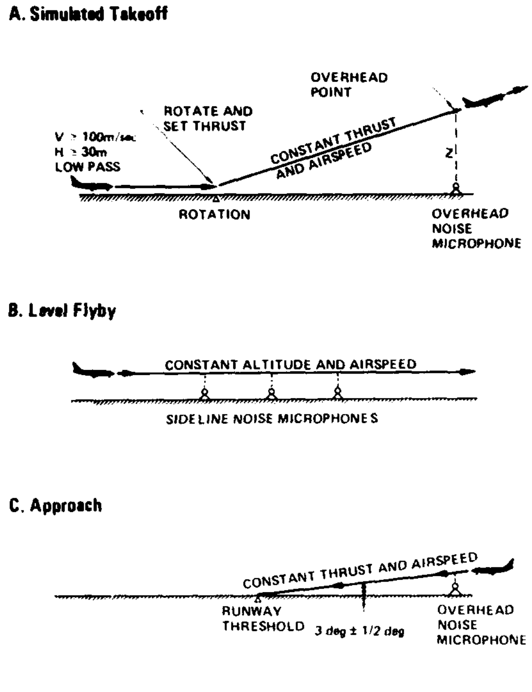


Figure 1. Noise Condition Profiles

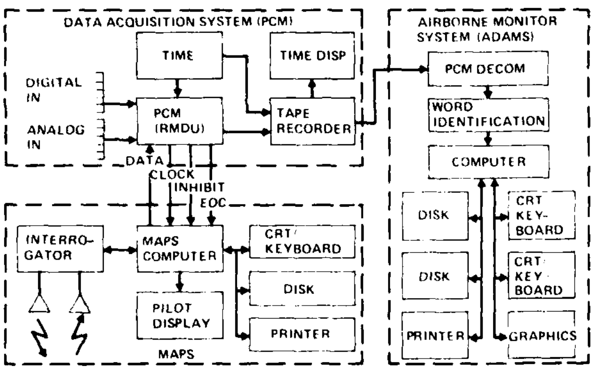


Figure 2. Airborne Systems

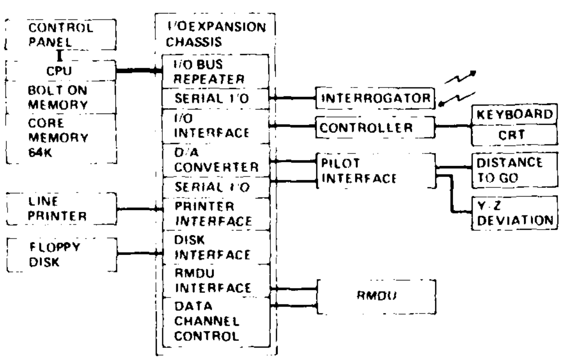


Figure 3. MAPS Component

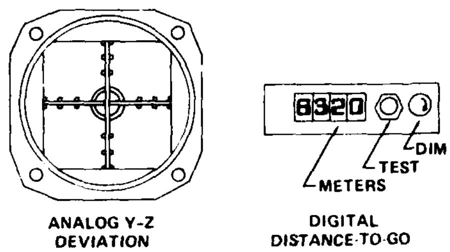


Figure 4. Pilot Guidance Instruments

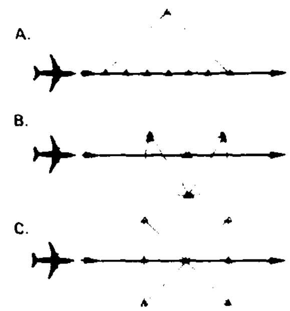


Figure 5. Transponder Arrays

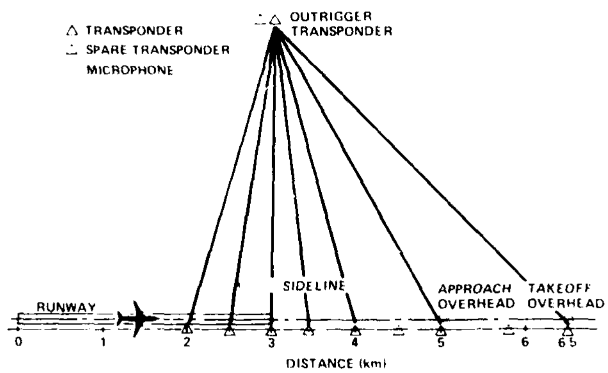


Figure 6. Typical Noise Test Array

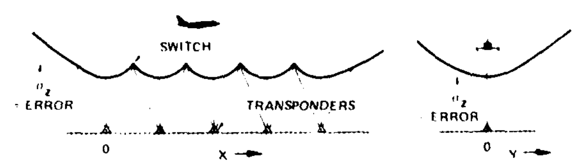


Figure 7. Error in z as Function of Airplane Position

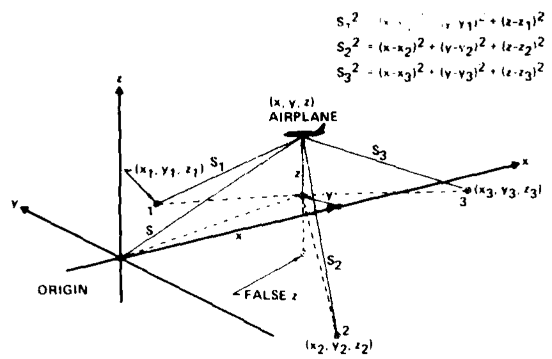


Figure 8. Three-Transponder Measurement of Airplane Position

PROFILE	PARAMETER	RANGE MIN	RANGE MAX	2σ ACCURACY METERS
SIMULATED TAKEOFF	x	0	9,000	±5
	y	-150	+150	±5
	z	90	900	±3
	Time			±0.001/sec
FLYBY	x	-900	900	±5
	y	-90	+90	±5
	z	90	600	±3
	Time	–	–	±0.001 sec
APPROACH	x	-900	900	±5
	y	-90	90	±5
	z	90*	300	±3*
	Time			±0.001 sec

*FOR z < 90m, ACCURACY MAY BE RELAXED TO ±5 METERS AT z = 60m

Figure 9. Position Accuracy Specified for Noise Tests

PRACTICAL ASPECTS OF INSTRUMENTATION INSTALLATION IN SUPPORT OF SUBSYSTEM TESTING

Robert W. Borek
NASA Dryden Flight Research Center
P. O. Box 273
Edwards, California 93523
U. S. A.

SUMMARY

Two topics pertaining to the practical aspects of instrumentation installation are discussed. First, some of the problems associated with using military specification MIL-W-5088H as a guideline for wire gage selection are discussed. Examples of proper use of this specification as a criterion for interfacing wire bundles and connectors are provided.

The second topic, which is discussed more briefly, is a technique developed for NASA known as sneak analysis. The quantitative results of 22 projects that have used this technique are reviewed, and examples of sneak analysis are given.

1.0 INTRODUCTION

Instrumentation system installation has many practical aspects; this paper is limited to a discussion of two topics related to practical installation design.

Experience at the NASA Dryden Flight Research Center has shown that the guidelines used to assist in the determination of wire sizes, connectors, and resulting wire bundle sizes require a thorough understanding before being used. An example of one such guideline is military specification MIL-W-5088H (Ref. 1), which concerns the wiring of aerospace vehicles. This paper provides some of the information that would be helpful in using this document.

Failures in aircraft and aerospace vehicle instrumentation systems must not affect the safety of the aircraft or its crew, and for this reason the systems and subsystems must be very carefully interfaced. One of the techniques used to review interface designs is referred to as sneak analysis. Sneak analysis is the other major topic of discussion in this paper.

2.0 AEROSPACE VEHICLE WIRE GAGE SELECTION

Various specifications govern the installation of instrumentation systems in military aircraft. However, military specification MIL-W-5088, entitled "Wiring, Aerospace Vehicle," probably has more influence than any other. This specification was initially published in March 1951, superseding specification AN-W-14a, which was issued in May 1945. The specification outlines the selection and routing of wiring used in aerospace vehicles and was intended to establish a design-selection criterion for basic aerospace vehicle wiring. In the 30 years since its inception, sufficient wire bundle information has been incorporated in this specification to permit its use as a guide for instrumentation installation. The document is the basis for most of the tables that list the continuous duty current-carrying capacity for single wires and wire bundles used in aerospace vehicles. However, more information than is usually provided by such tables or by excerpts from this military specification is necessary for the proper utilization of the document. The material below gives some of the information that would be helpful in using the military specification.

2.1 Use of MIL-W-5088H in Wire Gage Selection

Any wire carrying current in an aerospace vehicle electrical system must satisfy two basic criteria: it must be able to carry the required current without overheating, and it must carry the required current without greater than specified voltage drops. Both of these criteria can be met by selecting the proper wire size for the stated maximum current. The wiring table shown in Figure 1, which is taken from MIL-W-5088H (the version of MIL-W-5088 published in July 1979), was prepared for the guidance of the engineers and technicians responsible for selecting electrical wiring. However, this table departs from the wire current rating tables shown in previous versions of this military specification, and it fails to mention all the information necessary to its proper use. First, the table no longer applies to single wires; second, the basis for ampere rating is different, as explained in footnote 2; and third, the issuance fails to mention that the specified wire sizes are AN (aircraft number) designations, the size designations used for military aircraft (Ref. 2). The omission is important, because, as shown by the following discussion, this gage designation is susceptible to confusion with AWG (American wire gage) designations (Ref. 3).

