SOCIETY OF FLIGHT TEST ENGINEERS

14th ANNUAL SYMPOSIUM PROCEEDINGS

"FLIGHT TESTING TODAY: INNOVATIVE MANAGEMENT AND TECHNOLOGY"

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FOREWORD

Welcome to Orange County and the 14th Annual International Symposium of the Society of Flight Test Engineers. The goal of the Los Angeles Chapter is to provide professional exchange in an enjoyable Southern California environment. The theme this year, "FLIGHT TESTING TODAY. . . Innovative Management and Technology," reflects today's emphasis on productivity, cost reduction, and the demands on the flight test engineer for innovative management and technical methods.

The Los Angeles Chapter would like to thank all the authors who have contributed technical papers and the vendors for their participation with displays of the latest in state-of-the-art hardware and services.

Our sincere thanks go to the Douglas Aircraft Company, McDonnell Douglas Corporation, for its assistance and excellent support of the Symposium activities.

The goal of this Symposium could not have been accomplished without the efforts and professional dedication of the following Symposium Committee Chairmen:

Technical Papers .... John Cook
Facilities .... Steve Riter
Registration .... Bob Hanwell
Banquet/Luncheon .... Jim Agar
Reception/Tour .... Dave Houle
Publicity .... Dick Emmerson
Financial .... Gary Palmer
Vendor .... Ed Hoffmann
Corporate Support .... Dick Capiola

Spouse Activity Committee
Jan Palmer
Flo Agar
Christine Hanwell

Ginny Clare
General Symposium Chairman

KELLY JOHNSON AWARD

This year the annual "Kelly Johnson" award for outstanding achievement in the Flight Test Engineering discipline is awarded to Mr. Frederick N. Stoliker. Fred recently retired as Technical Director of the Air Force Flight Test Center and is currently employed by the Computer Sciences Corporation (CSC) to manage its support of various Defense Department flight test activities.

Fred has devoted his entire professional life to the flight test and evaluation of military aerospace systems. He has been personally involved in the development of every Air Force airplane and weapon system from the first-generation jets of the 1940s to the super-sophisticated aerodynamic and electronic marvels of the 1980s. Specifically, the award is in recognition of his outstanding and continuing effort to improve the quality of flight test support activities. Both the Integrated Flight Data Analysis Process System (IFDAPS) and the Integrated Facilities for Avionics System Testing (IFAST) owe their very existence to the drive and personal attention of Mr. Stoliker. These two facilities alone represent a $60-million Air Force commitment toward improved support of flight testing. Over the years, Fred has been personally responsible for keeping the AFFTC at the forefront of flight testing and his vigorous activities toward the development of IFDAPS and IFAST will ensure that the Center will continue to play a major role in future flight test developments.

GUEST OF HONOR/BANQUET SPEAKER
MR. RAY E. BATES
VICE PRESIDENT — GENERAL MANAGER
NEW COMMERCIAL PROGRAMS
DOUGLAS AIRCRAFT COMPANY
McDONNELL DOUGLAS CORPORATION

Mr. Bates has held his present position since January 1981. Prior to this appointment, Mr. Bates was Vice President — Advanced Programs in the Engineering department of Douglas since 1976 where he supervised research, engineering and development efforts on present and future aircraft. From April 1973 until December 1976, Mr. Bates served as Vice President — Engineering Design and Development, directing design activity on all programs within the Douglas Company. From October 1971 until April 1973, he served as Director — Engineering Design and Development. He has worked in the field of aircraft design for Douglas Aircraft Company for most of the years since he graduated from Notre Dame in 1946. He has previously served as Director of DC-10 Engineering, DC-9 Program Manager, DC-9 Chief Project Engineer and prior to that, Chief of Transport Advanced Design.

KEYNOTE SPEAKER
MAJOR GENERAL PETER W. ODGERS

Major General Odgers is Commander of the Air Force Flight Test Center at Edwards Air Force Base, California. He received a bachelor of science degree from the U.S. Naval Academy, Annapolis, and a master's degree in systems management from Southern Methodist University. He is also a graduate of Squadron Officer School and the Air War College, both at Maxwell Air Force Base, Alabama.

After receiving his commission and pilot wings in 1956, his first assignment was with the 20th Air Transport Squadron at Dover Air Force Base, Delaware (1956). Subsequent assignments included Wright-Patterson Air Force Base, Air Force Experimental Test Pilot School (1961), followed by 5 years of flight testing reconnaissance aircraft at Ling Temco Vought and General Dynamics.

Following a 1-year tour of duty at Tan Son Nhut Air Base, Vietnam, where he flew RB-57s, General Odgers was assigned again to Wright-Patterson Air Force Base (1968) working various assignments leading to deputy director of the A-X program. In 1974 he became deputy director of the F-15 System Program Office and in 1977 he became commander of the 4950th Test Wing at WPAFB. From 1979 to 1982 General Odgers was Deputy Chief of Staff, Test and Evaluation, Headquarters Air Force Systems Command, Andrews Air Force Base, Maryland.

A command pilot with more than 7,000 flying hours and many distinguished decorations, he was promoted to the rank of Major General August 1, 1982, and he assumed command of the Air Force Flight Test Center September 20, 1982, which is his current position.

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SESSION 3: TEST RESULTS
Virginia A. Clare — Session Chairman

SESSION 4: TEST METHODS
Stephen G. Riter — Session Chairman

TRAINING AND DEVELOPMENT OF ENGINEERS
AT THE AIR FORCE FLIGHT TEST CENTER -
AN OVERVIEW

By
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Air Force Flight Test Center, Edwards Air Force Base, California

Abstract

Training and development of engineers is a major undertaking for the 6520 Test Group at the Air Force Flight Test Center. Guidance and policy regarding training is provided in the Master Training Plan. The plan evolved as a result of some training and development deficiencies within the organization. This paper comments on means for identifying training deficiencies and discusses changes made to improve training and development of engineers at the Flight Test Center. The paper also briefly addresses such related items as why training is needed, assessing training needs, and preventing obsolescence.

Background

The 6520th Test Group of the Air Force Flight Test Center, (AFFTC), Edwards Air Force Base, CA, is an organization of over 600 people. The Test Group provides for the technical and administrative management of test engineering personnel charged with planning, coordination, executing, and analyzing a range of weapon system flight tests in a variety of disciplines. With some exceptions, all personnel (military and civilian) in the Test Group can be classified as professional managers, engineers, mathematical scientists, or technicians. Considering the high level of competence and professionalism required to accomplish this highly complex task, training and development is a major undertaking for the 6520th Test Group.

In most cases, new personnel do not inherently possess all the skills and technical knowledge required to be fully qualified in their career field. Journeymen level personnel require additional training and education to maintain proficiency and prevent obsolescence. Senior technical personnel need training to transition into management positions. Professional development is a continuous and systematic process of education, training and growth. Training and development must contribute to the organizational goals, and due to the high cost of training (in both time and money), must be both effective and efficient.

Certain problems with respect to training and development had previously been identified in the 6520th Test Group and these include a lack of concise guidelines for identifying, forecasting, and evaluating training needs for engineers, as well as lack of a list of training courses available for both engineers and supervisors for planning purposes. Additional problems included no single source of documentation of management policy, goals, and guidance for both engineers and supervisors with respect to training and development.

*Aerospace Engineer

Many of these deficiencies were identified and subsequently addressed as follows:

1. A survey was conducted of approximately seventy-five percent of the first level supervisors with respect to training. The purpose of this survey was to assess training needs, identify deficiencies, and provide inputs to the Master Training Plan.

2. A Master Training Plan was written and published. The plan provides guidance with respect to training and development, as well as management policy, goals and objectives.

3. The part time positions of training monitor, and training coordinators were established and manned. One coordinator, usually an engineer or supervisor, represents his or her respective Engineering Division in matters relating to training. Coordinators report to the Monitor, and meet monthly to discuss training and development.

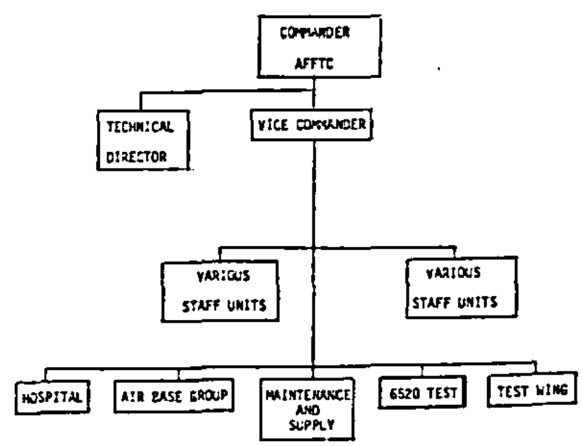

Figure 1 Air Force Flight Test Center Organizational Chart (Simplified)

Organizational Make-up

The AFFTC organizational chart is shown in figure 1. As previously indicated, the 6520th Test Group is composed primarily of engineers, scientists, and mathematicians. Within the Test Group are approximately 30 first-level supervisors whose responsibilities include being cognizant of the training needs of subordinates and the organization. As shown in figure 2, the Test Group consists of seven divisions responsible for functions typically found in a flight test organization, such as instrumentation, flight dynamics, avionics, computer processing, etc.

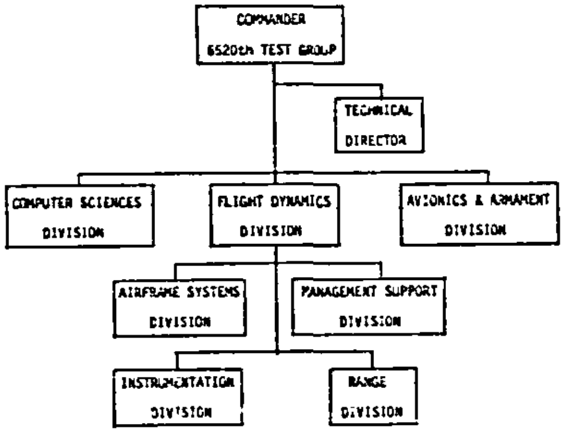

Figure 2 6520th Test Group Organizational Chart

One of the recent organizational developments initiated is that a training monitor and training coordinators have been designated within the Test Group to coordinate training and development, provide assistance, and serve as a focal point for training and development. One coordinator from each of the seven Engineering Divisions reports to the training monitor. An organizational chart is shown in figure 3.

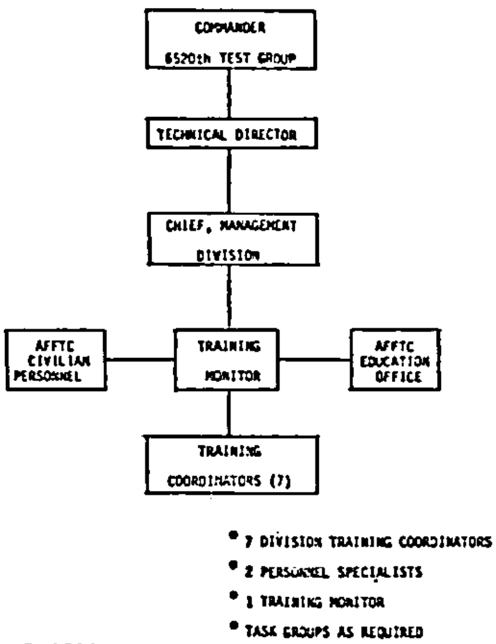

Figure 3 6520th Test Group Training and Development Structure

The duties and responsibilities of the training coordinators include:

1. Work with the Test Group monitor to administer the Group's training program,

2. Advise his or her Division Chief on training and development matters,

3. Act as a focal point for Division training and development business,

4. Act as a liaison between the Division and the training monitor.

The training monitor has responsibility to:

1. Advise the Test Group Commander on training and development matters,

2. Act as a focal point for the Division training coordinators,

3. Provide liaison between the Test Group and the education office,

4. Coordinate training forecasts, training budgets, and training requests with the appropriate organizations.

The coordinators and the monitor meet monthly to discuss training matters. The concept of training coordinators and monitor has proved very successful in streamlining the training related administrative matters.

Why Train

The overall objective of training and development is to increase organizational effectiveness. This includes the following specific objectives:

1. Productivity. Training and development helps employees increase their level of performance which usually leads to increased productivity.

2. Quality. Proper training and development increases the quantity of output as well as improves the quality.

3. Human Resources Planning. Good training and development programs help the organization to fulfill its future personnel needs and requirements.

4. Morale. The organizational climate and atmosphere are usually improved when proper training programs exist.

5. Obsolescence Prevention. Continuous employee training and development efforts are needed to keep abreast of current improvements in technology.

6. Managerial Goals. The most important resource of the Air Force Flight Test Center organization is its human resources. Thus, one of the most critical leadership tasks of management is the training and development of people who will best help the organization meet its goals.

Obsolescence

When an individual lacks new knowledge or skills, obsolescence exists. There are many definitions of obsolescence. In most definitions there [illegible] to maintain performance. There are various degrees of obsolescence. The degree that can be coped with most easily, the individual has become obsolete and knows it. Next, in degree is the individual who is unaware of his or her obsolescence. In these two forms, when obsolescence does become recognized, training is a possible solution. The most difficult form is the person who knows he or she is obsolete at the gut level, but denies it at the conscious level and refuses help.

Among organizational professionals, the problem of obsolescence appears to be most pronounced among engineers and scientists. This problem is expected to become even more pronounced in the future. Managerial obsolescence is currently or potentially a problem in many organizations. Managers in engineering appear to be the most threatened by obsolescence. Managers of technical professionals have an especially acute problem because they are subject to obsolescence not only in management methods but in their technical specialties as well.

Preventing Obsolescence

What can be done about obsolescence? Some managers recommend rather drastic action: reassign, terminate, or give the individual a featherbed job. Obviously it is more desirable to prevent obsolescence rather than try to treat it after it exists. The Air Force Flight Test Center, like many other high technology organizations, believes the answer to the problems of both existing obsolescence and preventing obsolescence lies in training and development. Some of the elements of the anti-obsolescence program at the Test Center include:

1. On-the-job training, including job coaching and counseling, and job rotation.

2. Formal education during off duty hours. Currently there are five universities represented on Edwards AFB. Courses of study leading to Bachelor's and Master's degree are available after working hours.

3. Long-term, full-time (LTFT) training. This involves full time study and research at a University or at an Armed Forces College. Also included in this program is the prestigious Sloan Fellowship. Due to the resources involved, LTFT is limited to a few individuals each year.

4. Test Center sponsored development. These include in-house courses, on and off-base short courses, and video courses.

5. Professional associations. The Air Force Flight Test Center management fully support professional and technical associations. These can be effective in preventing obsolescence providing the individuals avail themselves to the opportunities presented such as publications, journals, proceedings, meetings, symposiums, etc.

Assessing Training Needs

The assessment of training needs addresses two questions: who needs training, and what training do they need? To help answer these questions, the 6520 Test Group uses one or more of the following elements:

Individual Development Plan

The individual Development Plan (IDP) is an outgrowth of the annual performance appraisal process. It is completed as mutually agreed to between the supervisor and employee and is a plan for developing employee capabilities through self-development, formal classroom training and work assignments. It may include both short-range and long-range goals, and the planned training and development to help achieve these goals.

Training Forecast

The training forecast is submitted twice yearly by each Engineering Division within the Test Group. It is used to help establish training budgets and requirements for the coming period. The forecast helps the Division Chief determine what training is needed, and who needs it.

Training Monitor/Coordinators

As previously discussed, the monitor and coordinators meet periodically with representatives from both the civilian and military training offices to discuss training needs as well as problems.

Master Training Plan

The Master Training Plan was written to provide guidance to both supervisors and non-supervisors in matters relating to training and development. The plan evolved due to a demonstrated need for a single document to state organizational policy and objectives as well as provide practical information regarding training and development. Subjects in the Master Training Plan include:

(a) A training plan for new engineers, and a training plan for junior engineers,

(b) Descriptions of various types of on and off-base training programs,

(c) A training menu listing over 200 courses of interest and available to the various Engineering Divisions,

(d) Sample completed forms to facilitate paperwork processing

A partial table of contents for the Master Training Plan is shown in Appendix A.

Newsletter

A training and development newsletter is published approximately every quarter as a current source of information to employees regarding training and development. The newsletter receives wide distribution and serves as a reminder of the importance of training and development in the 6520th Test Group. Typical items published in the newsletter include training courses completed by individuals since publication of the last newsletter, a tentative list of upcoming courses, minutes from recent training coordinators meetings, and an article from the Test Group command section regarding training and development.

Conclusion

Training and development at the Air Force Flight Test Center must be efficient and effective in the highly technical environment of weapons system testing and evaluation. The 6520th Test Group implemented several plans which were successful in improving the training and development environment and awareness. These include:

(1) Conduct a survey to help identify training deficiencies, assess training needs, and help provide inputs to the Master Training Plan.

(2) Write a Master Training Plan to document organizational goals and policies with respect to training, as well as to provide a source of information on training and development.

(3) A training monitor and a training coordinator from each Engineering Division act as a focal point for training. They meet monthly to discuss training and development.

APPENDIX A

Partial Table of Contents of the Master Training Plan

APPENDIX B

(Some training and development data for the Flight Dynamics Division).

The Flight Dynamics Division is one of seven divisions within the 6520th Test Group, Air Force Flight Test Center. The division has approximately 60 engineers, and five first-level supervisors, all reporting to the Division Chief. The Flight Dynamics Division has responsibility for testing and evaluating air vehicle performance, flying qualities, and propulsion. In addition, the Division is responsible for flight simulations.

TABLE B-1, Education Level

	Engineers	Supervisors
Bachelor's degree		
One degree	60 (100%)	5 (100%)
More than one degree	2 (3.3%)	0 (0%)
Master's degree		
One degree	19 (32%)	5 (100%)
More than one degree	1 (1.7%)	1 (20%)
Test Pilot School Graduate	4 (6.7%)	0 (0%)
Professional/EIT Registration	4 (6.7%)	0 (0%)

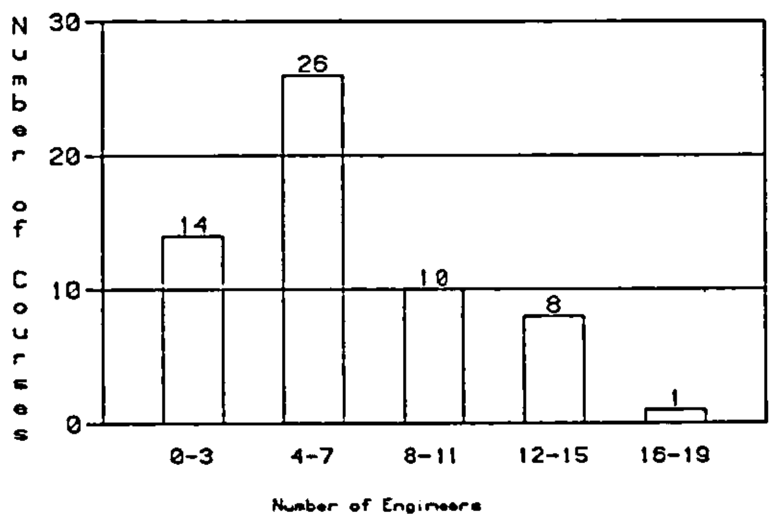

Figure A1: Distribution of Short Courses Completed

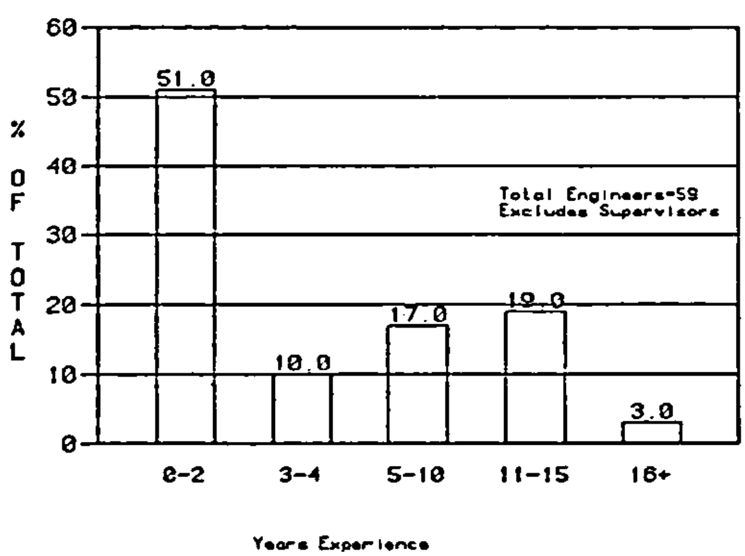

Figure A2: Flight Dynamics Division Engineers Experience Distribution

IMPACTS OF AUTOMATION
Automation and Flight Test Engineering

by
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Abstract.

Some consequences of computerization on day to day practice and on managing aspects of experimental work are discussed.
Attention is given to the position of flight test engineering within the total picture of experimental work and the influence of automation on this relation.

Introduction.

Automation in several forms, like data collection, data logging, voice recording, etc., has been a tool in flight test engineering for a long time, however due to the arrival of the computer the level of automation has taken such a different proportion that it can be useful to look at the impacts.
The electronic approach of automation was started by the computer in its rôle which is best characterized by its french name "ordinateur", the rôle of ordening and adapting information to help decision making.

The direct advantages are:
- the ability to absorb greater changes in workload,
- the transportion of streams of data,
- on-board computation.

Indirectly the management aspect of the test work itself and its interface with design and with the production of user information is influenced by automation . This influence will be discussed under the heading: "Automation and flight test management".

Under the heading: "Prerequisites of automation" attention will be given to:
- the importance of programming procedures
- a low vulnerability of the system itself and its infrastructure.

The impact of automation on the task of flight test engineering is evident. It bears similarity with the way in which automation has influenced and will influence cockpit crew tasks and procedures.

In implementing automation within the flight test engineering task it will be necessary to maintain an integration strategy and to be aware of the danger of uncoordinated installation.

Technical consequences of automation.

Although the development of "CAD" (Computer Aided Design) tends to diminish the length of some of the typical routine flight test programmes, as will be discussed later in this section, peak loads in the flight test work will happen in the future as they did in the past.

Variations in workload.

The resources and the personnel must be available to manage these peak loads. By applying automation for operating the flight test instrumentation, sufficient flexibility can be obtained to vary instrumentation set-ups during flight as required by the tests and to minimize instrumentation checkout time between flights.

Reconfiguring of the instrumentation selection during flight and the associated administrative data logging should be a standardized routine among all running programmes, a purpose for which computer intelligence is the proper means. Development and release for use of this intelligence in hardware and software can to a certain extent fill the gaps between periods of high occupation with flight work. The effect is a more stable base load of work for flight test personnel and better capacity to cope with the peak loads. For rapid installation and integration the flight test instrumentation should be of modular design and commonality principles should be applied.

It was mentioned above that application of CAD tends to diminish the extent of certain flight test programmes with a typical routine character. The ever increasing possibilities of the computer result in a shift of experimental work from real world (flight) test to simulation: mathematical model, scale model (wind tunnel) and representative system (flight simulator).

A calculation for a transonic wingflow condition can be done by the present generation of supercomputers in a couple of minutes. Supported by wind tunnel research and verification, computer aided wing design has reached a remarkable perfection (ref. 1).

The same applies to the possibilities of simulating automatic approaches and landings, including wind shear and turbulence cases, by representative systems. Before the flight tests have started, the automatic flight control system can already be well understood with a high confidence level. The purpose of flight test can then be restricted to validation of the simulation tests. Especially digitally controlled systems will benefit from such an approach.

Generally speaking the character of flight testing changes as a result of the possibilities of automation from a development tool to a verifying tool. This change in character has consequences on test management, which will be commented upon later.

Data transportation.

Especially for noise and performance measurements in take-off and landing but sometimes also for other tests the necessity can exist to perform the tests away from the home base.

Because the test airplane needs and produces data there will be a requirement for transportation of data.

Data transportation may also be required between computers, for instance when one or more dedicated computers serve as first receivers of flight data and pass the data on to a data base in a central computer system. As long as only short distances have to be covered telephone connections can be used.

A change of theater of the flight tests, as it can be necessary in the cases mentioned above will ask for some special precautions to enable data transportation. Interactive access to the computer system of the home base will be easy to organise, whatever the distance to the home base is. Larger scale data transportation could require physical dispatch of tapes until satellite services become a competing possibility.

For the combination of very large distances and dense streams of information satellite communication comes in the picture. High frequency K-band facilities in the 12 - 14 ghz range seem to give prospects, as soon as they become commercially available (ref 2).

On-board computation.

For some tests real-time telemetered or on-board-computation can support the correct execution of the tests. Examples are ILS beacon line synthesis, a system which offers the opportunity to sense the characteristics of any fictitious or existing ILS system without the need of having it available. Another example is a system, in use at Fokker, which gives the pilot guidance on the normal ADI/HSI display to fly exactly a certain flight path for noise measurements (here the computer processes distances between the airplane and a number of transponders on the ground and feeds appropriate signals into the flight director system).

Quick-look monitoring supported by an on-board-computer is maturing to a powerfull instrument enabling during and between the tests to decide on acceptance or rejection of runs and in some cases on small adaptations of running procedures. When the test procedure consists of a stepwise approximation of a critical condition on-board computation can be used to support the decision, whether or not to continue with the next step. For best results the on-board-computer should be provided with the facilities for graphic display of the refresh type. Quick-look monitoring in this way allows more efficient use of the flight time.

For standard flight tests on F.28 production airplanes Fokker is using a very simple, but also very usefull device for acceptance/rejection of cruise performance runs. Instrument readings are corrected, converted and compared with Handbook data and the results are printed. The work is done by a hand calculator and a printer. By an interface loop the two can speak with each other.

Automation and flight test management.

Flight test management refers to a chain of events, which can be characterized by five phases:

1. Logistics: estimate required resources in terms of time, available

airplane(s), weather conditions, instrumentation, telemetry, data handling, ballasting facilities, etc.

2. Preparations: define procedures, predict as closely as possible the outcome of the tests, envisage safety considerations, discuss purpose, conductance and probable results with flight crew, produce flight test cards.

3. Test: execute and monitor the tests, give comments in the post-flight meeting and make a record of these comments as given by all participating parties, decide whether the test request has been covered or a repeat should follow, or alternate course of action should be taken.

4. Presentation: secure accessibility of all results by a reference system, process measured data and make them available for analysis.

5. Generalisation: have predictions confirm- ed or corrected,produce user information, submit certification evidence.

To manage this complex of tasks various methods are used.
One old recipe is to develop standard routines, use a check-list of necessary actions and apply the chosen instrument, called flight test card.
One next step is to go for text processing. Text processing is a great help for documentation, but it offers only minor advantages to the management function.

In order to:
- support decisions
- spread up-to-date status reports under all interested persons

automation can also be applied.
The information should be presented in the following forms:

- list of flight test requests;
- annunciation, whether a test is ready for execution or not;
- if not ready, reason why (preparations still to be made);
- remarks on priorities;
- contents of test cards;
- listing of test cards, with airplane, flight number and data, if completed;
- conditions for tests, for instance altitude, center of gravity position;
- total man hours and flight hours devoted to subject, as compared with authorized hours;
- reference to reports;
- test results summaries;

By logging all these data in a data base and providing easy access methods to users, both aims mentioned above can be served. Entering of the data is started at the moment that agreement has been reached on the logistic aspects of the flight test request (phase 1).
At the same time the conditions to be fulfilled by other departments for the test to become ready for execution are added under the heading of the responsible department. While the preparations by the flight test department are taking place the agreed test procedures are specified on the flight test cards. These activities, together with the record of preparations to be made by other departments, are entered in the system, which can give status reports on the actual situation (phase 2).
After each flight the test cards are completed with flight number, date of flight, references to recording number and manual notes made during the flight (phase 3). Flight number and date automatically show up in the status report, so that the progress made in execution of the program is available for consultation.
Also a record is kept of the costs (manhours and flighthours).
The results of the post-flight meeting under a number of standard headings (test result summary) are also entered and measured date are given release for analysis (phase 4).
The final phase 5 (generalization) can now be started by the engineering department.
An automated system as described here highly improves the flexibility and efficiency in managing the flight tests. It requires a certain discipline from the flight test engineer, but it keeps his actions accessible for other people; it has the advantage that his work can be taken over immediately in case he has to interrupt it for any reason and that there is a standardized language that allows easy communication.

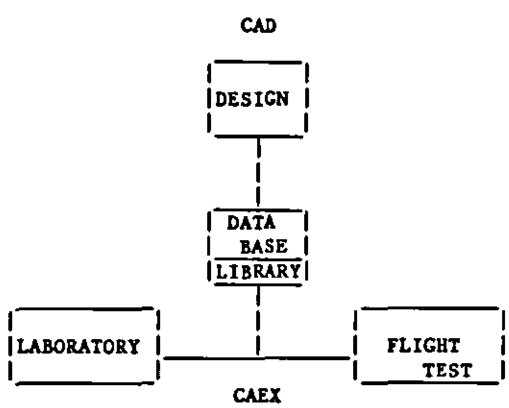
Coherence in test activities.

For some subjects the experimental work, supporting design and development, is mainly laboratory testing, while for other subjects flight testing is the primary source of information. There is however an increasing number of subjects where laboratory and flight testing are supplementing each other (ref. 3).

In that case a conceivable situation might be, that the experimental work is started in the laboratory under simulated conditions with a wide variety of inputs, the next phase being verification of the laboratory results by selected flight tests, followed if necessary by a final laboratory program based upon the results of the flight tests.

Taking into account that the laboratory tests themselves are based on a mathematical model and have been preceeded by fault analyses it is clear that the test activities and their preparations require a coordinated effort of design, laboratory and flight test in order to obtain coherence between the various activities.

Automation is a valuable means to achieve quick exchange of information, punctuality in purpose and exactness in reporting of test results in these cases. In the same way as it is gaining its place in design as CAD it is worth to be considered in experimental work as CAEX.
The following diagram illustrates the relationship



The library function does not only assist the experimental department to handle the data but does also expedite data presentation to the different departments in design. Basic data reduction activities like calculations of true airspeed, ambient temperature etc., interpolation and plotting rules, should be available in the library, while processed data, general flight data, comments given in the post-flight meeting and administrative data relevant to the measurements can be presented by the data base.

At Fokkers flight test department the development of this philosophy is reflected by some organization adjustments:

The flight test engineering function has in the past been supplemented by a "flight test documentation" function and recently by the "data acquisition systems" function. The flight test documentation function is held by a "jobkeeper informatics", a computer oriented worker. He relieves the flight test engineer from knowing the specialties needed to maintain proficiency in "speaking with the computer".