The differences between the two gage designations are illustrated by Table 1, which is based on Reference 4. First, AWG sizes pertain only to solid wires. However, stranded wiring is often referred to as being "AWG equivalent" in size. The difficulty is that there are variations in the circular mil area of the wires so designated. (In Table 1, the AN wire gages alluded to in MIL-W-5088H are identified by asterisks. In this paper, only the remaining wiring is referred to as "AWG equivalent.")

To avoid confusion, most wire manufacturers include stranding characteristics or circular mil area or both in addition to wire size. The importance of this practice is shown by a comparison of Tables 1 and 2: the sizes listed in Table 2 as AWG sizes would more accurately be referred to as AN sizes.

The important thing is that based on circular mil area the current-carrying capacity of a wire of a given AN gage is not the same as that of a wire with the same AWG designation. Table 1 shows that AN wire sizes 22, 20, and 18 have a greater circular mil area than either AWG solid or AWG equivalent wiring. Further, AN wire size 16 has less circular mil area than either the AWG solid or AWG equivalent wiring. These variations are also reflected in the outer diameter and weight of each wire gage. Since MIL-W-5088H is based on AN gage size, it is imperative that wiring identified according to AN gage size be used.

2.2 Use of MIL-W-5088H in Wire Selection and Bundling

In order to illustrate the use of MIL-W-5088H as a design guideline, applicable figures from this document have been reprinted in this section.

The current rating for a single wire in free air (that is, a stranded AN gage wire) is shown in Figure 2 (Fig. 4 in MIL-W-5088H). For clarity, only wire sizes AN 24 to AN 6 are shown (wire sizes 4 to 4/0 have been omitted). The figure relates wire current-carrying capacity in amperes to environmental and wire temperature ratings as measured at sea level.

Current derating curves for wire bundles are shown in Figure 3 (Fig. 5 in MIL-W-5088H). This is the first complete bundle derating graph to appear in this military specification. Before this issuance (H), bundle loading was fixed at 20 percent. This graph includes 40, 60, 80, and 100 percent loading and has increased the usefulness of the specification.

The current derating curve for altitude is shown in Figure 4 (Fig. 6 in MIL-W-5088H). The altitude information as shown here has been augmented to include altitude in kilometers.

The following two examples illustrate the use of these figures. For the first example, assume that a wire bundle consisting of 12 AN 20 gage, 200° C (392° F) rated wires are to operate at an ambient temperature of 25° C (77° F) at an altitude of 3 kilometers (10,000 feet). For example 2, assume an ambient temperature of 110° C (230° F) and an altitude of 15.2 kilometers (50,000 feet). Further assume that all 12 wires will be operating at maximum capacity. MIL-W-5088H can be used to determine the maximum current capacity of the wire bundle for both altitudes as follows.

(a) For both examples, the first step is to derate the wiring for ambient temperature. Refer to the appropriate curve in Figure 2 to determine the difference between the manufacturer's wire rating and ambient temperature, ΔT. For example 1,

$$\Delta T = 200° \text{ C} - 25° \text{ C} = 175° \text{ C } (347° \text{ F})$$

For example 2,

$$\Delta T = 200° \text{ C} - 110° \text{ C} = 90° \text{ C } (194° \text{ F})$$

Therefore the free air ratings of AN 20 gage wire are 28 amperes for example 1 and 20 amperes for example 2.

(b) The next step is to derate the wiring for bundling effects. Bundle derating curves are shown in Figure 3. The 100 percent loading curve is used, because each wire is to carry its maximum loading. Locate 12 on the abscissa (since 12 wires are being used); the derating factor is 0.53.

(c) Now derate the wire using the 0.53 factor. For example 1,

$$28 \text{ amperes} \times 0.53 = 14.8 \text{ amperes}$$

For example 2

$$20 \text{ amperes} \times 0.53 = 10.6 \text{ amperes}$$

(d) The wire bundle must then be derated for altitude. Refer to the altitude derating curve in Figure 4. For example 1, the derating factor for 3 kilometers (10,000 feet) is 0.92. For example 2, the derating factor for 15.2 kilometers (50,000 feet) is 0.71. To derate the wire bundle for altitude, apply these derating factors to the current values determined in (c) above. For example 1,

$$14.8 \text{ amperes} \times 0.92 = 13.6 \text{ amperes}$$

For example 2,

$$10.6 \text{ amperes} \times 0.71 = 7.5 \text{ amperes}$$

(e) The total bundle current capacity for example 1 is then

$$13.6 \text{ amperes} \times 12 = 163 \text{ amperes}$$

The current capacity for the same bundle under the conditions set for example 2 is

$$7.5 \text{ amperes} \times 12 = 90 \text{ amperes}$$

These examples show the magnitude of the difference in the amount of current the same wire bundle can carry. In selecting wire size, the instrumentation engineer may choose to design for the worst case condition (example 2) or for specific flight conditions (example 1).

In examples 1 and 2, both the size and the number of the wires in the installation were specified and supporting calculations were made to determine the maximum current capacity of the wire bundle. If the results had been unsatisfactory, new calculations using a different wire size would have been required.

The order in which the calculations are made is reversed for situations where a required current is known. If 9.0 amperes instead of 7.5 amperes of current were required per wire under the same conditions and with the same wiring installation as in example 2 above (12 AN 20 gage wires at an ambient temperature of 110° C (230° F) and an altitude of 15.2 kilometers (50,000 feet)), the only design change possible would be to alter the bundle derating factor. The only way to alter this factor would be to separate the wires into two or more bundles, reducing the number of wires per bundle. To find the maximum number of wires for a single bundle, take the following steps:

(a) Determine the free air rating of AN 20 gage wire. For a ΔT of 90° C (194° F), the rating is 20 amperes.

(b) Derate the value in (a) by the altitude derating factor as follows:

$$20 \text{ amperes} \times 0.71 = 14.2 \text{ amperes}$$

(c) Then, since the bundle size is unknown, use the current value thus determined and solve for the bundle derating factor that results in 9.0 amperes:

$$14.2 \text{ amperes} \times \text{Derating factor} = 9.0 \text{ amperes}$$

Then

$$\text{Derating factor} = \frac{9.0}{14.2} = 0.63$$

A 0.63 bundle derating factor permits only eight wires in a bundle under these conditions. At this point it might seem reasonable to use the initial design of 12 conductors, but the wiring can be separated into two bundles and then recombined at the connector into a bundle of 12. This may or may not be a good idea, however, since the interfacing connector must also be rated as a wire bundle equivalent.

3.0 CONNECTOR SELECTION CRITERIA

When flight test instrumentation is installed or modified, the wiring already existing in the aircraft is often used. Therefore, connector selection must be based partially on the interface with the existing subsystem.

MIL-W-5088H is careful to point out that the current ratings in Figure 1 cannot be directly applied to connector contacts. The military specification can be and often is used as a design aid, however, and many connector manufacturers publish connector data accordingly (as shown in Fig. 5, from Ref. 5). One of the problems associated with correlating connector and wiring specifications through the use of this military specification is that updated versions of the specification may be difficult to apply to designs based on earlier versions of the specification. For example, the maximum current ratings and test current values shown in Figure 5 cannot be easily correlated with the current ratings shown in MIL-W-5088H, although they can be directly correlated with the ratings in MIL-W-5088E and earlier issuances.

Nevertheless, the connector manufacturer's approach is correct in that it considers the effects of the wire bundling in rating the connector. In effect, a connector converts all of the wires routed through it into a bundle, particularly if the contact density is high (Ref. 6). Thus, all of the factors that limit bundle configuration should also be considered in selecting a connector.

To assist engineers and designers in the choice of connector, connector manufacturers usually provide design criteria like the time-temperature curves shown in Figure 6 (Ref. 7). Figure 6 shows the time and temperature limits of a connector designed to operate at high temperatures. The shell size is 12 and the contacts are AN 20 gage. Although the information shown represents operation at sea level, the manufacturer states that the connector will operate satisfactorily at altitudes greater than 18.3 kilometers (60,000 feet). That, in addition to the fact that the temperature curve limits are based on a connector contact temperature of 239° C (462° F) permits the information in MIL-W-5088H for wire bundles to be used to evaluate the connector. If the connector is assumed to be equivalent to a wire bundle, the effects of the dielectric block in which the contacts are positioned should be considered equivalent to the characteristics of the wire insulation. In the following discussion, the conditions specified for example 2 discussed above (an ambient temperature of 110° C (230° F) at an altitude of 15.2 kilometers (50,000 feet)) and MIL-W-5088H are used to evaluate the connector.