The data acquisition function is rather more intricate than the documentation function. Its task is to steer the application of flight test instrumentation in the airplane , the data processing, the data base, the library layout and is thus oriented towards instrumentation department, data handling stations and central computer.

The net result is not that the flight test engineer has got to his disposition a "computer system" -whatever that may be- but that a unit consisting of several functions is working together to do computer aided experimental work.
A consequence is that departments must have porous borderlines to a higher degree than it was usual until now.
In fact it means that integration takes the place of a variety of stand alone facilities.

To finish this section the certification aspect has to be mentioned. The use of an ever increasing number of digital computers in the new technology airliners asks for a comprehensive certification procedure also including the software (ref 4). This can only be given if the above mentioned requirements of quick exchange of information, punctuality of purpose and exactness in reporting of test results are met; again automation seems to be the way to take over some routine tasks in order to spare energy and time for the essential decision activity.

Rewards of adequate result prediction.

It was mentioned earlier that quick-look monitoring in its present form enables to decide upon run acceptance or rejection by presentation of the essential results during and between the runs.

The most desirable solution is an on-board-computer or a telemetry -and-computer station, configured to:

- select data
- apply the necessary corrections and calculations
- convert data to engineering units
- present data on screen and/or in printing.

If an on-board-computer is not available dial presentation of some measurands or a hand held calculator-printer or pocket calculator with printing facility in an interface loop are possible replacements. Success criteria for direct comparison with the test results must be available in the same presentation.

Quick-look monitoring applied in this way can contribute to more efficient flight testing, provided that:

- preparation and predictions have received sufficient attention,
- cockpit crew, instrumentation engineer and flight test engineer have developed a very effective cooperation.

The first proviso should be the result of coordinated test preparation as mentioned in the previous section. All efforts have to be made by theoretical studies and simulations to make the flight test a predictable occurrance. If the result of a flight test is a big surprise, then the quality of preparations is questionable.

The second proviso follows from the first one. It requires that each of the participants is fully cognizant on the execution of the mission in its totality. Also here porous borderlines are needed, in this case not between departments but between functions, i.e. cockpit crew, flight test engineer and instrumentation engineer. One possible means to promote this is flight simulator sessions in which certain tests are studied with the three mentioned functions "on board". In this context the consoles for instrumentation engineer and flight test engineer should be provided with repeaters of the most important cockpit instruments.
Beside cooperation in the task during the flight also knowledge of weather forecast, day planning and status of airplane for all participants in the flight test is of vital importance.

Prerequisites of automation.

A system designer and programming capacity would be the simple answer to the question of prerequisites. Exactly here the trouble starts because a program made exclusivily on its own merits will immediately become a burden by lack of maintainability and compatability with other programs. A company strategy is needed to avoid or remove conflicts between interests, and the strategy has to be built up from bottom to top which means that it has to start with the programming function. Of course company computer departments work that way by handbooks and rules on system design but it is often felt as cumbersome to relay on programmers of the company computer department. Instead of that, local solutions are selected and a dedicated computer is installed, sometimes assisted by a number of slave computers.
The preference may be understandable from a viewpoint of success oriented creativity, but integrated systems are potentially better and cheaper. Here again porous borderlines between the company computer department and the producers as well as the users of the information are very important.
The development of better programming procedures like ADA (ref. 5) seems to indicate that the software and technical means are becoming available that will allow application of distributed intelligence in an integrated information network, without compromising future developments.
It is distributed intelligence operating in a company computer network, which can provide the flexibility and immediate response, needed to fulfill the changing requirements of automation in flight test engineering.

The design and development of such a system is time consuming. Attempts to accelerate the process may well fail on the point of view of vulnerability. The cautious approach is to divide the total concept in a number of modules and to check every new module in one of the running flight test programs as a stand-by system. The release for use should follow only when a comprehensive analysis of this tentative use has given sufficient confidence. Tentative use with available data from earlier programs should only be considered as a first step in the validation.
The correct execution of this task requires a separate quality assurance function in the instrumentation design and development group.

Since communication user-system is of utmost importance a user friendly command structure has to be built.
Adaptation of the command structure to the learning curve of the user requires attention, leading to maximum efficiency of the command structure for a trained user (ref. 6).

In the case of test programs flown from other locations than the home base the normal precautions should be taken to avoid stagnation. They belong to the logistic task. It may well be necessary to bring to the location a dependable electrical power source. A transportable quick-look system for initial data confirmation will serve two purposes: early corrective action after detection of unsatisfactory results and less dependence on telephone or other connections with the home base (ref. 7).

Conclusions.

The task of flight test engineering is to extract valid and needed data for the design process and for production of user information from efficient experiments in flight.
For test management purposes advantage can be taken in this task from automation, backed-up by proficient and flexible organizational linkages.
The quick exchange of information, punctuality in purpose and exactness in reporting, as achievable by automation, are desirable features to promote coherence in testing activities.
Coherence in testing activities is advantageous for those test subjects, for which simulation and real world test are supplementing each other.
Flight Test Documentation and Data Acquisition Systems are to be considered as specialized functions in flight test engineering.
Telemetered or on board computed quick-look monitoring is becoming a valuable instrument for cost-effective testing.
Adequate result prediction is a basic requirement for quick-look monitoring. Success criteria for direct comparison with the test results must be available on the quick-look presentation.
When flight test engineer and instrumentation engineer participate in the tests on board of the airplane, consoles are needed provided with sufficient indications to inform them on the execution of the mission in its totality.

A company policy on automation is of vital importance. An integrated information network has to be seen as the final stage of applying automation with computer aided experimental work (CAEX) as one of its components.

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THE USE OF ENGINEERING SIMULATION TO SUPPORT AIRCRAFT FLIGHT TESTING AT THE U.S. AIR FORCE FLIGHT TEST CENTER

by

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Abstract

The engineering simulator has become an integral part of the flight test tools to analyze flight test results and increase the knowledge gained from the flight tests themselves. The test team has been forced to closely scrutinize their weapon system and the spectrum of possible tests to determine the minimum tests required, determine the most important tests to fly, and to get the most from each flight hour. The engineering simulator provides an inexpensive means to closely scrutinize the weapon system and the proposed test plan. The U.S. Air Force Flight Test Center has successfully used the engineering simulator to educate test personnel, determine flight test envelopes, optimize test plans, enhance command and control procedures, develop modifications to flight control systems, investigate unexpected test results, investigate accidents, develop math models for training simulators, and provide practice flying for test pilots. Aircraft simulated have included the Space Shuttle, X-2, Dynasoar, lifting bodies, SR-71, C-133, A-7, F-5, F-15, F-16, and AFTI/F-16.

Introduction

The catch phrase in the Air Force test business these days is "test smarter". What test smarter really means is more for your money, greater efficiency, more information per flight hour. The emphasis on greater efficiency is forcing test managers to insist on fewer repeat tests and to obtain only the data required. This paper is about how the Air Force Flight Test Center (AFFTC) uses aircraft simulation to get the most from the flight test dollar.

At the AFFTC, the simulator is considered a teaching aid. The motto:

Learn before flight test

Learn the most from flight test

has been adopted. It has been said "the purpose of computing is insight, not numbers". The simulator is a tool that provides the test team with some insight into how and why their aircraft works the way it does.

History

The simulation facility began in 1954, simulating such aircraft as the X-2 and the Dynasoar. Over the years, over two dozen aircraft have been simulated for a large variety of purposes. The facility began with analog computers, evolved into a hybrid system, and in 1979 was refurbished into a completely digital facility. Table 1 lists some of the major projects and the analysis they accomplished using the simulator.

Facilities

The AFFTC simulation facility has four computer systems and four cockpits available. Three cockpits are fixed based and one is a moving base. All cockpits are fabricated, modified, and maintained in-house. The cockpits can be modified to satisfy any project's particular needs. A single CRT visual display is available. The CRT can be used on any of the fixed base cockpits.

The computer facilities consist of:

a) Perkin-Elmer 8/32

b) SEL 32/55

c) AD 10/PDP 11 Complex

d) EAI 2000

The Perkin-Elmer and SEL machines are the primary machines; handling 98 percent of the work load (Table 2).

In most cases two simulations can be run on the Perkin-Elmer machine simultaneously. One simulation at a time can be run on the SEL machine. Therefore, it is possible to be operating three simulations and have engineers doing batch processing all simultaneously.

However, in some cases a single simulation can be so large (i.e., requires most of the computation capability) that only one simulation can be run on the Perkin-Elmer machine. The AFTI/F-16 has succeeded in doing this. Most simulations require no more than two CPUs and run with frame times of 20 or 40 milliseconds. The AFTI/F-16, however, uses four CPUs and runs with a frame time of 15.6 milliseconds.

Four cockpits are available, three fixed base and one motion base. The fixed based cockpits include an AFTI/F-16, Space Shuttle, and a generic fighter. The motion base simulator is currently configured similar to an F-16 instrument panel but with a center stick. It can be used for any type aircraft, including side-by-side seating with yoke and column.

Flight Test Simulation

Why simulate, and why is simulation more important as a flight test tool today than yesterday? Basically, three major factors have changed the flight test scene.

First, back in the "good-old" days, aircraft had simple flight control systems; cables, springs, simple actuators. It was easy to understand how the control system worked. The actuator was connected directly to the stick and when the stick moved, the surface followed. Today aircraft have high authority stability and control augmentation systems with numerous feedback loops. An engineer can no longer understand how the control system, aerodynamics, and pilot will all interact simply by looking at block diagrams. Computers fly airplanes today, and the stick is merely a command lever for the computer. An engineer must do more than look at block diagrams to understand these systems. That "something more" is to build and fly a simulation.

Second, back in the good-old days, testers were concerned with testing the aircraft as a separate entity. That is, without considering the pilot as an integral part of the control system. By comparing an aircraft to a set of specifications they hoped to determine if it followed the recipe for a good aircraft. The tests were open loop, highly quantitative; pitch pulses and steady heading sideslips and the like. Today the pilot is recognized as an integral part of the feedback control system and testers are more concerned with operationally oriented testing. There is more concern with pilot work load and pilot transfer functions. In other words, how does the total system (pilot, control system, and aerodynamics) operate as a whole in a high gain situation; like when you are at low altitude, high speed, monitoring aircraft systems, looking for your target, and looking for the bad guys that are shooting at you.

Third, money; flight test projects are more expensive than previously. The simulator, in some cases, has permitted a 50 percent reduction in flight time.

For the engineer to understand his weapon system, for the engineer to gain some insight into how all this will work in a high-pilot-gain situation, he needs a tool that duplicates the real-life environment as much as possible. Aircraft designers make extensive use of simulation in some form during the design process. If the designer needs a simulator in order to understand and optimize his aircraft, what makes the flight test engineer think he can understand the system without using a simulator?

At the AFFTC, the flight test engineer is also the aerodynamicist and analyst. Those work tasks are not segregated. The engineer not only flies the tests, he also oversees the data processing, analyzes results, and reports on the completed tests and completed analysis.

If the flight test engineer is to:

a) understand how the control system works at the aerodynamic limits of the aircraft,

b) understand how the aircraft will respond when the flight control system is at its limits of performance,

c) design a test plan that fully exercises the system throughout the flight envelope safely and efficiently,

d) analyze unexpected flight test results, he is going to have to be very perceptive or make use of a simulator.

Education

The act of "building" a simulation forces the engineer to become very intimate with the aerodynamics and flight control system. An engineer cannot build a simulation without knowing a lot about his aircraft. By the time the simulation is built and checked out, the engineer has learned the effect of each stability derivative, and purpose and authority of every component in the control system. Even if the simulator is never flown the engineer has learned the intimate details of his aircraft.

Rehearse Flights

The second major use of the simulation facility is to rehearse flights. As in many situations practice makes perfect. The AFTI/F-16 Test Force estimates that they obtain 30 percent more useable data per flight by practicing each flight on the simulator. The improvement comes about as a result of the pilot and each engineer knowing exactly what to expect, knowing exactly who is going to be responsible for what, perfecting the pilot-to-ground-station communications, insuring the pilot understands the control input required to obtain the test data. In other words, it serves as a "super" pre-flight briefing; much better than sitting around a conference room table talking about the flight cards. Any inefficiencies in the flight plan or misunderstandings are readily recognized during the practice flight. During the actual aircraft flight fewer test points need to be repeated. When a repeat is necessary, a minimum of explanation is required. The time required for the engineers to evaluate the last maneuver (and clear the pilot to proceed with the next maneuver) is minimized because the engineers know exactly what response they should be seeing in the critical parameters.

The earliest and most avid users of the simulation facility were the engineers working on the X-series aircraft and lifting body vehicles. These fellows were taking giant steps forward to expand the known flight envelope with aircraft of radically new design. They also were faced with relatively short flights and relatively few flights. The aircraft were not flown twice a day, five days a week, as we sometimes do with A-10s and F-16s. They had to get the most out of their flight time and they had to understand as much as possible about the flight envelope before they flew. As a result, the simulator became a mandatory tool for the flight test engineer. That kind of utilization of the simulator is still with us and will continue to grow even for relatively conventional flight test projects.

Parametric Studies

The third major use of the simulator is to investigate or analyze:

a) changes to the control system

b) unexpected test results

c) hazardous or unusual flight conditions.

It is uneconomical and sometimes hazardous to investigate unexpected or unusual test results in the aircraft. The simulator gives the test team the ability to quickly try new control components, or repeat hazardous test conditions while observing any parameter. Very seldom does the test team have the opportunity to add new instrumentation in the middle of a test project. But, on the simulator any parameter can be called to the output queue in a matter of seconds.

were flown with a "controller" talking the pilot down. Under these conditions, in a matter of days, the most critical failures and most probable recovery techniques were identified. The flight test program then investigated a relatively small matrix of failure conditions and recovery techniques. This short flight test program was able to determine an easy recovery technique that would be acceptable under all the critical failure conditions.

The net result of utilizing the simulator in these three major areas is a tremendous amount of insight. It makes the entire test plan and test team more efficient. The testers learn where to spend most of their time testing and analyzing. They learn what the benign test conditions are and what the critical test conditions are. They learn what causes problems or unexpected test results; with emphasis on causes. They gain the insight to make better recommendations on improving the aircraft. It enhances the safety, as well as the efficiency, of test operations by improving their ability to make good decisions, in real-time, during hazardous flight tests.

Case Histories

C-133 Fixed Prop Pitch

When the Air Force was still flying C-133s, a problem was encountered with the variable propeller pitch mechanism. In simple terms, there was a failure mode that resulted in all props locking at whatever pitch they had at the moment of failure. In this failure mode, the characteristics of the fuel control mechanism were such that if the engine speed dropped below 88 percent the engine no longer responded to throttle commands. As you might expect, this situation made it difficult to land the aircraft.

The task given the Test Center was to develop a landing pattern for C-133s with fixed propeller pitch and fixed throttle. Obviously there was a large range of pitch-angle that the props could lock at. This combined with the range of available flap settings and descent profiles resulted in a large matrix of possible emergency procedures and landing patterns. It clearly was not safe nor efficient to flight test every possible combination.

The approach to the problem was as follows:

1) determine which prop-pitch failure conditions were most critical

2) determine which emergency procedures were clearly unacceptable and which showed the highest probability of success

3) flight test the most critical failures with the emergency procedures and landing patterns that showed the best chance for a successful landing.

Steps 1 and 2 were conducted on the simulator, step 3 with actual aircraft flight tests.

The simulator provided the opportunity to investigate a large matrix of conditions quickly, cheaply, and safely. With aircraft position displayed on an X-Y plotter and important parameters displayed on strip chart recorders, IFR approaches

A-7D Pure Yaw Departure

The A-7 aircraft is normally considered to depart with a relatively large rolling motion and relatively little yawing motion. A few years ago an A-7 departed in a pure yaw departure with no roll motion. The pilot reported that while applying normal control inputs to recover from a high angle-of-attack he experienced a very large sideslip and yaw rate without the usual rolling motion. The aircraft eventually yawed off into a spin.

It obviously being too hazardous and too expensive to conduct a departure-spin flight test program, the Test Center was asked to evaluate this unusual departure mode. An analysis of stability and control derivatives was insufficient to determine the feasibility of the reported departure. A real-time simulation was required to evaluate the effects of control inputs just prior to the departure. The maneuver that resulted in the aircraft departure was flown on the simulator. If normal recovery procedures were vigorously followed, no departure occurred. However, about 10 percent of the time if the pilot timed his aileron input just right, the aircraft would yaw to about 50 degrees sideslip with no roll motion and then continue into a spin. The simulation showed that with just the right control input a pilot could counteract the normal dihedral effect and allow the aircraft to develop a large yaw rate (and sideslip angle) with no rolling motion. Once the yaw rate was established it was impossible to stop it.

The simulator provided a safe and efficient tool to confirm what the pilot had reported. Without the simulator it is most probable the pilots comments alone would not have been sufficient, to convince people that this departure mode was possible.

F-5F and F-15 High Angle-of-Attack Tests

The high angle-of-attack, departure, and spin testing of fighter aircraft requires a large test matrix and is rather time consuming. The procedure is to start with simple 1-g stalls then gradually build up to more violent maneuvers; stalls with normal and moderate control inputs, stalls with large and vigorous control inputs, and finally aggressive maneuvers with multiple control inputs and pro-spin control inputs. The hazardous nature of the test requires a very methodical technique. Fortunately, many modern fighters are quite spin resistant, which means many of the preliminary steps in the spin testing are benign.

In the case of the F-15, the simulator was used to eliminate those maneuvers which clearly would not cause a departure and to determine what control inputs most probably would cause a departure. Thus saving considerable flight time.

The F-5F project took this procedure one step further by determining, on the simulator, what the angles-of-attack and sideslip boundaries for controlled flight were.

Each maneuver was flown on the simulator. Benign test conditions were eliminated from the flight test plan and the trends of controllability were watched closely. At the beginning of the actual flight test careful comparisons were made between simulator and aircraft. Once the accuracy of the simulation was established numerous build-up points were eliminated from the flight tests. The airplane was flown up to the predicted boundaries of controllability. If the aircraft reacted similar to the simulation, flight tests were terminated at that point.

The F-5F project estimated that they flew only half the number of flights they otherwise would have needed. Using the simulator in this fashion probably prevented them from departing the aircraft on several occasions. An experience of this nature clearly points out the cost effectiveness of a simulator.

YF-16 Control System Development

The YF-16 aircraft went through a fairly extensive control system optimization. The nature of a computer controlled fly-by-wire system allows frequent and rapid modifications to be made. Flight safety requires that changes to the control system be fully checked out on the ground before they are flown for the first time.

The simulation was constructed such that the aircraft's flight control computer could be connected directly to the simulator. After modifying the computer, but before actually flying it in the aircraft, the control box was flown on the simulator. This gave the test team an opportunity to check that the modification gave the desired effect, as well as checking that the box was still working correctly.

Having the simulator available for this purpose saved substantial time and allowed one-day turn around every time a flight control system change was made. It also gave the test team confidence that the modfication did not adversely affect the overall operation of the flight control system. That confidence allowed them to move directly into evaluation tests, without first performing excessive safety checks.

YF-16 Inadvertent Departure

Regardless of how complete your preparations are, occasionally you get surprised by unexpected test results. During the YF-16 tests the aircraft departed controlled flight under circumstances previously thought to be benign. It would have been too hazardous to repeat the maneuver without an anti-spin chute installed, and it would have been too time consuming and costly to install one at that point in the project.

Instead, the test team flew the maneuver on the simulator. They were able to repeat the maneuver numerous times. What they found was that under these particular circumstances the horizontal tail was hardover due to saturated feedback signals. With the tail already against a stop, the angle-of-attack limiting features of the control system could do no good.

An obvious advantage of the simulator, in this situation, is that any signal anywhere in the control system can be recorded and monitored. On an aircraft with hardwired instrumentation, it is not practical to add new instrumentation half-way through the test project. But with the simulator, the test team was able to determine the magnitude of every signal going to the actuator and thereby pinpoint the exact cause of the problem.

AFTI/F-16 Structural Limits

The AFTI/F-16 test team also got surprised on a test flight and exceeded a structural limit on the vertical tail during a flat turn maneuver. A quick analysis of the flight test results indicated that the control system was not operating as intended.

Within 24 hours of the actual incident the test team had repeated the maneuver on the simulator, isolated the control component which allowed the overshoot, and had a recommended fix established. But, perhaps most important of all, the test team acquired complete understanding of the cause of the problem. The insight into the problem gave the test team the knowledge they needed to avoid repeating the incident. As a result they could resume flying the aircraft immediately. If the test team did not have the simulator available to them it would have been many days or weeks before they could have isolated the cause and probable fix. Once again, the inherent capability to monitor any signal within the control system and to quickly change transfer functions proved the simulator's worth as a flight test tool.

F-15 Aileron-Rudder Interconnect

During the F-15 development flight test project, the pilots complained of sluggish directional response in high gain tracking tasks. The cause of the problem was not immediately apparent by analyzing the flight test results.

Similar tasks were flown on the simulator. The simulator had a better response than the aircraft. Comparison of simulator test results and aircraft test results showed the aircraft had a larger hysteresis in the aileron-rudder interconnect (ARI) than the simulator.

Revision of the interconnect's hardware components was not practical, even though that would have been the technically cleanest solution. The simulator study was continued to see what could be done in the flight control computer to alleviate the problem. Hysteresis was added to the simulation to match the aircraft. The study showed that by making a relatively simple change to the ARI gain the problem was alleviated. The hysteresis did not go away, but the change in gain provided enough rudder deflection to speed up the directional response in a typical tracking task.

The utility of having a simulator at the test site was demonstrated. The test pilots and engineers could use it in a timely fashion. The people flying the tests and analyzing the data were readily available to participate in the simulator study. Thereby providing the fastest possible assessment of the problem and the fix. The net result was that the development program was able to continue with a minimum of delays.

SR-71 Minimum Control Speed

The SR-71 aircraft has a very large operational weight range, a very large asymmetric thrust moment with the loss of one engine, and a fairly long engine spool down time. With the large amount of excess thrust available, even on one engine, it is possible for the aircraft to lose an engine below the static minimum control speed and accelerate to a speed faster than minimum control and only suffer a momentary loss of directional control. The circumstances were such that during takeoff the effective minimum control speed in the dynamic situation was significantly different than the classical static minimum control speed.

Part of the test team's job was to document this situation. It was determined that the large matrix of test points made it impractical to do a complete minimum control speed flight test program. The solution was to put the aircraft on a simulator, analyze the dynamic minimum control speed over the full range of weight and thrust, and determine the most critical set of conditions. A relatively few flight tests were conducted to confirm the accuracy of the simulation and to demonstrate the most critical conditions. However, it was not practical to demonstrate the single most critical condition, that of sea level thrust. However, the simulation's accuracy had been sufficiently proven and the simulator was used to determine the minimum control speed with sea level thrust.

Summary

These are just a few examples of utilizing a simulator to support flight test efforts. The entire list is far too long to include here. The important point is that almost every flight test project can make some use of a simulator. All projects will not use the simulator for all things. A simulator like any other tool has an appropriate use in some situations. But, almost all flight test projects, sooner or later, come across a situation where a simulator will significantly enhance their operation's safety, efficiency, or both. Having that simulation at the test site minimizes the time required to conduct the simulation study and maximizes the involvement of the test personnel in the study.

Abbreviations

AFFTC	Air Force Flight Test Center
AOA	angle of attack
CPU	central processing unit
CRT	cathode ray tube
DOF	degrees of freedom
FCS	flight control system
Kb	kilo byte
Mb	mega byte
PIO	pilot induced oscillation
RCS	reaction control system

Table 1 Some typical AFFTC flight test simulations

Project	Date	Utilization
X-20A Dynasoar Analog 6 DOF	1959-1963	Flight Planning Cockpit Evaluation Handling Qualities and Energy Management Studies
X-15A-2 Hybrid 6 DOF	1964-1969	Flight Planning Pilot Training (Backup) Real-Time Heating Handling Qualities and Energy Management Studies
F-104 Strake Analog 5 DOF	1964	Stick Kicker Boundary Test Technique
NF-104 Analog 5 DOF	1964-1966	Handling Qualities Low Speed RCS Kicker Design Accident Investigation Engine Gyroscopic Effects Drag Chute Spin Recovery Student Training Concepts
Analog 3 DOF		Optimum Zoom Profile Optimum Accel Profile Flight Planning Pilot Training (Limited) Parametric Studies (Wind, Weight, etc.) Placards for Students Student Training Concepts
M2-F2 Hybrid 6 DOF	1965-1966	Glide Flight Planning FCS Design Optimization FCS Failure Analysis Launch Characteristics Trim Changes Handling Qualities All Pilot Training Powered Flight Planning
SR-71 Hybrid 6 DOF	1965-1969	Handling Qualities Survey Flight Planning Performance (Engine) Minimum Control Speed Single Engine Performance Engine Failure at Cruise Longitudinal Stability at Cruise Accident Investigation
X-24A Hybrid 6 DOF	1969-1971	Performance Envelope Definition Handling Qualities Envelope Definition Launch Characteristics Envelope Expansion Flight Planning Emergency Procedures Pilot Training Operational Flight Support FCS Design Optimization Test Maneuver Development
X-24B Hybrid 6 DOF	1971-1975	FCS Design Aerodynamic Design Other Same as X-24A
A7 Lowspeed Analog 5 DOF	Feb 1972	Accident Investigation Departure Characteristics Digitac FCS Validation
F-15 Analog 5 DOF	1974	Handling Qualities Survey Departure Characteristics
F-5F Digital		Test Maneuver Development Departure Characteristics

Table 1 (Continued)

Project	Date	Utilization
YF-16 Analog 5, Limited 6 DOF	1974-1975	FCS Development Handling Qualities Survey High AOA Evaluation Departure Characteristics Test Maneuver Development Missile Flutter
Space Shuttle Hybrid 6 DOF	1976-1978	Low Speed Handling Qualities Survey Energy Management Test Maneuver Development Tailcone-off Flight Plan PIO Evaluation
Space Shuttle Hybrid 6 DOF Digital 6 DOF	1977-Present	Re-entry Handling Qualities Survey Entry Cross Range Capability FCS/Guidance Evaluation Aero Heating Evaluation Thermal Protection System Capability FCS Design Optimization Test Maneuver Development Auxiliary Power Unit Fuel Consumption Hydraulic System Management RCS Fuel Consumption
AFTI/F-16 Digital	1980-Present	FCS Development Test Maneuver Development Pilot Training Test Planning
Test Pilot School F-16A Digital	1981-Present	Student Projects
ASAT Digital	Present	Missile Separation Dynamics
F-15 Digital	Present	Test Maneuver Development Departure Characteristics

Table 2 Computer facilities

	Number of CPUs	Memory Per CPU	Shared Memory	Number of Disc Drives	Size Of Disc Drives
Perkin-Elmer	4 Real Time	256 Kb	256 Kb	3	80 Mb
	1 Batch	768 Kb			
SEL	2	192 Kb	64 Kb	1	40 Mb

GETTING A PARTNERSHIP INTO THE AIR.

TESTING OF THE SAAB-FAIRCHILD 340

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Abstract

In January 1980 an agreement was signed between Saab-Scania AB and Fairchild Industries Inc. to start a joint venture in developing a fuel-efficient commuter type aircraft in the 30-40 passenger class, intended for short-haul and multi-stop routes and suitable for operation in noise sensitive areas.

After a Program Definition Phase the decision was made in September 1980 to go on with the Saab-Fairchild 340 aiming at having the aircraft certified against FAR 25 but also, as a pilot case, against the complete European JAR 25 in time for first deliveries to customers in early 1984.

Within the agreement between the two partner companies a large part of the testing, including all flight testing, was assigned to Saab-Scania.

This paper gives an overview of the test program as a whole, from early preparations through first flight in January this year to scheduled certification date. It will also include a report on present status of the flight testing.

The objectives to be described are:

- The airplane itself and the goals that eventually led to its present design
- Certification basis, i.e. structure of the joint organization held together by the Swedish Board of Civil Aviation
- Approach to the test concept with emphasis on utilization of test rigs and development simulator in conjunction with early preparation of detailed test plans.
- Lay-out of the flight test program with designation of test aircraft to different objectives
- Special equipment utilized for flight testing, including test instrumentation and facilities for monitoring and processing of test data
- Summarized report on present status of the ongoing flight test program.

Introduction

The Saab-Scania Aerospace Division has for a long time been devoted to design and manufacturing of advanced military fighters. Since the mid 70´s however, facing restrictive Swedish export rules for weapons systems and longer life for each military project, we decided to look into the commercial airplane business. The company had left that field in 1951 when production of the 32-seat SCANDIA airliner was stopped in favour of military production.

Marketing research and design studies were started and the most peculiar designs came out from the scetch boards. As the studies went along it became more and more clear that the first ideas of slightly oversized bush plane designs had to be given up because what the market demanded was rather a high technology airliner. At this point it became clear that such an undertaking was too big for our company alone. We needed an equal partner for design and manufacturing but also for marketing so we started to look for such a partner where the big market is, in the United States. This finally led to the agreement with Fairchild Industries, who had a very similar situation and whose future plans showed up to conform very well with our own in this field.

The Airplane

The Saab-Fairchild 340 has been described in detail in many magazines so rather than going into those details I would like to present the airplane from the test engineers point of view. To be able to compete in a market like this the design had to meet certain goals that taken together would make it live up to the name of beeing first in a new generation of airliners of this size. The main goals were these:

- low fuel consumption
- low noise - inside and outboard
- long life
- low maintenance costs
- flexible interior layout

All this had to be designed into an airplane looking as conventional as possible because that in itself was considered as a selling point.

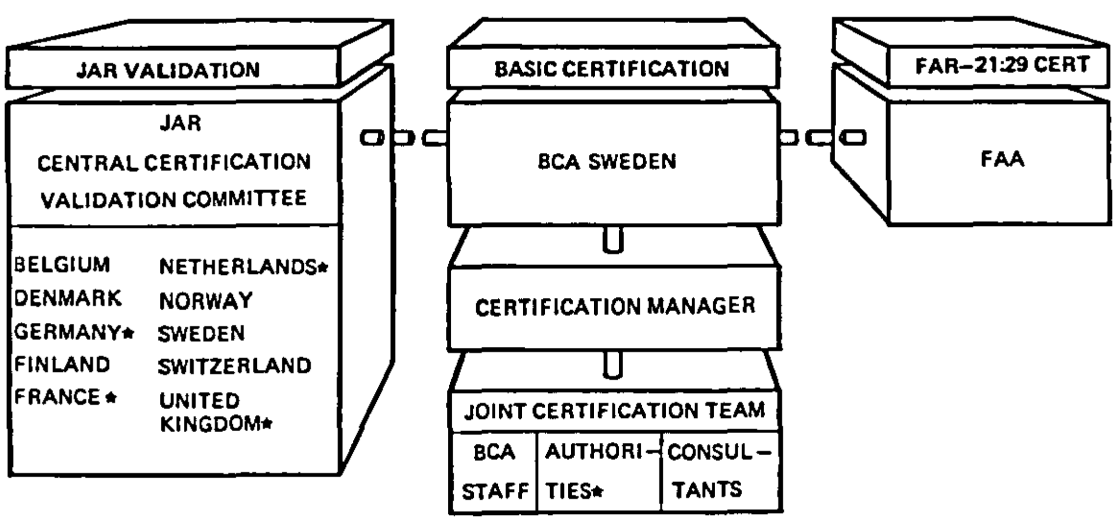
At first sight this does not seem to create a very exciting test program. But if you put together all the "news" and "firsts" that became necessary in order to fulfill these goals and the comparatively short time available to prove the whole design it showed up to be quite a challenge.