As shown previously, 9.0 amperes can be continuously routed through a wire bundle of eight AN 20 gage wires at 200° C (392° F). The question to be answered is this: if eight of the 12 wires in the connector represented by Figure 6 carry 9.0 amperes, can the connector tolerate the other four wires carrying 7.5 amperes? If it cannot, a new design approach will be necessary for the connector.

The problem may be approached as follows.

(a) Determine the single wire in free air rating:

$$\Delta T = 239° \text{ C} - 110° \text{ C} = 129° \text{ C} \ (264° \text{ F})$$

From Figure 2, obtain the current rating of a size 20 wire for a ΔT of 129° C (264° F). The resulting rating is 24 amperes.

(b) Refer to Figure 3 (the bundle derating curve) and derate the connector (wire bundle) for 100 percent loading. The derating factor for 12 contacts (wires) is 0.53.

(c) Now derate the free air rating from (a) by the bundle derating factor found in (b):

$$24 \text{ amperes} \times 0.53 = 12.7 \text{ amperes}$$

(d) The connector (wire bundle) must now be derated for altitude. From the altitude derating curve, the derating factor for 15.2 kilometers (50,000 feet) is 0.71.

(e) Derate the connector (wire bundle) using the bundle derating current value from (c) and the altitude derating factor from (d):

$$12.7 \text{ amperes} \times 0.71 = 9.0 \text{ amperes}$$

The 9.0 amperes from (e) indicate that each of the contacts in the connector can continuously carry 9.0 amperes. Thus, the connector can easily tolerate eight wires carrying 9.0 amperes and four wires carrying 7.5 amperes.

One further point may be of interest. From the calculations in example 2, the bundle current capacity at 15.2 kilometers (50,000 feet) for a 110° C (320° F) ambient temperature is 90 amperes. The 90 amperes can be distributed in any way deemed desirable as long as the individual wire current capacity is not exceeded, and as long as the concentration of high-current wiring does not cause uneven connector overheating. For example, 10 wires could carry 9.0 amperes each, or six wires could carry 9.0 amperes each and six wires could carry 6 amperes each. In either case the total is 90 amperes.

4.0 DETERMINATION OF CABLE BUNDLE SIZE

The installation of instrumentation and its wiring usually requires the fit of the components, hardware, and cable bundles to be carefully checked. The problems associated with routing wire bundles through existing bulkhead access holes and areas of similarly restricted space can be alleviated if the size of the wire bundle can be estimated when the design is still on paper. The following formula provides a useful method for estimating wire bundle size (Ref. 8). The estimate is made as follows:

$$D = 1.13\,(d^2N)^{1/2} \qquad (1)$$

where

D diameter of bundle

d diameter of cable or wire

N total number of wires of diameter d used in the bundle

A common type of cable to which this calculation may be applied is a twisted-shielded pair. The cross section of this type of cable is oval with a twist rather than round, and when it is measured for the value d, two values result: the maximum value (in this example, 0.61 cm (1.24 in.)), and the minimum value (0.41 cm (0.16 in.)). For the following calculations the maximum value will be used.

First, let N be 40 (the total number of twisted-shielded pairs) and let d equal 0.61 centimeter (0.24 inch). Then

$$D = 1.13\left[(0.61)^2 \times 40\right]^{1/2} \text{ cm}$$

$$= 1.13 \times 3.86 \text{ cm}$$

$$= 4.36 \text{ cm (1.72 in.)}$$

The circumference of the wire bundle would be πD, where π equals 3.14 and D equals 4.36 centimeters (1.72 inches). The calculated circumference, 3.14D, equals 13.7 centimeters (5.4 inches). The measured circumference equals 13.2 centimeters (5.2 inches).

The formula for diameter is also valid for bundles of mixed wire sizes. Consider a wire bundle consisting of wires of two different sizes, 0.14 centimeter (0.054 inch) and 0.61 centimeter (0.24 inch). There are 25 wires of the first size and 16 wires of the second. Then the diameter of the bundle is the total of the d^2N values, as follows:

$$(0.14)^2 \times 25 = 0.49 \text{ cm}^2$$

$$(0.61)^2 \times 16 = 5.95 \text{ cm}^2$$

The total of the d^2N values is 6.44 square centimeters (0.99 square inch). From equation (1),

$$D = 1.13\,(6.44)^{1/2} \text{ cm}$$

$$= 2.87 \text{ cm (1.13 in.)}$$

The calculated circumference, 3.14D, equals 9.0 centimeters (3.6 inches). The measured circumference equals 8.9 centimeters (3.5 inches).

5.0 SNEAK ANALYSIS

Another subject of concern to those involved in instrumentation is the increasing complexity of aerospace vehicles and their systems. This complexity has made it necessary to adopt a systematic approach to the verification of system and subsystem interfacing.

One such approach is a computer-aided sneak analysis technique developed for the space program (Ref. 9). Sneak conditions are hidden or unrevealed conditions existing in electrical designs which usually appear to be satisfactory, but may under a given set of conditions prevent a desired type of operation or invite an undesired type of operation. Seven sneak conditions are described below.

Sneak circuit or path: An unexpected current flow path.

Sneak timing: A problem that at an unexpected time either produces or prevents a flow of current, thereby either activating or inhibiting a function.

Sneak indicator: An indicator that displays ambiguous or faulty system operating conditions.

Sneak label: An ambiguous or abbreviated label that may result in operator error.

Sneak procedure: A set of incomplete or poorly written instructions that may result in improper system operation.

Drawing error: An error in an electrical schematic or wire list or a discrepancy between the two.

Design concern: A design weakness, such as improper or inaccurate redundancy or unnecessary circuitry.

The sneak conditions listed above are all latent problems that may develop immediately or long after initial system operation. Although sneak analysis can be applied to any electrical system, it is most cost effective to apply it to subsystems that jeopardize unique or costly larger systems.

5.1 Quantitative Results of Sneak Analysis

The contribution of sneak analysis to flight safety and subsystem interfacing can best be demonstrated by comparing the different types of sneak conditions quantitatively.

Table 3 lists the results of sneak analyses that were performed on 22 vehicles or systems. Some systems were ground based; others were airborne, such as aircraft and missiles. The column entitled "combined" consists of five types of sneak conditions (sneak paths, sneak timing, sneak indicators, sneak labels, and sneak procedures). The two remaining columns, entitled "design" and "drawing," complete the list of sneak conditions considered.

A comparison of the values in the three columns reveals that drawing errors are by far the most common sneak condition. Drawing errors outnumber other sneak conditions in 14 of the 22 systems listed. In five of the eight remaining systems, drawing errors rank second. This underlines both the difficulty and the importance of completeness and accuracy in the schematics and listings used to describe aircraft/instrumentation system interfaces.

The network tree in Figure 7 illustrates a problem of omission encountered during the development of a computer-aided sneak analysis. Since the device at point A is not identified and its circuitry routing is not identified, it is designated a high impedance point and computer analysis stops. If similar omissions exist in other areas, the documentation for the overall system will be difficult to interpret. This type of error, which in and of itself may be considered a sneak condition, also prevents the circuitry from being analyzed for other types of problems, such as sneak paths. The solution to the problem of incomplete documentation is obvious, but documentation is incomplete for most instrumentation installations.

5.2 Examples of Sneak Conditions Revealed by Analysis

Figures 8 to 14, which are taken from References 9 and 10, exemplify different types of sneak conditions.

Figure 8 shows a report on a sneak path. The condition was found during an analysis of the B-737 research support flight system. The sneak path would have allowed the 28 volt engage interlock bus to have been shorted to a ground through a suppression diode, which in turn would have caused the relay contacts to weld together. locking the system in the engage mode.

Figure 9 is an example of sneak timing. The altitude hold switch could have dropped out after being engaged. In addition to being a design concern, this is a problem of timing and is caused by the normal operation of break-before-make contacts in a relay. The problem can be eliminated by changing to a make-before-break relay.

Figure 10 shows sneak indicators. A ground indication falsely indicates that four relays are reset when only two of the four are monitored by the indicator. This problem could cause a ground test checkout to verify a circuit that could be failed.

Sneak labels are shown in Figure 11. The sneak label is on a telemetry measurement that indicates that the right landing lights are on. The label, however, indicates "landing lights on". This is an unmanned aircraft, and the measurement could cause the operator to believe that all landing lights are on when only the right landing lights are on.

Figure 12 illustrates a sneak procedure. The problem involves the lost carrier check, which is supposed to be performed between 7 minutes and 2 minutes before the vehicle is launched. If this check were to be performed, it would activate the timing circuits for the vehicle's drag and main chutes. This would cause the mission to be lost.

Figure 13, from Reference 10, illustrates a drawing error. This illustrates the problem of incomplete documentation. Since the usage of the wires listed is unknown, each is assumed to be open ended.

Figure 14 shows a sneak design concern. Part of the circuitry of a flight control amplifier is shown. If any one of the diodes in the three-phase input power fails, the capacitors used in the circuit would fail.