- A new type of low drag wing profile
- The first airplane to use the new generation of compact and highly fuel efficient turboprop engines
- The first commercial airplane with all composite propellers
- The first commuter with EFIS
- New bonding techniques for the structures
- The first airplane to be designed in inches and millimeters
- The first total joint venture between U.S. and Europe
- The new JAR 25 in parallel with FAR 25

Certification

For a long time there has been certain difficulties to offer a completely new designed airplane to the entire world market without beeing hampered by difficulties in national certification requirements. In recent years a number of European aviation authorities have gone together to solve this problem and the result has been presented as the Joint Airworthiness Requirements, JAR 25. It was an early decision in our project to apply for a complete certification according to JAR 25 as well as FAR 25. Because this had never been done before the Swedish Board of Civil Aviation was facing a huge task, particularly as a large

CERTIFICATION



transport airplane has not been certificiated by them for the last 35 years. It was clear that the Swedish authorities had to rely deeply on the know how held by their "big brothers" on both sides of the Atlantic.

The organization that was agreed upon shows the BCA as responsible for the basic type certificate which then has to be validated by the JAR committee and the FAA, but also as coordinator for the joint certification team, where representatives for some of the JAR nations and the FAA are actively participating in the certification work.

Test Concept

In order to keep the initial ambition to have the airplane certified and ready for first delivery in early 1984, it was clear that the time for flight testing had to be cut short. The fact that flight testing is the last part of a program always awakes a desire from everybody involved in earlier phases to steal time from it, and we were no exception. On the other hand our arguments to make flight testing more effective were widely accepted.

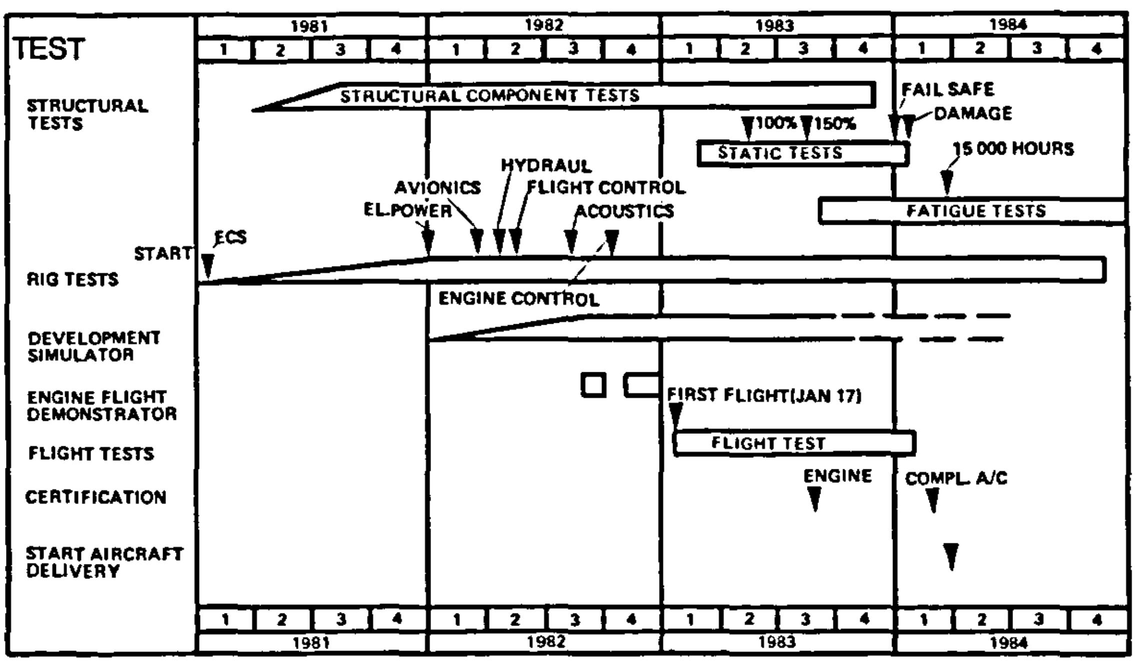
Our philosophy was to make every possible effort to prevent that design errors remained concealed until the test airplanes were ready. For that reason we asked for, and also got provided with, the following aids.

Numerous structural test articles including two complete airframes for static and fatique testing. Our common way for this has been to test static to ultimate load or in the fatique case to three life-times with undamaged articles. After that cracks have been inserted and testing continued and completed by determination of residual strength.

Separate rig setups for all significant systems. The advantage of this was that each system could be debugged independently and well ahead of the time when the system had to be built into the airplanes. All the test rigs have been paying off well in terms of early revealed remarks and the high reliability we have encountered in the test airplanes. The rigs have also been, and still are, useful tools for our designers when studying alternate designs and options.

A simple fixed base simulator, but working with quite sophisticated mathematical models of aerodynamics and flight control system has been utilized during most of the development. This has proved to be very useful, particularly for optimization

PROGRAM MASTER SCHEDULE



of handling qualities and assessment of emergency modes. A number of design changes and alternative fixes to potential problems have been developed based on simulation findings. Another important task in the simulator has been to check test procedures for different items such as performance, flying qualities and stall tests. Most of the simulation results have shown good similarity to the real airplane. An updating with flight test data is going on with the eventual goal to prove the simulator for use in the certification process.

The General Electric CT7 engine combined with new developed Hamilton Standard gear box and Dowty Rotol all composite propeller was considered one of the most critical items in the flight test program. General Electric provided a Grumman G-1 with a complete installation including front part of the nacelle on left side. A 50 hours demonstration program gave both us and them valuable experience that was fed into our own test program.

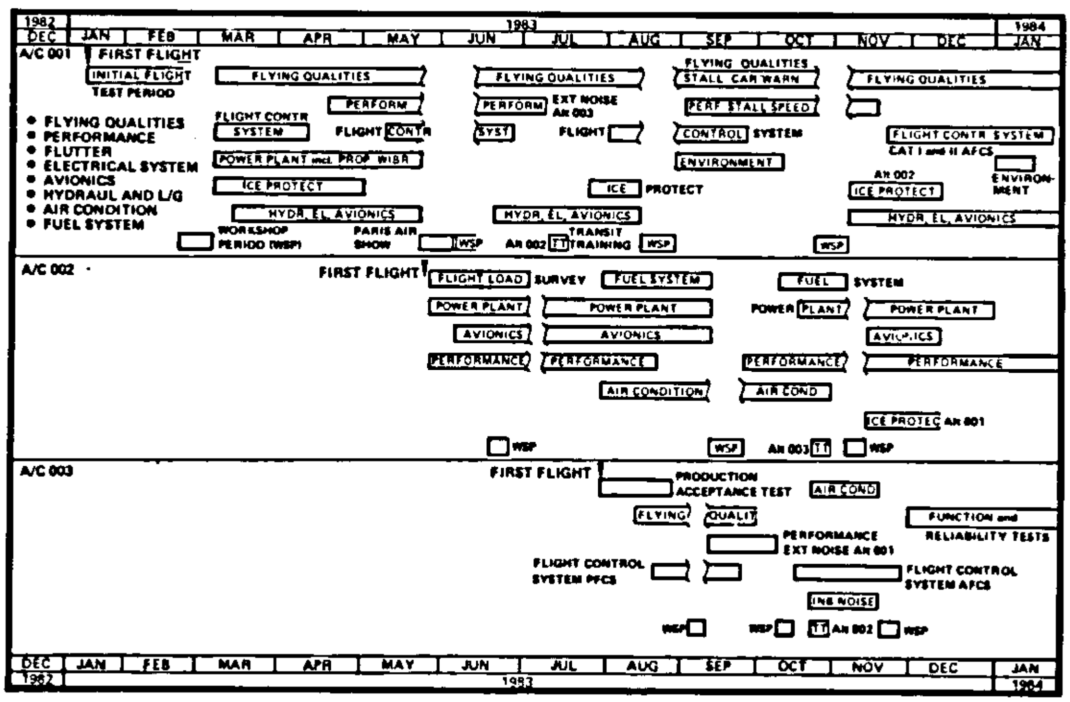
The task to get a complete flight test program accomplished in twelve months with only three airplanes called for a thorough pre-planning of all flights. Our goal was to get detailed plans finished for all testing that could be foreseen before first flight. Our concept was to work "backwards" meaning that we started by defining the reporting that was necessary to fulfill the certification requirements. That gave us the test conditions but also the needs for recording and reduction of data. The certification test plans were then put together and after reviewing by the authorities we succeeded in having them all approved before first flight in January. Next step was to make similar test plans for items where we knew that further development was necessary to bring the airplane to certificable status. That of course has been an ongoing work during the whole phase devoted to development.

Flight Test Program

The available ingredients for performing of the flight test part of the program were

- One fully instrumented airplane flying in January
- Another fully instrumented airplane flying in May
- A third non-instrumented but fully furnished airplane flying in August
- All the time we wanted until February 1 1984
- A test area mostly in the famous Scandinavian weather conditions, suitable - or unsuitable - for all kind of testing

FLIGHT TEST SCHEDULE



How to get the most out of this situation? First we decided to split the available time in three phases. The Initial Flight Phase was planned to be covered in five weeks with 40 flight hours planned in detail flight by flight in order to fulfill three main objectives:

- To expand the flight envelope
- To identify possible problem areas
- To get crews checked out

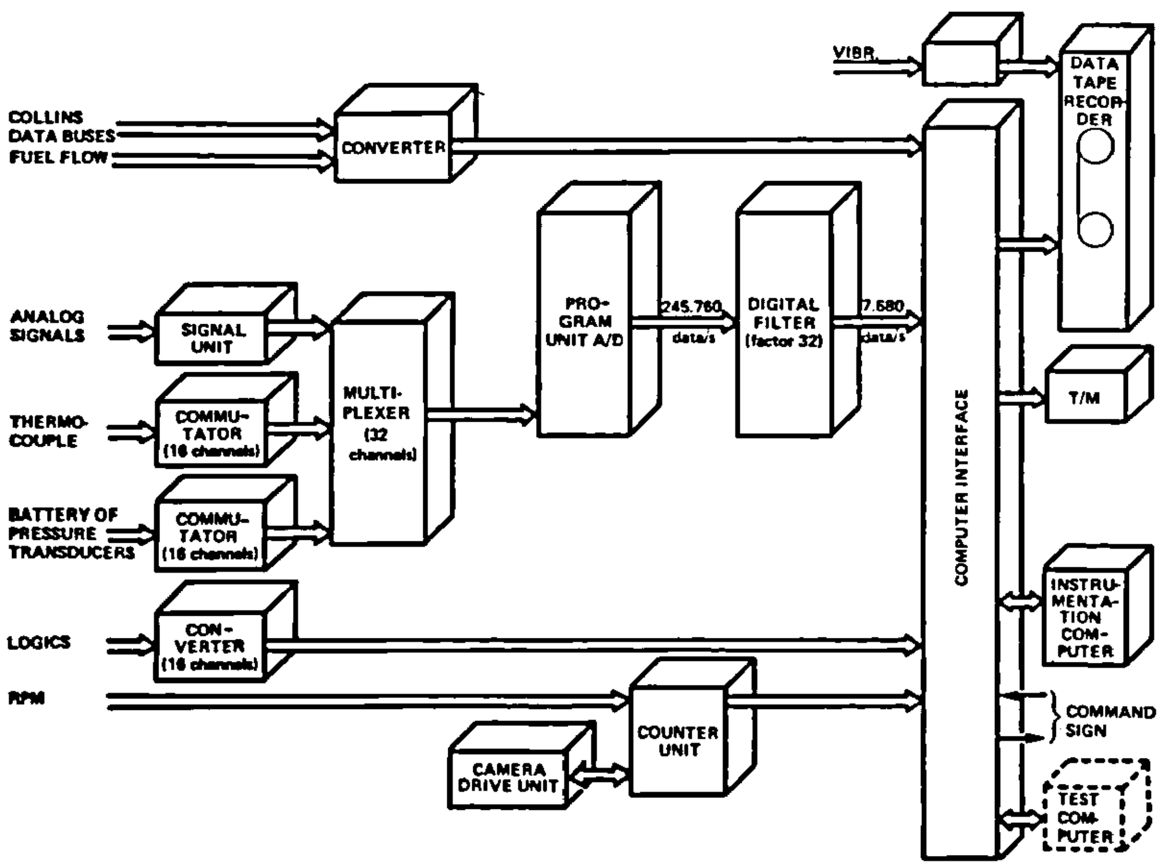
This was followed by the Development Flight Phase where all kinds of fixes were allowed to be checked out in order to bring the airplane configuration up to an acceptable status both for certification and for "Sales Appeal". From mid-year this phase was successively to be brought over to the Certification Flight Phase with the goal to have the airplane and all its systems in shape for certification by the end of September.
To fit into this plan the first airplane had to start its career as a "maid for all" until the second airplane was flying, but thereafter -001 has been devoted primarily to flying qualities, flight control, autopilot, avionics and electrical systems.

The second airplane started with a complete flight loads survey following extensive ground calibration. The other main objectives for -002 are performance, power plant, ECS and fuel system.
Test requiring certain environment, such as icing and external noise can be performed in either of these airplanes, depending on availability when the conditions are right.