6.0 CONCLUSIONS

The latest issuance of military specification MIL-W-5088, which is dated July 1979, has made it possible to design wiring bundles and to select compatible electrical connectors for instrumentation systems installed in aircraft for subsystem testing. However, the military specification must be used carefully. The design and selection procedure is facilitated by the use of a simple and quick method of determining the wire bundle diameter, which is described.

Since the interfacing between the instrumentation and other flight systems must be reliable and must not affect the performance of other systems, a method was developed to detect potential flaws in the performance of the instrumentation. In practice, the method, which is known as sneak analysis, has shown drawing errors to be the most common source of error.

7.0 REFERENCES

1. Engineering Specifications and Standards Dept., Wiring, Aerospace Vehicle, Lakehurst, N.J., Naval Air Eng. Ctr., 1979, MIL-W-5088H.

2. General Services Administration, Wire, Electrical and Nonelectrical, Copper, (Uninsulated), 1960, QQ-W-343, p. 26.

3. Donald G. Fink and H. Wayne Beaty, eds., Standard Handbook for Electrical Engineers, Eleventh ed., New York, N. Y., McGraw-Hill Book Co., c.1978, pp. 4-19.

4. Standard Wire and Cable Co., Fifth ed., El Segundo, Calif., c.1977, p. 47.

5. Cannon Standard Line Connectors, Santa Ana, Calif., ITT Cannon Electric, c.1979, p. 20.

6. Roland B. Lawrence, Electrical/Electronic Interconnection Systems, Third ed., Banning, Calif., The Deutsch Co., c.1975.

7. Amphenol High-Reliability, Environment Resistant Miniature Connectors, Technical Manual—MIL-C-26500/38300/26518, Broadview, Ill., Amphenol Connector Div. [ca. 1962], pp. 2-12.

8. Louis F. Caso, Approximating Cable Diameters, Design News, Vol. 25, no. 1, Jan. 5, 1970.

9. The Boeing Co., Uncovering Electrical Problems . . . Before They Cause You a Problem [1975]. (Available from The Boeing Co., Space Div., Houston, Tex.)

10. Robert C. Clardy, The Boeing Co., Sneak Circuit Analysis of F8 Digital Fly-by-Wire Aircraft, 1976, D2-118582-1. (Available from NASA Dryden Flight Research Center, P. O. Box 273, Edwards, Calif.)

TABLE 1.—COMPARISON OF AN AND AWG DESIGNATIONS
[Ref. 4]

Wire size	AWG solid wire		"AWG equivalent" stranded wires		
	Circular mil area	Approximate outer diameter, cm (in.)	Stranding, number of wires X size	Circular mil area	Approximate outer diameter, cm (in.)
22	640	0.0643 (0.0253)	7 X 30 19 X 34* 26 X 36	700 754 650	0.076 (0.030) 0.079 (0.031) 0.076 (0.030)
20	1020	0.0813 (0.0320)	10 X 30 19 X 32* 26 X 34 41 X 36	1000 1216 1032 1025	0.089 (0.035) 0.094 (0.037) 0.091 (0.036) 0.091 (0.036)
18	1620	0.1024 (0.0403)	7 X 26 16 X 30 19 X 30* 41 X 34 65 X 36	1769 1600 1900 1627 1625	0.122 (0.048) 0.119 (0.047) 0.125 (0.049) 0.119 (0.047) 0.119 (0.047)
16	2580	0.1290 (0.0508)	7 X 24 19 X 29* 26 X 30 65 X 34 105 X 36	2828 2426 2600 2580 2625	0.152 (0.060) 0.147 (0.058) 0.150 (0.059) 0.150 (0.059) 0.150 (0.059)

*AN size referred to in MIL W 5088.

TABLE 2.—EXAMPLE OF AN GAGE WIRING LISTED AS AWG
[Ref. 4]

AWG Size	Catalog No.	Stranding	Cir. Mil. Area	Max. Bare Diam. (Inches)	Max. Resist. @ 20°C Ohms/1000 ft.	Nom. Fin. Diam. (Inches)	Wt. Lbs./1000 ft.
22	1106-4	19 x 34	755	0 033	15.2	086	7.1
20	1106-7	19 x 32	1 216	0 041	9.42	096	9.0
18	1106-10	19 x 30	1 900	0 052	6.03	108	12.1
16	1106-13	19 x 29	2 426	0 060	4.72	116	14 3
14	1106-16	19 x 27	3.831	0 074	2.99	141	21.5
12	1106-19	19 x 25	6.088	0 093	1 88	160	30 5
10	1106-22	49 x 27	9.880	0 128	1 16	194	48.0
8	1106-25	133 x 29	16.983	0 176	0.70	.243	75.0
6	1106-28	133 x 27	26.818	0 218	0 436	292	114.0
4	1106-31	133 x 25	42.615	0 272	0.274	357	173.0

TABLE 3.—SNEAK ANALYSIS DISTRIBUTION FOR 22 SYSTEMS

System	Type of sneak		
	Combined, percent	Design, percent	Drawing, percent
1	11.7	60.3	28.0
2	14.9	0.5	84.6
3	39.0	29.0	32.0
4	24.7	53.4	21.9
5	11.3	5.9	82.8
6	11.5	4.9	83.6
7	10.0	41.4	48.6
8	34.2	15.8	50.0
9	14.8	7.9	77.3
10	13.7	2.2	84.1
11	7.5	10.0	82.5
12	12.5	50.0	37.5
13	27.8	16.7	55.5
14	12.1	0.8	87.1
15	74.2	11.9	13.9
16	13.7	23.9	62.4
17	11.0	4.0	85.6
18	60.5	18.5	21.0
19	97.0	1.6	14.0
20	7.5	10.0	82.5
21	3.7	55.6	40.7
22	25.0	16.7	58.3

TABLE I. Current Rating of Wires. 3/

(See 3.8.8.1, 3.8.8.1.1, 3.8.8.3)

Conductor material	Wire size	Continuous duty current (amperes) 2/ Wires in bundles, groups or harnesses — Wire temperature rating 105°C	150°C	200°C
Copper or Copper Alloy	1/26	2	3	4
	24	3	4	5
	22	4	6	7
	20	5	8	10
	18	7	11	14
	16	8	12	16
	14	11	17	22
	12	15	22	29
	10	19	30	38
	8	26	39	50
	6	35	53	68
	4	48	72	93
	2	66	100	130
	1	78	115	148
	1/0	89	135	173
	2/0	106	159	203
	3/0	125	186	241
	4/0	146	220	287
1/ Aluminum	8	17		
	6	23		
	4	30		
	2	43		
	1	50		
	1/0	57		
	2/0	68		
	3/0	76		
	4/0	86		

1/ The use of these wires requires procuring activity approval.
2/ Rating is for 70° C ambient, 33 or more wires in the harness with no more than 20% of harness current capacity being used, at an operating altitude of 60,000 feet.
3/ Current rating of wire terminating hardware is not covered by this table.
Notes: 1. Ratings are for copper conductors size 4/0 through size 22 and copper alloy for size 24 through 26.

Figure 1. Current ratings of wires from page 42 of MIL-W-5088H.

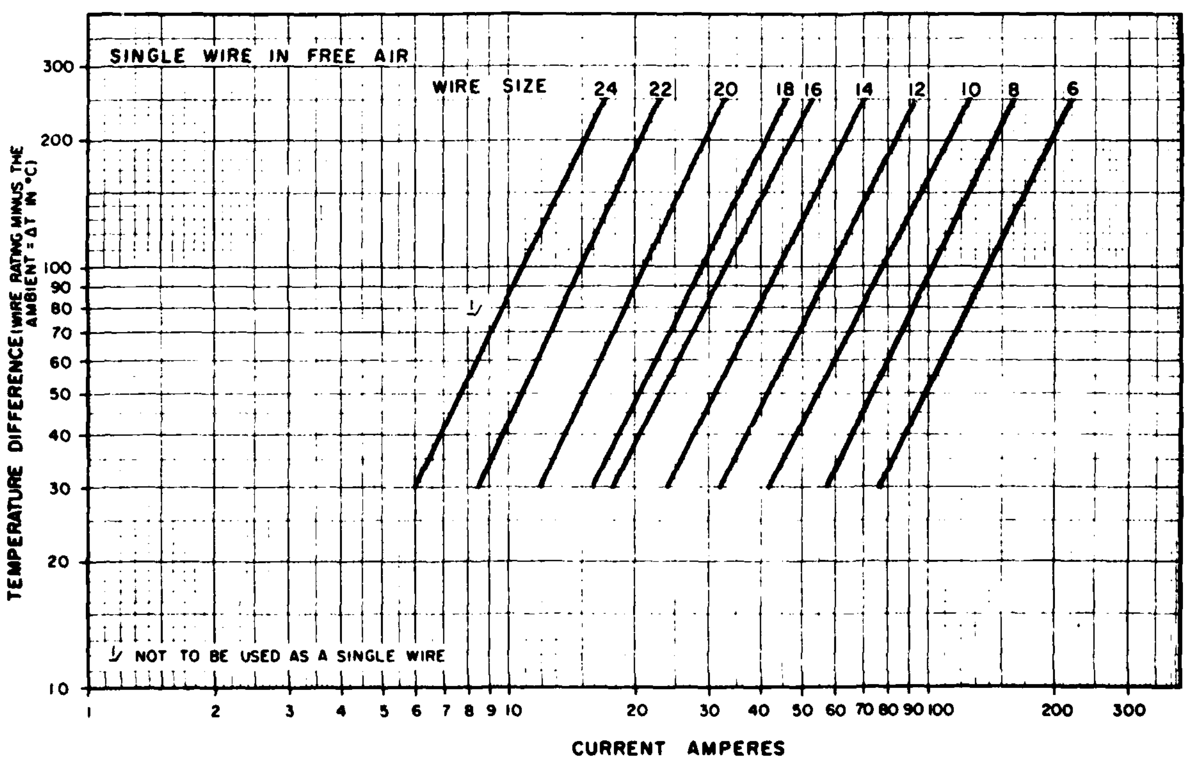


Figure 2. Current ratings for single wires in free air.