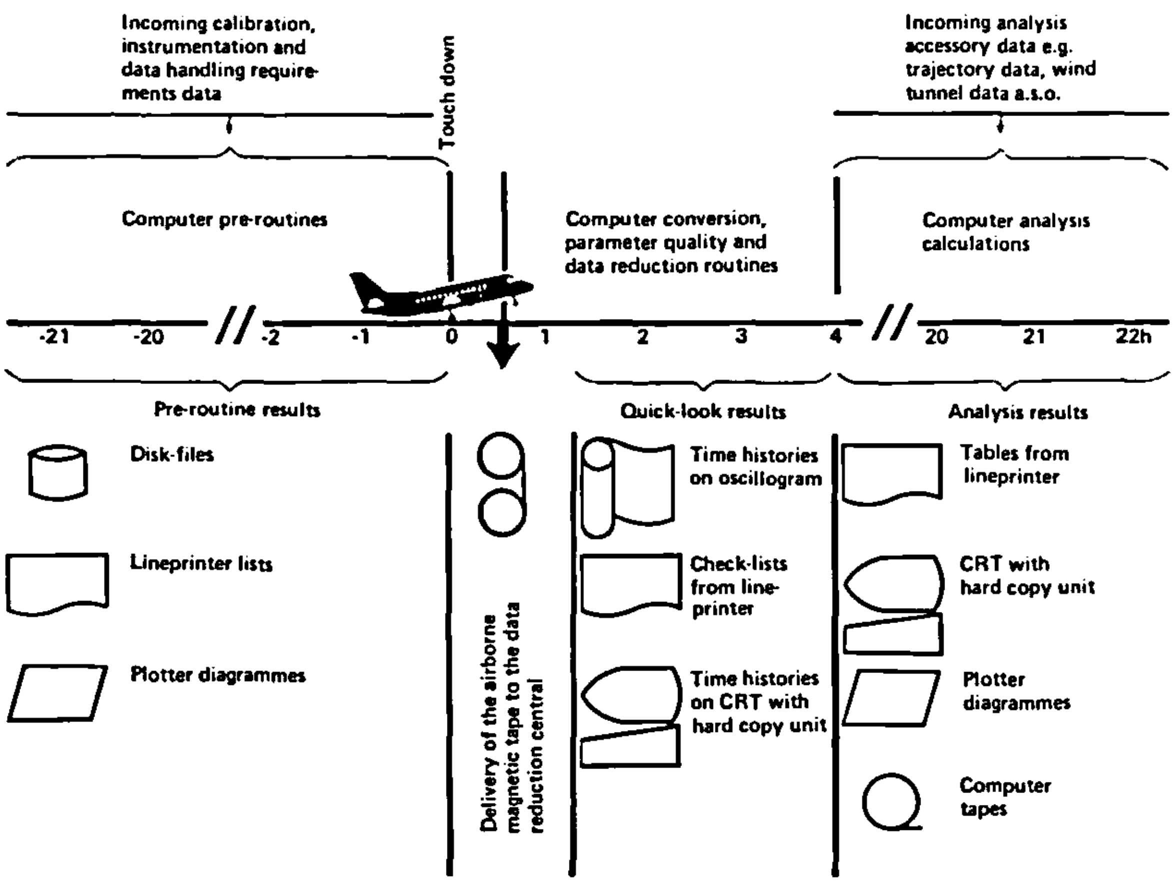
The third airplane is to be utilized for pilot assessment of handling qualities and autopilot but also for comfort testing of ECS system and inboard noise measurements. The eventual function and reliability testing is assigned to this airplane.

Test Equipment

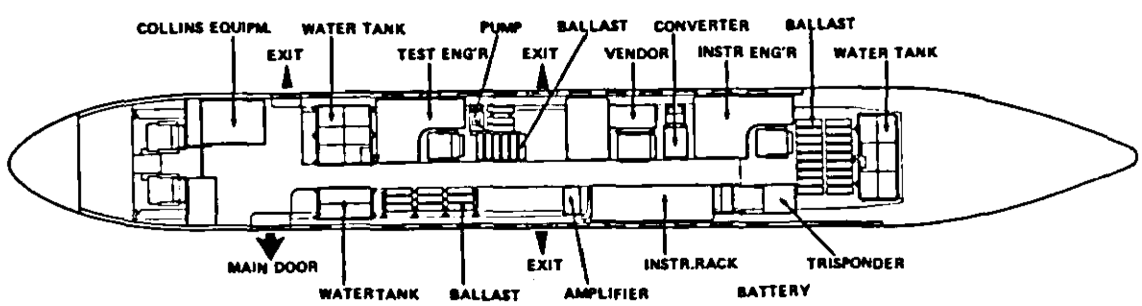
During the years of military flight testing an extensive development of test instrumentation and data reduction systems has taken place within our company. In the late seventies it was decided to unify this development to one system suitable for all thinkable projects through the eighties, large or small. So the MOD 80 was born. The symbol stands for a completely integrated system ranging from the parameter chains with transducers and signal conditioning through the central data acquisition unit to a data processing system on ground. The MOD 80 was first introduced in a JA37 Viggen test airplane and since then the expression "instrumentation error" has been scarce.



THE DATA REDUCTION PROCESS AS A FUNCTION OF TIME



CABIN LAYOUT SF 340-001



Data reduction capacity is such that presentation of time histories on CRT or hard copies normally can be done within four hours after landing. Data requiring more complex calculations, such as determination of flight loads or aerodynamic derivatives and coefficients can be available within 24 hours.

As our concept builds on utilizing each flight as much as possible by allowing concurrent testing of several items, the test airplane cabin is furnished with permanent stations for test oberver and instrumentation engineer. An additional observers desk can be installed for vendor tests such as propeller stress or engine fire extinguishing.

The test observers desk is equipped with airspeed, altitude and IRIG time indicators but also with six indicators and an eight channel trace recorder that easily can be programmed for presentation of any recorded parameter in engineering units during flight. A switch panel enables him to operate recording, cameras, event marker etc. The instrumentation engineer has equipment for operating the test computer and monitoring test data. If one flight comprises tests with large and highly different parameter needs he can re-program the data acquisition system during flight.

On flight deck the pilot has the same switch panel for handling of instrumentation system as the flight observer. Indicators for angle of attack, load factor and four programmable parameters are installed on the instrument panel. In airplane -001 is also a complete backup ADI and HSI installed on left side with conventional gyro reference for use during development testing of the EFIS/AHRS system.

During high risk flights such as flutter and stall tests, when the crew is restricted to pilot and co-pilot, the entire PCM data recording is transmitted to the ground by a telemetry link. Telemetry is also used in cases where computer processing of data from a certain part of a long flight cannot wait till after landing. The telemetry ground station is also equipped with trace recorders and indicators for real time monitoring by two or more observers.

Loading to required weights and c.g. positions is done by a combination of fixed lead ballast and a water ballast system, enabling c.g. shifting during flight while recording water levels and flow.

Among other special test equipment can be mentioned:

- Nose boom with vanes for sideslip and attack angles
- Camera installations for filming of ice build up on empennage, wings and air inlets
- Trailing cone, manually extended from the tail cone. Our test engineers just love that kind of exercise
- Low effect laser system for measuring of propeller blade angle
- Lear-Ziegler KT70L gyro reference installation (borrowed from the Viggen INS system)
- Shock charge installation in tips of wings, stabilizer, fin and rudder for flutter tests

Status Report

The earlier mentioned Initial Flight Phase philosophy proved to be quite a success in terms of achieving the goals we had set up. The 40 flight hours, five weeks plan, eventually became 47 flight hours during four weeks.

Within this phase the entire speed envelope was opened through flutter tests up to maximum design speed of 312 kts. Stall speeds were roughly verified and other performance data recorded. Altitude was increased in steps to maximum design 31000 ft and in parallel differential pressure was raised to specified 7.3 psi.

The c.g. envelope, which is very large for this airplane, was explored to aft limit and we performed take offs and landings at maximum weights. We also did a number of simulated single engine landings and wave offs preceeding complete engine shut downs and airstarts.

Very early the avionics and handling qualities showed a status allowing us to fly in limited IMC and also in darkness which was an important benefit at that time of the year.

In this period all seven pilots involved in the project had flown the airplane and ten test engineers had been engaged as crew members. Apart from the first flight, flutter and stall tests, all flights were made with the standard crew of four.

And what about problem areas?
Well we asked for it and we also could encounter a number of minor problems and a few major ones. But we also noted some nice surprises.

The basic flying qualities of the airplane were good in most respects but some deficiencies called for improvements. A tendency for elevator overbalancing with stick near full forward was corrected early by reshaping the elevator leading edge. Low directional dynamic stability and high pedal forces were both cured by changing the aerodynamic balancing of the rudder and introduction of a spring tab. Roll control forces were on the high side but this was a matter of optimizing the aileron tab gearing.

The propulsion package, including engine, gearbox, propeller and accessories has been performing flawless. A planned inspection of engine and gearbox after more than hundred engine flight hours revealed no signs of wear or damage. Noise levels are impressingly low and preliminary measurements indicate we are well below specified values.

The avionics system, particularly the new EFIS, created a number of early remarks but successive updates of the equipment have squared away most of them and the EFIS presentation is now much appreciated by the pilots.

The Development Flight Phase is now well underway and the airplanes had by end of June logged some 220 flight hours together. Airplane -002 made its first flight May 11, two weeks ahead of orginal schedule. Its first objective, the flight loads survey, was finished by mid June and the program is now concentrated on performance and power plant with certification flights starting in July. In airplane -001 some finetuning of flight controls still remain before certification of flying qualities is planned to begin,while most systems are ready for certification.

TESTING SMARTER

by

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Abstract

This paper presents decision techniques for test managers which allow them to investigate the needs for various aspects of test and evaluation at different phases of the system development. First, the need for a severe change in test philosophy is motivated. In response to the "Carlucci initiatives," or at least their underlying principles, several economies must be pursued. Second, decision theory and utility concepts are reviewed with an eye toward application in the test and evaluation arena. Third, the basic types of test information are addressed. There are, in general, three types of information which can be derived from testing:

1. Developmental
2. Decision point
3. Engineering model

The choice of information depends upon answers to several questions:

What is the ultimate use of the information?
What do we know about the system already?
What phase of the acquisition process are we in?
What are the program constraints?

These four critical questions are the keystone for the remainder of the paper. They can almost be used as a self-examination litany for test and program managers.

Next, the above three types of information (developmental, decision point, and engineering model) are discussed. Examples are given of how each type of information might fit into a decision tree for test and evaluation. Also how that decision tree might be truncated to reduce costs by accepting greater technical risks.

Finally, each concept is applied to an actual flight test program to show where savings could have been achieved. Alternately, of course, more about the system could have been learned given the resources expended.

Introduction

Mr. David Packard, Deputy Secretary of Defense from January 1969 to December 1971, stated, "As I joined Melvin Laird (Secretary of Defense) in the Spring of 1969 to help in the management of the Department of Defense, one of the most serious problems we faced was the unsatisfactory record of the Defense Department, the Military Services, and the industry in the job of development and procurement of new defense systems." Over a decade later, in his memorandum of March 27, 1981, Mr. Carlucci observed that, "We have a unique opportunity to improve significantly the way we do our planning and manage our resources." From these statements and from your personal knowledge it should be obvious that the problems associated with improving the weapons systems acquisition process represent a long-standing challenge. While some will view the Carlucci initiatives as just another iteration in the continuing struggle to conduct military industrial business dealings on a rational basis with the highest possible return on investment, this paper examines a philosophy to implement the intent of the initiatives. The heart of this philosophy is contained in Mr. Carlucci's statement that, "We all, as part of our management responsibility, have to assure that the large amount of funds being proposed for Defense are used wisely, effectively, and efficiently."

To provide a frame of reference for subsequent discussions, we will first discuss some of the factors that have a bearing on the need to implement existing management techniques for improvement of the weapons acquisition process. Using this background as a basis for change, decision theory and utility concepts which can be used to correct the perceived problems will be reviewed. Finally, these techniques and potential advantages will be discussed in the context of specific applications.

Background

As with any non-trivial problem, different perspectives usually obscure the method of solution. In the weapons acquisition process, the contractors are concerned with the economic aspects that affect their company and the government is concerned with countering hostile threats in the most economical method possible. Both perspectives deal in economics, yet from totally different positions.

Since the aerospace industry typically operates on the leading edge of technology, large investments are commonly required to develop and manufacture their products. The industry perspective is further affected by the high degree of variability in the demand for their products and the desire for a high return on investment as compensation for the risks taken. These factors have often been discussed from the manufacturing

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and capital investment viewpoint. The point to be made from a test and evaluation point of view is that since high risk programs are often fixed price contracts, the profit available to the contractors is decreased by cost growth and schedule delays. Schedule delays also function as a trigger mechanism for a counter-productive chain of events. With a drawn-out program, the threat that the government wishes to counter may change, in turn causing redesign and still further delay. Therefore, one of the key elements to the solution lies in the quality of the initial output. Doing something over because it wasn't done right the first time decreases efficiency, wastes money, and lowers productivity for both the government and the contractor. As test managers, this means we need to step through the test and evaluation phase crisply and professionally. This can be accomplished by a well-synchronized strategy. The major elements of this strategy must include taking full advantage of the scientific and technical capabilities in both industry and the government without unnecessary duplication. Assuming a reasonably cooperative and synergistic relationship between government and industry, the next section reviews decision theory and utility concepts for applications in the test and evaluation area.

Concept of Utility

Most of decision theory is based on economic principles. We choose the alternative with the highest profit, highest return on investment, lowest cost, etc. As we shall see below, decision rules generally are very clear cut for examples with known outcomes, even if the outcomes may occur with some uncertainty. In some instances, however, the costs, the benefits, or both may not have neat price tags on them. Utility theory attempts to provide a basis for rational decision making in those instances. Consider the following example.

Frequently (seemingly weekly), the Reader's Digest publishers mail out contest announcements. Some lucky individual, maybe even you, can win \$10, \$100, or even \$50,000. Have you ever stopped to figure the odds against winning? Reader's Digest has. The odds are phenomenal. Because of truth-in-advertising, those odds are included in the mail out. If you calculate your expected winnings for each time you play, you will find a result on the order of 9-10 cents. Recall the expression for expected value—

$$E(\text{Winning}) = \sum_{i=1}^{n} [Pr(i)*prize(i)]$$

where Pr(i) is the probability of winning prize money equal to prize(i) with n possible prizes. Now consider your cost to play the game: a 20-cent US Postal Service stamp (plus your valuable time). On a purely economic basis, you expect to lose approximately 10 cents each time you play the contest. Why do you and millions of Americans continue to make these seemingly irrational decisions? Utility! The utility of winning \$50,000, no matter how low the probability, is far greater to us than the negative utility of costing us a 20-cent stamp. Instead of comparing a 10-cent expected win to a 20-cent cost, the real comparison is between the utility of a \$50,000 win (or \$10,000 second prize) and a 20-cent cost. Clearly all but the most poverty-stricken choose to enter these types of contests.

Now let's move to a more abstract and applicable example involving flight test. Consider a new fighter aircraft undergoing envelope expansion. The test has reached the point where the high angle of attack boundary must be determined. If the test is run, a mishap may occur and aircraft, pilot, or both may be lost. If a mishap does not occur, valuable data may or may not be gathered. If the test is not conducted, either an artificially low angle-of-attack limit may be placed on aircraft operations to assure safety, or a mishap may occur and an operational pilot, aircraft, or both may be lost. In decision tree form, the problem looks like Figure 1. (Decision trees are discussed later). Not only do we not know the probabilities (p_1-p_7), we dont't have any means of identifying the relevant costs (C_1-C_7). This is not a hypothetical example, decisions like these are made quite frequently by test and program managers.

How would this decision be made? Well, a great deal of research could be done and the probabilities could be estimated. Some costs could be estimated: C_2,C_5,C_7, and (if you believe the numbers published in accident regulations) C_1. But what about the total cost of sustaining combat losses. Besides people and airplanes, isn't there a possibility of losing a battle, hence a war, hence our freedom? ("All for the want of a nail.") It turns out that these decisions are made by application of seasoned judgment, intuition, and concern over what people might think about the decision-maker.

Happily, as we continue, we will not attempt to resolve this latter example, but will apply utility theory to flight test engineering problems which arise after the programmatic issues are decided.

Our concept of utility which may, in itself, lead us to test smarter was borrowed from Bernoulli. The eighteenth century mathematician, Daniel Bernoulli, hypothesized that the true worth of an individual's wealth was the logarithm of the amount of money possessed. To paraphrase: in general, the usefulness of flight test data is some logarithmic function of the amount of data collected. When little or nothing is known about a system being flight tested, every dollar spent acquiring data returns a relatively large marginal amount of useful data. As more and more data are gathered, a dollar of data expense may only return to you a miniscule, insignificant, or meaningless amount of data. See Figure 2. Don't gather data when its marginal worth is less than cost.