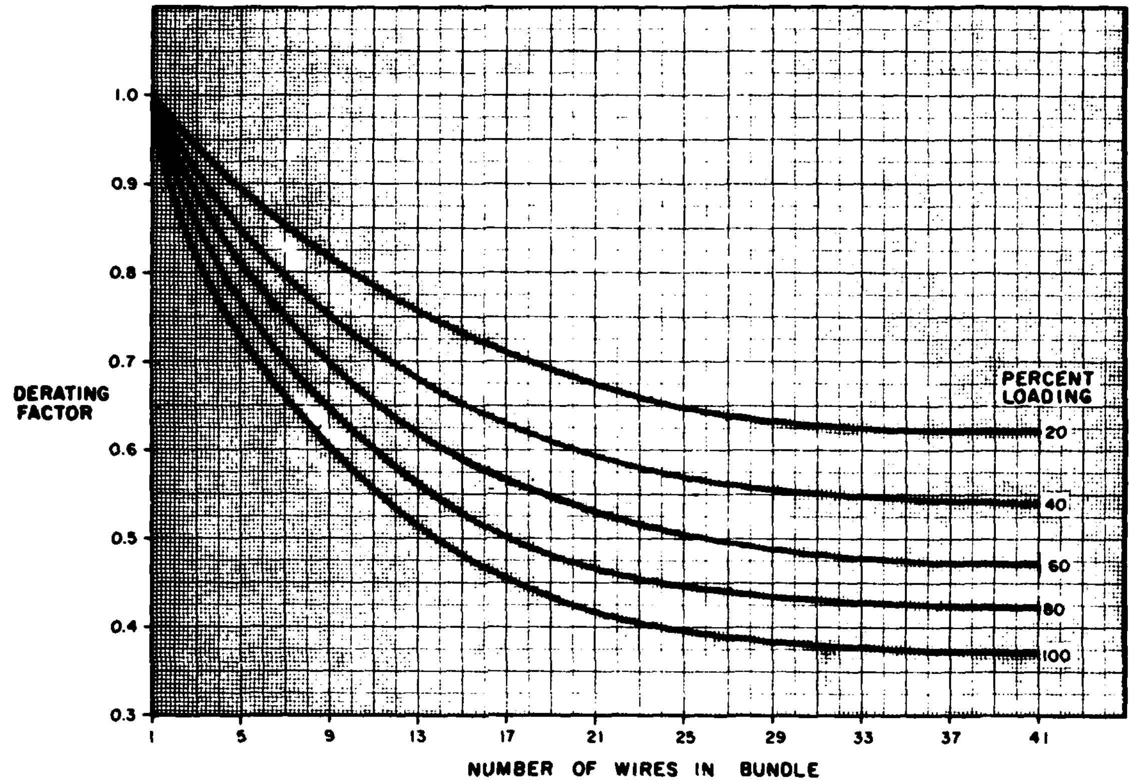


Figure 3. Current derating curves for wire bundles.

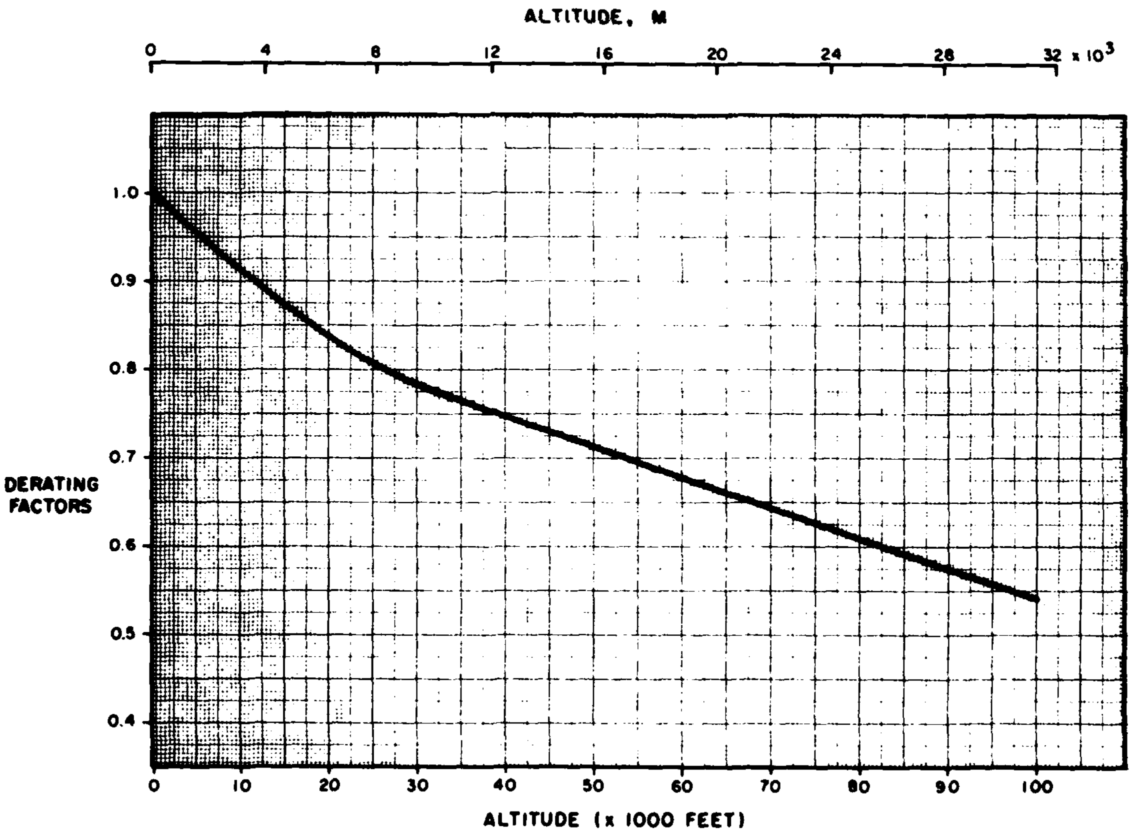


Figure 4. Current derating curve for altitude.

TEST CURRENT

Maximum current ratings of contacts and maximum allowable voltage drop under test conditions when assembled as in service are shown below. Maximum total current to be carried per connector is the same as that allowable in wire bundles as specified in MIL-W-5088.

CURRENT RATING (*MIL-C-5015G test method*)

Contact Size	Max. Current Rating (amps)	Test Current (amps)	Potential Drop (millivolts)
16	22	13	50
12	41	23	50
8	73	46	25
4	135	80	14
0	245	150	12

Figure 5. Example of connector contact current ratings referencing MIL-W-5088 (Ref. 5).

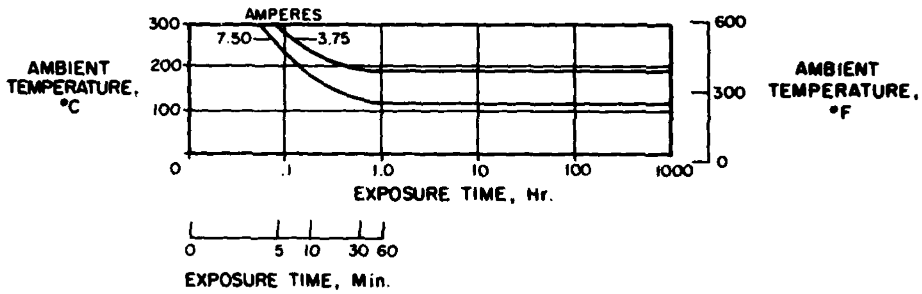


Figure 6. Exposure time and temperature limitations for a 12 pin connector based on a contact temperature of 239° C (463° F) (Ref. 7).