Decision Theory

Decision Theory is a systematic approach for choosing the alternative which best meets the decision maker's needs. An underlying assumption is that the rational decision maker will choose the alternative which provides maximum profit, minimum cost, or otherwise optimum desired outcome (objective function). One graphical tool of decision theory is the decision tree. The example below shows a decision tree for the following decision:

	Act	
Event	Carry Umbrella	Leave Umbrella Home
Rain	Stay Dry	Get Wet
No Rain	Carry Unnecessary Burden	Be Dry and Free of Burden

In graphical form, the events, acts, and outcomes might look like Figure 3. Let's assume you wish to maximize comfort (objective function). If you know which event to expect, you know which act will provide the most desirable outcome. If you do not know which event to expect, you would have to provide comfort values or utility, discussed above) to each outcome, then pick the one which was maximum:

Stay Dry = 7
Get Wet = 0
Carry Unnecessary Burden = 4
Be Dry and Free of Burden = 10

You would prefer the last, which would lead you to choose "Leave Umbrella Home". The conservative individual might add the outcomes for each act and choose to "Carry Umbrella" because it sums to 11 while "Leave Umbrella Home" sums to 10. The intelligent individual might try to determine the odds of events occurring. By listening to the radio in the morning, you might find the chance of rain is 30%. If you put these numbers into an expected value equation, the following results:

E{comfort by carrying umbrella}

= (probability of rain) * (comfort index of 'stay dry')

+ (probability of no rain) * (comfort index of 'burden')

Therefore,

E{carry umbrella} = .3*7+ .7*4 =4.9 and,

E{leave umbrella home} = .3*0 + .7*10 = 7.0

You, as a rational decision maker, would leave the umbrella at home because that act gives you the highest expected value of your comfort index. (This decision process, choosing the act with the maximum expected value, is sometimes called Bayes Decision Rule.) On the other hand, if the chance of rain were 80%, the expected values would be:

E{carry umbrella} = .8*7 + .3*4 = 6.8

E{leave umbrella home} = .8*0 + .3*10 = 3.0

and you would carry the umbrella. (Note that by equating the two expectation equations, with the chance of rain as the unknown, you can discover the point of indifference: If the probability of rain is exactly 6/13, you must employ some other decision criterion because the expected values are equal.)

Using the first set of numbers, how would this look in decision tree form? See Figure 4. If you were a gardener, pruning unnecessary branches of a tree, you would prune the "Carry umbrella" branch (indicated by the double slash). In more elaborate decision trees, this is necessary to clearly mark the optimum decision path.

Obviously, the decision is a function of the comfort values assigned to the outcomes and the estimates of event probabilities. However, with a little algebraic manipulation, you can see that the decision is relatively insensitive to errors in these values and estimates. The decision will be essentially correct as long as the preference order of outcomes is maintained and their relative importance signified.

Information Classes

There is an adage that a smart man is able to extricate himself from a difficult situation, but a wise man avoids difficult situations. For the test and evaluation community the difficult situation in the adage could be taken as the need to make rapid decisions in the face of considerable uncertainty. While we will probably never truly avoid this difficult situation, we can lessen the degree of difficulty by proper planning using the tools discussed above. A number of books and papers have been written on how to lessen the difficulty. In general, the recipe is to systematically reduce the uncertainty without committing a major blunder under the pressure of time constraints. To apply this recipe you have to define what you want from the test program.

To define the desired result, it must be realized that the product of a test program is information which can be used to reduce uncertainty. Each program participant, however, needs a different type of information to carry out his or her function. For example, the design engineer needs technical information about how the system performs in the "real world" so he can refine the design. The program manager needs information to rationally choose between alternatives. The ultimate user needs to know how well the system will perform its intended mission. More formally, we define threes classes of information which can be derived from testing: (1) Developmental, (2) Decision point, and (3) Engineering model. Developmental information is fedback to design engineers to refine the design or to suggest changes to the system specification. It reduces technical risk to future phases of a program or future related programs. Decision point information is oriented toward specification compliance and can be acquired via hypothesis test techniques. Engineering model information is concerned with providing trends in dominant variables to quantify cause and effect relationships. This type of information may be used to validate analytical models. The models then can be used to quantify capabilities and limitations at operating conditions not physically examined in the course of the test program.

The basic function of program management is to make decisions. The test and evaluation problem facing the program manager lies in what information should be acquired. This decision must be influenced by several factors. Since acquired information has an associated cost, he must consider the benefit relative to the expenditure of resources and most programs have constraints on the resources available. Resource, as used here, includes money, qualified people, equipment, facilities, and time. Therefore, the program manager must make decisions and plans relative to the allocation of available resources to obtain the highest information return for the expenditure of those resources. The different types of information must be assigned priorities which depend on: (1) The phase of the acquisition process, (2) What is already known about the system, and (3) The environment that influences the constraints. Obviously if the program is in an early stage and the system is immature, the developmental information should enjoy a relatively high priority. However, decision information cannot be totally neglected since a program can be terminated or the resources further constrained if it no longer appears to be a viable solution to an operational problem. At this point it is necessary to consider the concepts of risk, uncertainty, and ignorance. Decision under risk implies that all of the possible outcomes are known and can be assigned a meaningful probability of occurrence. Uncertainty is a situation where all alternative outcomes are known, but there is no reasonable basis for assigning probabilities to their occurrences. Ignorance means that neither the alternative outcomes nor the probabilities of occurrence of a decision can be defined. The application of decision theory cannot create certainty where there is none, but it can be used as a guideline for rational decisions. At the outset of the decision process it would appear that the manager is operating on ignorance. Upon closer examination, this attitude commits two common errors: (1) Not using all of the information available and (2) Assuming constraints not really present. To make the decisions on testing, the manager has a test team consisting of both government and contractor representatives to assist. Unfortunately, an adversarial relationship among these representatives has been created by their different perspectives. Assuming that the needs of both groups could be met with a cooperative spirit, the remainder of this paper suggests how rational test decisions could be made.

Prior to the initiation of a program either the contractor, through IR&D, or the government has most likely undertaken a technology development effort or a series of related developments. The product of these development efforts is information about parts of the all overall system. The engineer who designs the system which will eventually reach flight test should have refined the initial design using information from the developmental effort. Furthermore, the designer had system specifications, a user-defined mission objective, and analysis tools available to him to evaluate his design prior to any construction or laboratory tests. Although it is almost certain that technical problems will arise due to interactions among the subsystems as they are assembled into the total system, at least some degree of knowledge is available. The point to be made is that the program manager and the test team need not operate on the basis of ignorance if they consider the designer as part of the test decision process.

Rational Test Management Decisions

Where is all this leading – Carlucci initiatives, decision theory, information classes? What is the test manager supposed to decide? The answer is intuitive, but repeated here for completeness: The test manager is supposed to decide how much testing, to what degree of accuracy and with what depth of analysis, is necessary to assure that a good product is fielded. How does he or she do this? Let's look at each of the three kinds of information and decisions in terms of a decision tree and Bayes decision rule.

Development Information

The purpose of development testing, as stated earlier, is to provide information which allows the designer to refine his design. What is the right amount of testing and how much post-development risk is acceptable? That is the management decision. The objective function can be minimum total "life cycle cost", maximum total system performance, or some combination of these. The decision about how much development testing is necessary is basically independent of the corollary decisions concerning specification compliance and operational testing. Here the manager must apply his or her judgment to the question of utility. Every designer wants "just a few more trials" before declaring the new system ready. To the decision maker this looks like Figure 5.

As in the case of the umbrella, each outcome can be given a utility value; perhaps even a cost. Depending on the design, initial technical risk, and results (trends) of development trials, accurate – albeit subjective – estimates can be made of the probability that the system is ready. Let's assume that the designer has asked for 100 more flight hours for testing – approximately \$1,000,000. If the system were ready, all that cost would be wasted. If the system were not ready, the savings in production dollars (because the problem was identified early) would be \$3,000,000.

If the decision were not to test and the system were ready, there is no relative loss or gain. If, on the other hand, the system were not ready, the fix would cost \$3,000,000. when retrofitted after production. The decision tree would appear like Figure 6. The point of indifference occurs when the probability of the system being ready is 5/6. Thus if the manager can determine through analysis of previous tests, combined with a priori data, that the probability that the system is ready exceeds 0.833+, the request for additional testing should be denied. Assume that an educated guess says the system has a 90% chance of being ready. See Figure 7 for the new decision tree.

The request would be denied because the better choice results in a \$300,000 net expense rather than a \$700,000 net expense. Note that the expected values are both negative – meaning that the manager is trading one risk for a lesser one. Also note that we didn't allow for discontinuing development altogether. Remember this discussion addresses test-related decisions– those made after programmatic decisions.

At the flight test stage, the model may be a simulator (with a cockpit) as most of us might expect, or it might be a computer program which runs either much faster or much slower than real-time. It might be a model based on first principles which very nearly duplicates the non-linear, stochastic world, or it might be simple transfer functions, empirically derived, which represent the influence of the various components on one another. In either case these models are useful to the test manager.

To see how the test manager might incorporate the engineering model into the decision process, let's look at Figure 15. Once the decision to test is made, either there is a model or there is not. Usually there is, so instead of an event circle, indicating uncertainty, there is a decision block to use or not to use. If the test manager uses the results of the model and the system meets the requirements, only those areas of high risk or marginal performance need to be tested. Other areas can be spot-checked. If the system model does not meet the requirements, what chance is there that the actual system can? Closer study of the component error budget revealed that the integrated system has little probability of passing its test criteria. However, the manager does have the option of changing the specification before testing if the <u>requirements</u> may still be fulfilled.

If the test manager is going to conduct a full-envelope test program anyway, the simulation or model can still be useful for determining test conditions and instrumentation requirements. Still, it seems rather ineffective to test in areas where the results are already known with a high degree of confidence. Spot-check, of course, but to run a full program anyway is a waste of money.

The other use of good engineering model information occurs during product improvement. This is especially important now that phased improvements are planned into an acquisition to reduce technical risk. The decision tree is much as in Figure 15. The decision, however, should often come out as a flight demonstration only since the improved system performance can be easily estimated by exercising the engineering model with the new component inserted into the system. The critical (corner) test conditions can be tested. The remainder of the performance envelope can be derived from exercising the model.

Examples

Modesty and our personal career goals prevent us from identifying the specific source of these examples. However, they should serve to illustrate the point. If they are recognizable, it is probably because similar situations are so frequently found.

Developmental

On a recent fighter air-to-air gunnery flight test program, several developmental flights were flown without all the systems being operational. In the system default modes, performance was degraded beyond specification values. Because of the requirement to include all data in the final performance evaluation, the system did not appear that it would achieve specification compliance. Hence, development was continued well beyond that necessary to fulfill the user's requirements. Ultimately the system performed so well that it met the specification including the erroneous data from early development.

Two excessive costs were incurred:

1. The extra development and refinement costs to obtain a system far better than necessary to meet the requirement (more is <u>not</u> always better).
2. The cost for 12-15 extra flight test sorties to allow further data to be gathered and analyzed.

Had the test manager used decision theory to analyze the need for extra development and flight test, he or she would have noted that the system had a high probability of meeting the requirement, even though the specification had not <u>yet</u> been demonstrated. At that point, the development could have been terminated and compliance testing begun. Within a few flights, the requirements would have been demonstrated. Several hundred thousand dollars could have been saved.

Decision Point

During a past fighter air-to-ground bombing flight test program, a series of 32 missions were flown to evaluate the performance of the six modes of a weapon delivery system. Nearly 400 bombs were dropped as part of this test. When the test was completed, the CEP was calculated. The system met its specifications with approximately 98% confidence. What's wrong with that? Nothing, except the test manager chose to ignore all prior information known about the system. Much data had been gathered from testing accomplished during the feasibility and development phases. The new system was functionally identical.

At the termination of development the system showed better than an 80% probability of meeting the requirement. Had this prior information been included in the analysis and combined with the compliance test results, the CEP would have been demonstrated with nearly the same confidence. The test program using the <u>a priori</u> information, however, would have taken <u>only 38-40</u> bombs.

There is always valid engineering information available on every system about to enter test. It may be from vendor tests, simulations, feasibility demonstrations, or engineering common sense. Judicious use of this <u>a priori</u> knowledge can significantly reduce the <u>cost of</u> flight test.

Engineering Model

The example here is difficult, because the program has not yet flown. All integrated avionics suites have sensor systems. At least one such system is about to enter flight test. The basic functions of the avionics suite have not changed significantly since it was first designed and flight tested some years back. At that time, however, no engineering model of the system was developed or retained, at least not by the test manager. Thus, as the new sensor component is about to begin flight test, the test manager is forced to test the entire system again. Had such

Decision Point

Of the three information classes, decision point information is the easiest to discuss. The first assumption of this part of the discussion is that the decision maker knows what his or her decision criteria and objectives actually are: cost, schedule, performance, politics, etc. The second is that the manager knows how much risk he or she is willing to take. The third assumption and the most significant is that the test manager has some a priori knowledge of the system.

Let's examine a system ready for compliance testing. This step is after developmental testing since using data from the early stages of development can lead to erroneous results. At this stage, the manager knows the required performance, the expected system performance, and the risk he or she is willing to take. You don't?! Of course you do. The required performance is contained in the specification, the expected system performance comes from the last few development trials, and the risk is a combination of the uncertainty in the evaluation process and the system readiness that led you to terminate development testing.

Let's say the requirement is for a certain attack aircraft to be able to drop unguided munitions, without the pilot seeing the target, with a circular error probable (CEP) of 300 feet. Let us further say that development testing was discontinued when the system had a 90% chance of meeting the point estimate of its specification. Also, let us suppose that instrumentation and analysis error has a root-sum-square value of 30 feet (one sigma). What then would be the test manager's decision?

First, a brief review of decision making using experimental information is needed. Generally, there is some evidence to indicate expected performance - called prior probabilities. Then, there is some probability that the system will meet the requirement even if it fails the test, and vice versa. For example, if there's a 50-50 chance of water in a proposed well site, and the local diviner can correctly predict water 90% of the time and can correctly predict no water 80% of the time, the corresponding probability tree would look like Figure 8.

To make this useful, we turn the probability tree "inside-out" and calculate the conditional probabilities of water or no water given the test results. See Figure 9. Recall Bayes' Theorem to reach the result. The notation is standard for probability representation.

The implication is that by testing, using the local diviner, we can improve our chances of being right about potential outcome. Because of the test we have improved our chances of success from 50% to over 80%. Now to apply this to our attack aircraft.

Figure 10 indicates the chronological sequence for the attack aircraft: The probabilities come from the prior analysis you've done. When development testing was terminated, before compliance testing was conducted, the probability of meeting the requirement was 90%. The conditional probabilities came from the following analysis:

Figure 11 represents the probability density functions for passing the test (achieving a 300-foot point estimate or better) and failing the test (achieving a 300-foot estimate or worse.) To be "fair" to a system which 'meets' the requirement, you might allow two sigma uncertainty for evaluation errors. Therefore, you would choose 360 feet as the cut off. A good system will therefore have a 0.9772 probability of passing. Pretty good odds. A failing system (10% is quite a bit over the requirement these days) might be a system with a point estimate of 330 feet or greater. If so, the probability of rejecting a good system is 0.1587. These decisions whatever the numbers might be, are up to the test manager and should be made before the test begins.