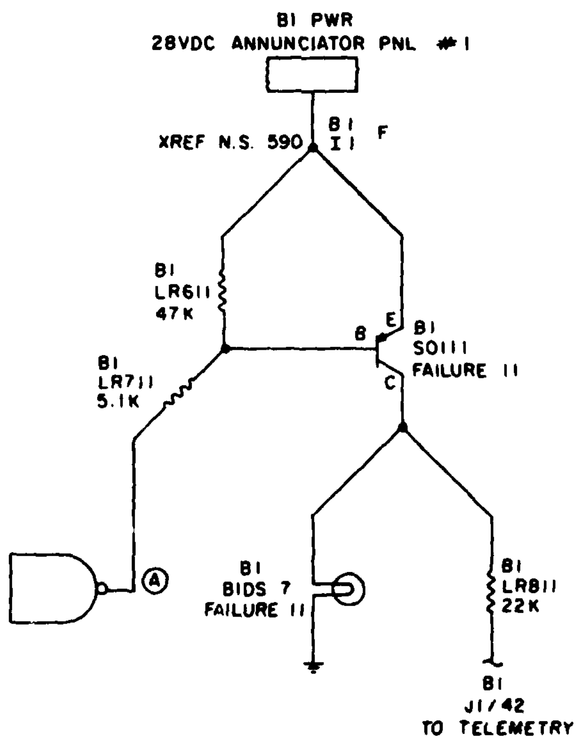


Figure 7. Sneak circuit network tree.

DATE	3-18-74
ENGINEER	C. Crawford C. Crawford
Dk 3-18-74	

TITLE POSSIBLE POWER TO GROUND SHORT IN ENGAGE CIRCUITRY

REFERENCES

1. AG-40-010, Basic Revision, Autopilot Engage and "A" - "B" Select
2. AF-61-00, Basic Revision, Autopilot Engage Interlock
3. 737 RSFS Nodal Set 102

MODULE/EQUIPMENT

AFD/B237

EXPLANATION

When the FFD "CWS" Engage Switch relay K1 is energized, a switch is enabled which changes the contact positions of K1 and opens a microswitch which inhibits the K8 circuitry. At this time the aileron force limiter solenoid is energized and its contacts change state. If the emergency disconnect switch in B237 is opened, the aileron force limiter solenoid loses power and its contacts return to a normally closed state. If S149 is thrown to its normally open position, a power to ground path exists as shown by the sneak path arrows in Figure 1 (attached). This path exists only momentarily unless the spring loaded switch enabled by K1 is physically held in the engaged position. A similar path exists through the suppression diode of K8 also.

POTENTIAL IMPACT

Damage to circuitry

RECOMMENDATION

Place a blocking diode below pin 1 of S149.

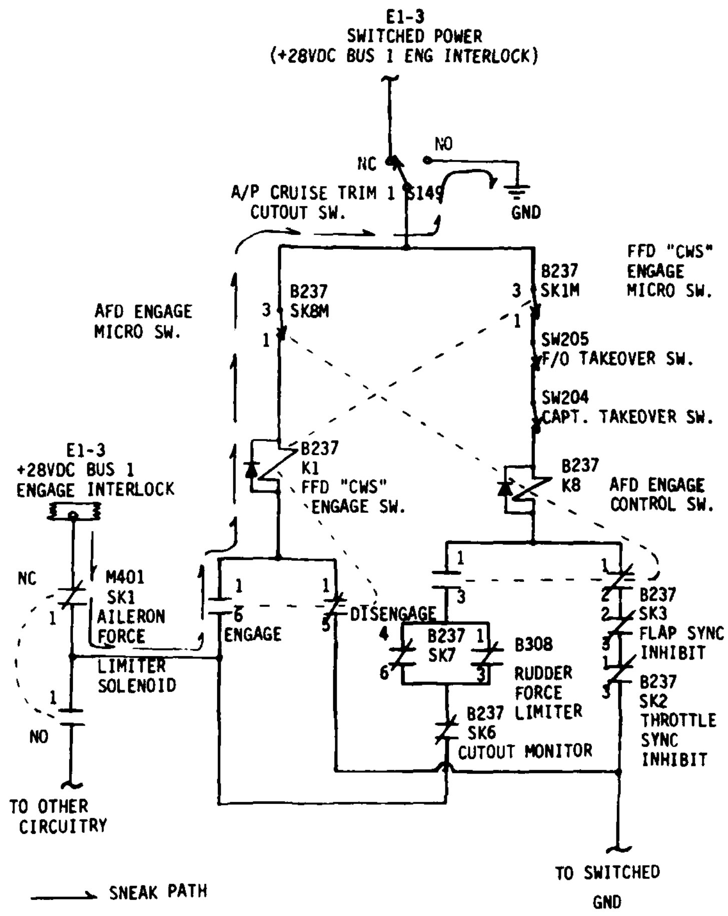


Figure 1 - AFD AND FFD ENGAGE CONTROL CIRCUITRY

Figure 8. Sneak path. (Sneak circuit report 737 RSFS 16, Ref. 9.)

TITLE Altitude Hold Switch May Drop Out After Engagement

DATE	8/19/74
ENGINEER	Gordon Buckley Gordon B. Buckley

REFERENCES

1) G.E. Drawing 925C292, Revision C, 1-26-66

2) G.E. Drawing 281E402, Revision B, 2-5-74

MODULE/EQUIPMENT

Control Amplifier Part #230E420G4, Ref. Des. 65-AR305

EXPLANATION

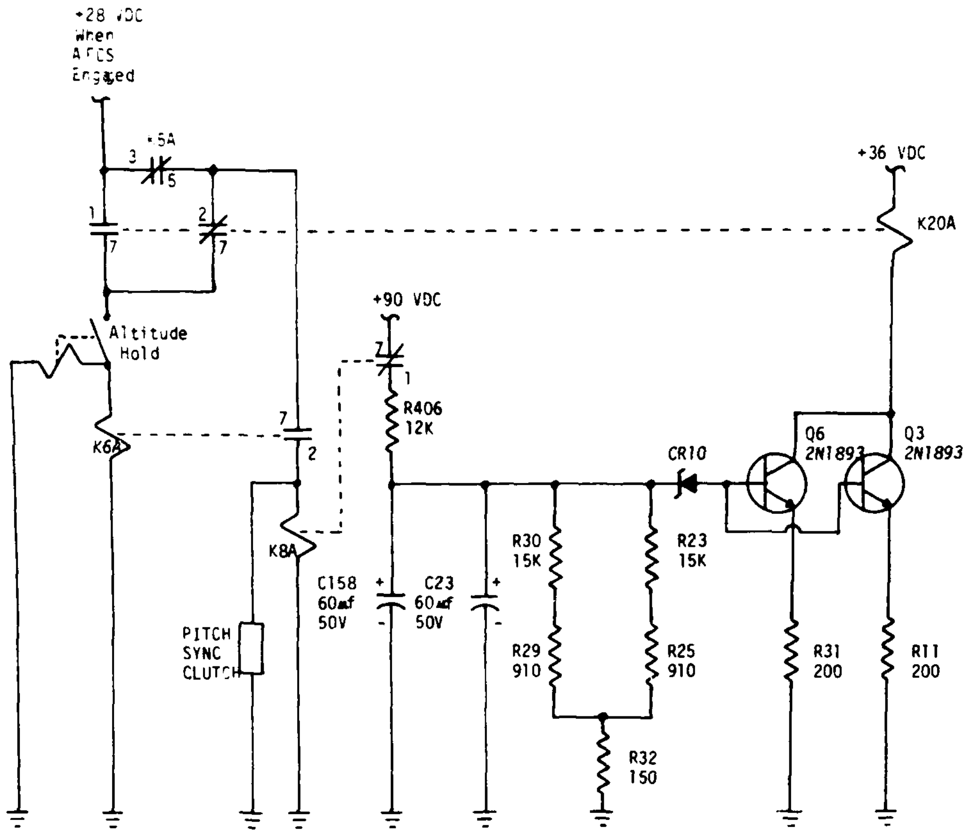
When Altitude Hold Switch is engaged, K6A energizes and closes contact Pins 2 and 7. Then K8A energizes and opens contact Pins 1 and 7, thus removing the 90 VDC from the Adder Attenuator. After C158 and C23 discharge below approximately 21 volts transistor Q3 and Q6 turn off and K20A loses its path to ground and de-energizes. When K20A de-energizes K20A contact Pins 1 and 7 close and K20A contact Pins 2 and 7 open. When these contacts are switching, the Altitude Hold Engage switch loses voltage to its holding coil.

POTENTIAL IMPACT

Altitude Hold may disengage while K20A contacts are switching.

RECOMMENDATION

Replace relay K20A with a relay with make before break contacts.



Contacts shown in normal position during AFCS operation prior to Altitude Hold Engagement

Figure 9. Sneak timing. (Based on sneak circuit report F-4C-4, Ref. 9.)

TITLE False Indication Of "I&A Relay Reset" Status

DATE 2-19-75

ENGINEER J. B. Campbell

REFERENCES

1. 381093 - Schematic Diagram, PAL/ACS Spacer Sheet 2, Zone 32, 33

MODULE/EQUIPMENT

PAL/A4

EXPLANATION

The indication that I&A relays are reset, as shown in Figure 1, is a false indication. Only relays K5 and K6 are checked by this indication.

POTENTIAL IMPACT

If Relays K3 and K4 fail in the set state and a false "I&A Relays Reset Indication" would be monitored.

RECOMMENDATION

Add relay contacts for K3 and K4 to circuit or change title to "K5 and K6 I&A Rly Reset".

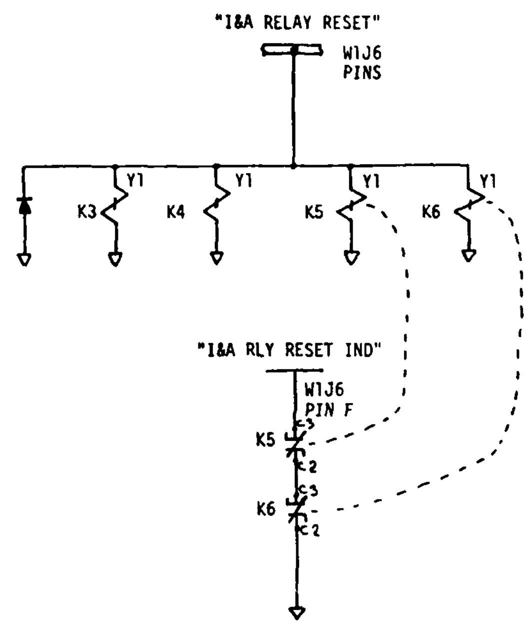


FIGURE 1. I&A RELAY RESET CIRCUITRY

Figure 10. Sneak indicator. (Sneak circuit report SSTTP-2, Ref. 9.)

TITLE

MISLEADING "LANDING LIGHT ON"
TELEMETRY MEASUREMENT

DATE February 22, 1973

ENGINEER E. Bryan

REFERENCES

1) Electrical and Electronic Wiring Diagram, 225-18005, Rev. G, Page 33-40
2) Multiplexer Measurements List provided by Organization 2-2575

MODULE/EQUIPMENT

Air Vehicle/Exterior Lights

EXPLANATION

The "Landing Lights On" measurements shown in Figure 1 are not true under certain conditions. It is possible for the measurements to be on and the LH Landing Lights OFF if CB17 is open. Also, it is possible for the measurements to be OFF and the LH Landing Lights ON if CB18 is open. The measurements shown in Figure 1 actually monitor the RH Landing Light status.

RECOMMENDATION

Rename existing measurements "RH Landing Lights ON"; and add new measurements for "LH Landing Lights On".

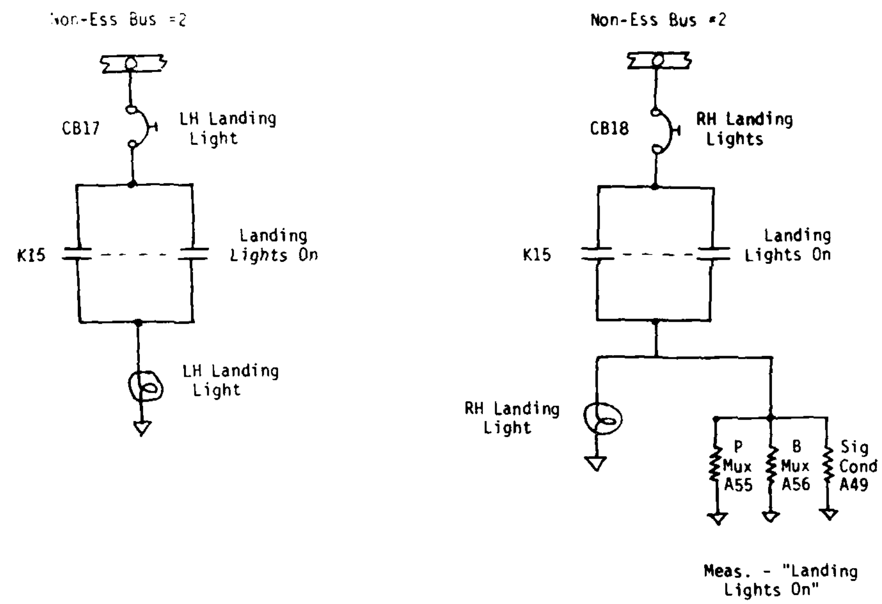


Figure 1. LANDING LIGHTS POWER AND CONTROL CIRCUITRY

Figure 11. Sneak label. (Sneak circuit report 1148-7, Ref. 9.)

TITLE Lost Carrier Check In-effective in Resetting Recovery System

DATE 12/2/74

ENGINEER Gordon Buckley

GB Gordon B. Buckley

REFERENCES

1) T.O. 147SD-1, Page 2-86, Flight Controller Manual

MODULE/EQUIPMENT

AQM-34M/Chute Timer

EXPLANATION

If the Lost Carrier Check is performed between seven minutes and two minutes prior to SPA Launch, the following condition exists.

In Step 2 of the Lost Carrier Check, the Chute Timer Reset switch is depressed and the Drag and Main Primer lights go off. In Step 3 t.e Carrier Test switch is switched to the Carrier Test position and its light comes on. If Step 3 is not initiated within 30 seconds of Step 2, Drag and Main Primer Timing Circuits (Chute Timer) will provide a continuous output which will result in the chutes being released as soon as the SPA is released from the launch aircraft.

RECOMMENDATION

Step 2 and Step 3 should be interchanged.

Figure 12. Sneak procedure (Ref. 9).

DOCUMENT NO.	REFERENCE DESIGNATOR	SUBSYSTEM
220601B	See Below	Digital

UNIT NOMENCLATURE

Global Signal List

DISCREPANCY:

Listed below are signals and associated connectors which have no apparent "to" termination point. These wires are not labeled as spare or in any other way designated as unused.

Signal Name	Connector-Pin	Signal Name	Connector-Pin
+15MOND	AJ09-129	+28VDH	E000-051
+15TESTD	AJ09-153	+28VDL	E000-052
+15VD	E000-046	+5MON1D	AJ09-118
+28MOND	AJ09-140	+5MON2D	AJ09-119
+28TESTD	AJ09-155	+5TEST1D	AJ09-151
+28VDCTA	J017-093	+5TEST2D	AJ09-152
+28VDCTB	J117-093	-15MOND	AJ09-130
+28VDCTC	J217-093	-15TESTD	AJ09-154

(Continued)

ASSUMED CORRECTION:

For the purposes of Sneak Circuit Analysis these wires are assumed to be open-ended. This wire list should be revised to reflect usage of all wiring.

REPORTED BY B. Alexander DATE 1-8-76

SNEAK CIRCUIT GROUP ACTION BY ______ DATE ______

CONTACT NAME ______ DATE ______

CONTRACTOR CONCURRENCE BY ______ DATE ______

Figure 13. Drawing error. (Sneak circuit drawing error report FBW-15, Ref. 10.)

TITLE Capacitor Failure Due to Diode Failure

DATE	7/30/74
ENGINEER	P. F. Stokes

REFERENCES

1) G.E. Drawing 281E402, Revision B, Dated 2-5-74.
2) G.E. Drawing 925C292, Revision C, Dated 1-26-66.
3) G.E. Drawing 702C190, Revision C, Dated 3-31-65
4) Illustrated Parts Breakdown, T.O. 5A3-50-4, Change 1, Dated 1 April 1973
5) MIL-STD-198C, Dated 29 September 1972.

MODULE/EQUIPMENT

Control Amplifier, P/N 230E420G4, Ref. Des. 65-AR305

EXPLANATION

Diodes CR50, CR51 and CR52 rectify the incoming 115VAC, 400 Hz, 3 phase power to provide 135 VDC to the Adder Attenuator and Electronic Switch Monitor. This voltage is filtered by capacitors C50, C51, C158 and C23. C23 is located in the Adder Attenuator Module. The remaining capacitors are chassis mounted components. Should any of the diodes fail in the shorted mode (then fail open) each of the four capacitors will be subjected to reverse current during the negative portion of the cycle for the phase containing the shorted diode. Per reference 4 these capacitors are of the Tantalum Sintered-Slug type. Per MIL-STD-198C, Section 802, paragraph 2.1.2 these capacitors cannot withstand any reverse voltage.

POTENTIAL IMPACT

Capacitors may fail immediately or be damaged to the extent that they will fail later - possibly in flight. Capacitor C51 failure will prevent AFCS from being engaged. Capacitor C158 or C23 failure results in the loss of the pitch fader effect. Capacitor C50 failure results in the loss of both pitch and roll fader effects.

RECOMMENDATION

Add one additional diode in series with CR50, CR51 and CR52 as shown in the attached figure.

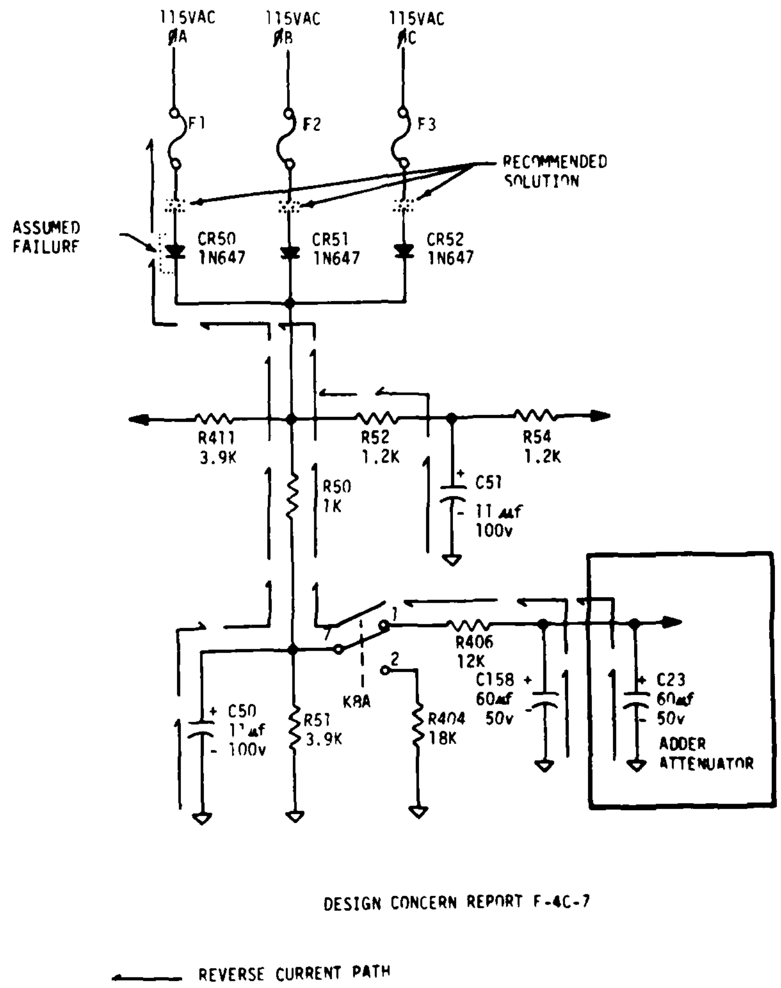


Figure 14. Sneak design concern. (Design concern report F-4C-7, Ref. 9.)

SUMMARY OF SESSION IV - INSTRUMENTATION TECHNIQUES

The trends indicated in instrumentation are toward the use of distributed digital systems and microcomputers and therefore to the installation of digital data busses. These trends have certainly been stimulated by the availability of the small, power-efficient, rugged and inexpensive digital components but they have also been necessitated by demands for increased data rates - to measure increased numbers of parameters with higher frequency content. Other topics which receive much attention are the installation time and flexibility of the installed system and real-time data analysis or at least shorter turn-around times.

Examples were given of multiplexed data busses in an advanced fighter configuration, a programmable multi-purpose instrumentation system, and the impact of integrated circuits (mini and microcomputers) on, for example, on-board real-time data reduction. An airborne system for measuring takeoff and landing performance, almost fully automatically, using an inertial sensing system was discussed and a multi-transponder microwave system for real-time position measurement of an aircraft was described. Both of these latter systems have been developed to streamline portions of the data reduction process during civil aircraft certification flight testing; however, each will require validation trials before it will be accepted by the certifying agencies.

Finally, the importance of the human element has not been eliminated by these more "intelligent" instrumentation systems. Much of the drudgery can be removed from the flight test analyst's job but he must still make intelligent use of the data.

(During this Session, the FMP Chairman M. Jean Renaudie presented Flight Mechanics Panel AGARD Plaques to Mr. A. Pool of the Netherlands National Aerospace Laboratory, and Mr. Kenneth C. Sanderson of the United States and formerly with the NASA-Dryden Research Center, in appreciation of their work on AGARDograph 160 Flight Test Instrumentation Series.

Mr. Ken Sanderson has served as an editor for the Flight Test Instrumentation Working Group since Spring 1975. He has been an extremely diligent member of this Working Group and his services will be sorely missed. Mr. Sanderson retired from NASA in June 1980 and will retire from the Working Group in April 1981.

Mr. Tony Pool has served as the European Editor for the Working Group since its inception in 1968. He was also a contributor to Volume IV "Instrumentation Systems" of the AGARD Flight Test Manual which was published in the mid-1950s. Mr. Pool has agreed to continue as an editor for the Flight Test Techniques Series which is just getting started.

Messrs. Sanderson and Pool are true AGARDians. Their unselfish devotion and many hours of night and weekend work are sincerely appreciated.)

SUMMATION BY THE TECHNICAL PROGRAMME COMMITTEE

The overall aim of the symposium was to review the current status of aircraft sub-system flight testing technology. Some general thoughts which have been provoked by the papers and discussion in relation to this aim are expressed in this final summary.

In the preface it was noted that one specific objective of the symposium was to illustrate the diversity of problem and solution in the field of sub-system testing. Most certainly this has been achieved; with presentations on topics ranging from micro-miniaturised avionics to the 'heavy' engineering of undercarriages and guns; from the impact of climatic extremes to the elusive effects of electro-magnetic radiation; from test techniques and instrumentation facilities of sophisticated complexity to those of simple elegance; and from aircraft to helicopters.

Embracing this variety, the four main subject areas of the symposium may be recalled -

- Navigation/Attack Systems Testing
- Aircraft Systems Testing
- Environmental Testing
- Instrumentation Techniques

The first session revealed that, from a pure test standpoint, navigation and attack system testing procedures are well defined and are aided by modern measurement and recording equipment. There is a need to plan, develop, and utilize ground-based simulations prior to and during flight testing to reduce test flying, to evaluate failure modes and redundant systems, and to validate changes prior to flight. For both types of systems, the need for a "global" view of testing emerged, where major components are treated as parts of a total system rather than as isolated sub-systems.

In the second session, test and instrumentation techniques for evaluating various, unrelated sub-systems were presented. These techniques emphasized the need to evaluate carefully the intended use of the sub-system, then prepare a test technique and a measurement technique. The need to consider the electromagnetic environment of the sub-system and aircraft was noted and a strong point was made that electromagnetic interference potentials must be recognized and corrected during the aircraft design stage. Methodologies to accomplish reliability and maintainability evaluations during initial testing of an aircraft were presented. Techniques to check early reliability and maintainability and to correct defects are available, but fixes are usually expensive.

Session III disclosed that environmental testing has achieved considerable maturity and is now normally regarded as an essential part of an aircraft development program. A number of environmental test capabilities were described as were their instrumentation systems. Environmental qualification is an incremental process starting with design support testing, then full scale simulated environmental testing, and finally, natural environment testing.

The final session showed that trends in instrumentation are toward the use of distributed digital systems utilizing micro-computers and digital data buses. The trend has been stimulated by the development and availability of small, power efficient, rugged, but inexpensive digital components. Examples were given of airborne systems that permit on-board real-time data reduction.

Despite the varied subject matter, one common theme emerged clearly from all sessions. It concerns a maturing attitude to sub-system development. There was a time when the flight proving of sub-systems was commonly regarded as almost incidental to 'mainstream' aircraft development in areas such as flying qualities, structural integrity and mission performance. Sub-system testing was often merely tolerated on a ride-along basis, to be tackled in depth only when a particular system gave trouble. By contrast, the presentations of this symposium have repeatedly demonstrated that sub-system development is now approached in far more positive fashion, and with deeper engineering concern. Four factors would appear to have contributed to this -

- Systems themselves are becoming more complex, and more dependent one upon the other. They are far less amenable than hitherto to the "we'll put it right when it goes wrong" approach.

- The facilities for measuring, recording and processing data now provided by the instrumentation community are already extremely powerful, and their power is growing steadily.

- Modern computation has dramatically increased predictive and analytic capability, particularly through mathematical models and simulation.

- There is now greater willingness, partially inspired by the cost of flight trials, to invest in sophisticated rigs for ground testing.

Whilst none is entirely new, these factors now combine in supporting a widespread initiative to meld all available contributions from analysis, prediction and ground test with elegantly instrumented flight trials in the forceful pursuit of more effective system developments; to consider from the outset the requirements and functioning of a system, and to devise a coherent and carefully constructed development process for it: in essence, to bring to sub-system development the level of engineering thinking which has previously been largely reserved to overall aircraft concepts. This trend can only be beneficial and must be encouraged.

A further general observation on the symposium concerns the preponderance of papers with a strong avionic content. Once again this emphasises the dramatic growth in aircraft electronics, and adds force to the conviction, expressed in discussion, that whilst air-vehicle trials will always remain, the future flight testing scene will be dominated by avionics. Directly associated with this development is the fact that avionic systems are particularly amenable to integration of instrumentation and the system itself, especially through the medium of the data-bus. Instances were quoted where up to 50% of the total data recorded in flight trials were already being acquired in this way. The interactive situation so produced lends further need to the carefully considered development processes admired earlier.

In final conclusion, the Technical Programme Committee were encouraged to believe that the aims of the symposium were substantially met. This was achieved through the considerable efforts of the authors and session chairmen, and the active participation of the attendees, all of whom we would thank. As evidenced here, rapid changes are occurring in the techniques of flight testing and further symposia in this field should be encouraged.

J. F. O'GARA
F. N. STOLIKER
Members, Flight Mechanics
Panel

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