To continue with the example, let's invert the sequence to determine the new conditional probabilities. See Figure 12. Thus we have learned that if the system passes the test, the probability is greater than 98% that the system will meet or exceed the requirements. On the other hand, if the system fails the test, we are better than 80% certain that it does not meet requirements. The only thing left is to put this into a decision tree.

Decision trees, as you remember, always require outcomes. Let's use the outcomes from the previous class of information and generate a decision tree from that data. (Equate 'readiness' with 'meeting the requirement'.) See Figure 13. By looking at this analysis, you see that the best thing that could happen would be to spend $1,000,000 flight testing and for the system to work. That branch produces the least cost (risk). Including all other possibilities only increases the cost by $300,000. Now, if we eliminate the obviously inferior decisions and incorporate the remaining tree into the "test-no-test" tree we obtain Figure 14. Surprisingly, the best decision is not to test. Maybe that's not so surprising. When development was terminated, the probability of succeeding was 90%. That's quite high. By being aware of the designer's efforts to refine the system and terminating at the point you did, you would have saved $700,000 (expected value). What if you had not been aware of the development? You, as a test manager, would have been forced to conduct the million dollar test.

With different relevant costs, and different prior probabilities, can be quite different. The cost of developing the system to obtain 10% more chance of success may be greater than the cost of testing to raise the posterior probability of success that same amount.

Engineering Model

This form of information is at once both iterative and continuing. The process of culling this information begins with the designer's initial functional block diagram. At some point, perhaps concurrent with the completion of feasibility studies, the design is evaluated for expected performance. Later as the design progresses from bread board to brass board to prototype, each stage is tested. The expected performance is evaluated against the requirement. These models are providing information vital to the test manager. Likewise, each stage of test is providing proof of model fidelity back to the designer.

a model been available, the modified engineering model could have been exercised to verify improved capability. Also, high technical risk test conditions could have been selected for flight demonstration. A conservative estimate indicated that at least half of the planned flights would have been eliminated.

Conclusions

Testing smarter does not necessarily mean something with "high technology" written all over it. What it really means is to stop testing dumb.

Briefly, test managers should decide what they really need to know; the adversarial relationships in flight test should be discontinued - the free exchange of information between designer and aviator will allow a better product to be fielded in less time and with less cost; all available information should be used when making test decisions; and both test and program managers should plan ahead for product improvement tests by acquiring engineering models of the systems they are buying and evaluating.

We haven't said anything new in this paper. In fact we have restated a lot of things everybody already knows. It's time, however, to take some of these old, known precepts to heart again. They will help us test smarter.

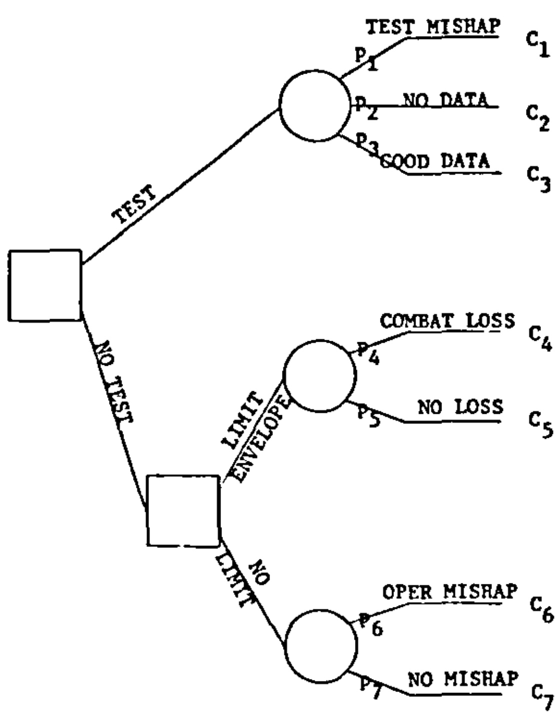

Figure 1. Test Risk Decision Tree

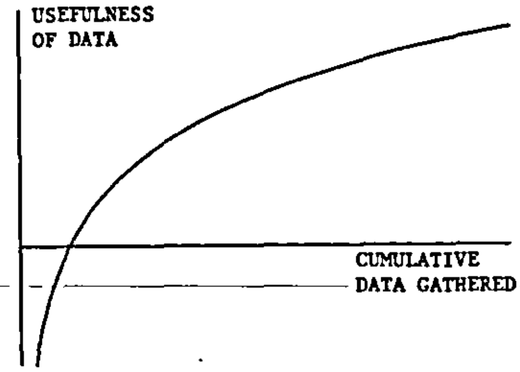

Figure 2. Utility of Data Gathered

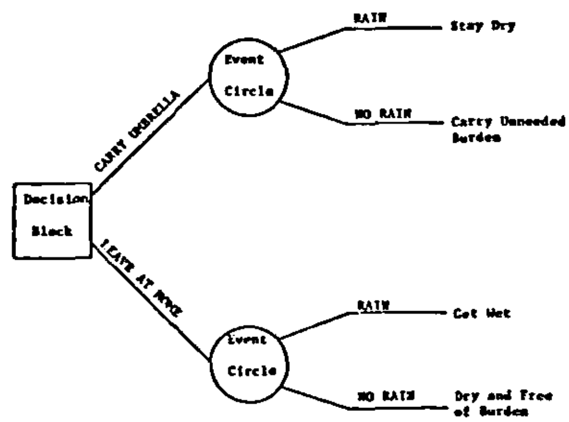

Figure 3. Sample Decision Tree

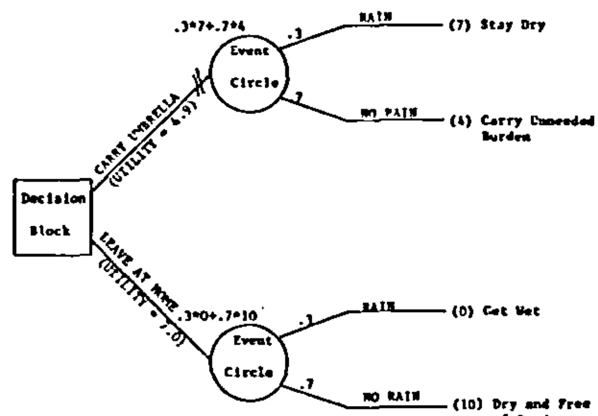

Figure 4. Decision Tree With Sample Utility and Probabilities

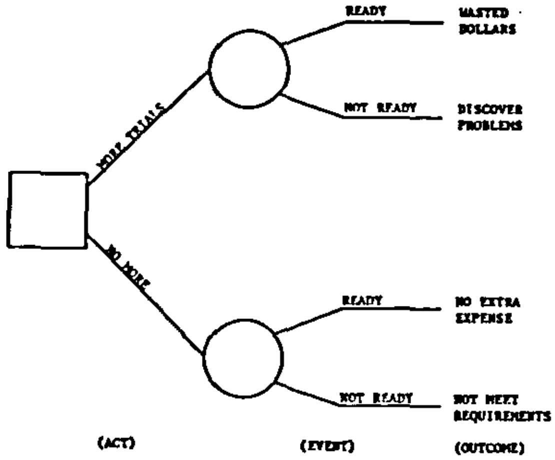

Figure 5. Decision Tree For Test Utility

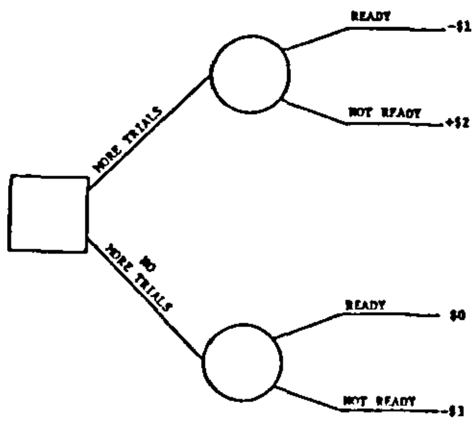

Figure 6. Decision Tree With Monetary Outcomes (Millions)

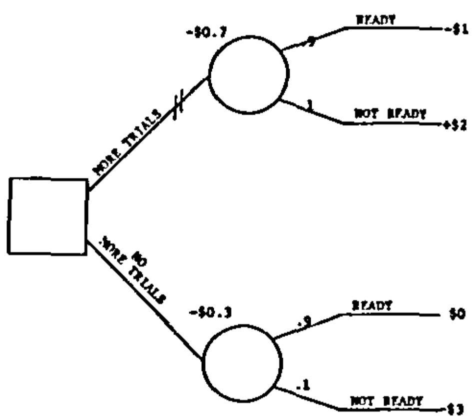

Figure 7. Decision Tree With Outcomes (Millions) and Probability

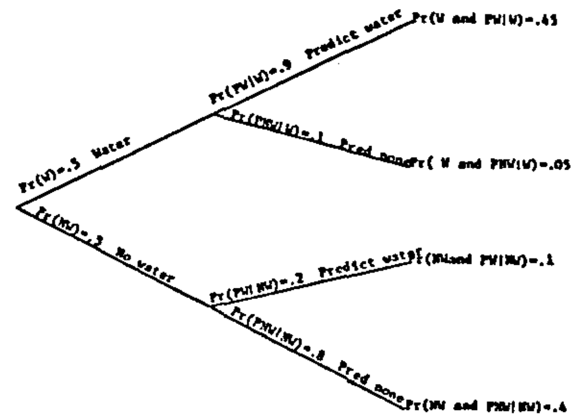

Figure 8. Probability Tree – Chronological

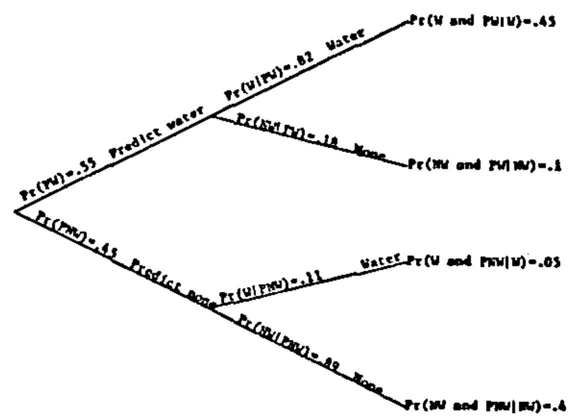

Figure 9. Probability Tree– Informational

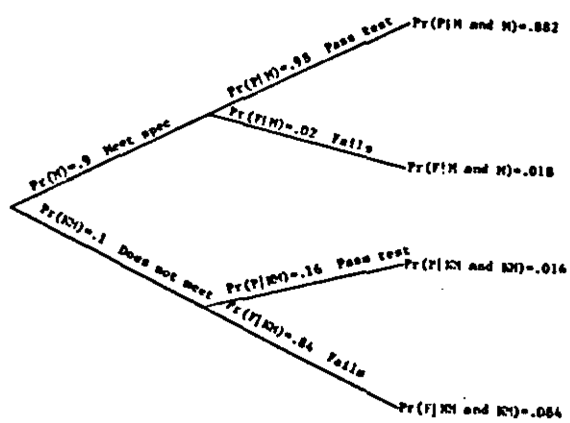

Figure 10. Probability Tree – Chronological

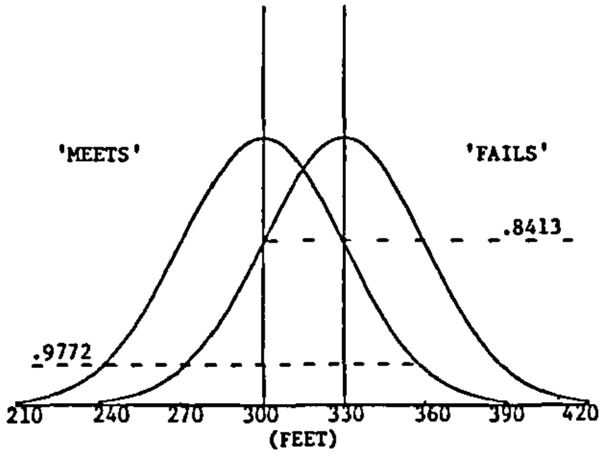

Figure 11. Probability Density Functions

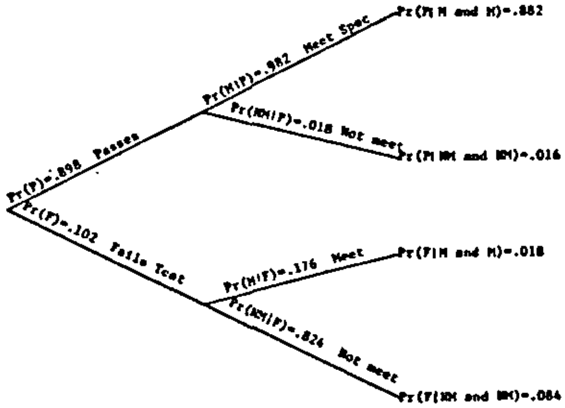

Figure 12. Probability Tree - Informational

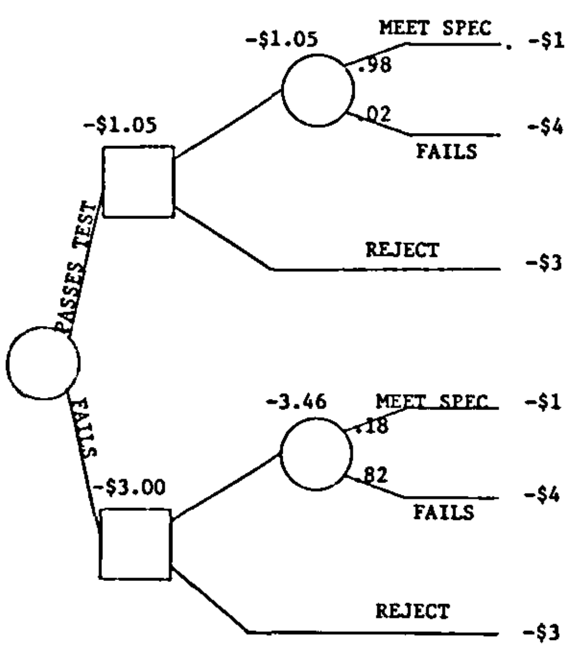

Figure 13. Decision Tree For Test Outcome (Cost In Millions)

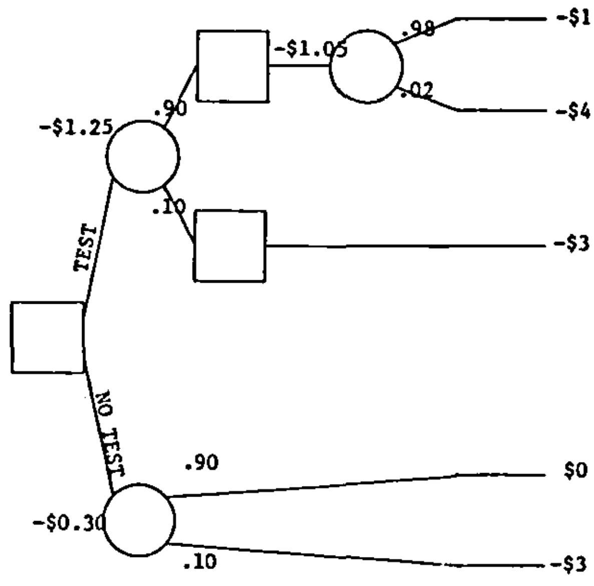

Figure 14. Decison Tree For 'Test-No Test' (Cost In Millions)

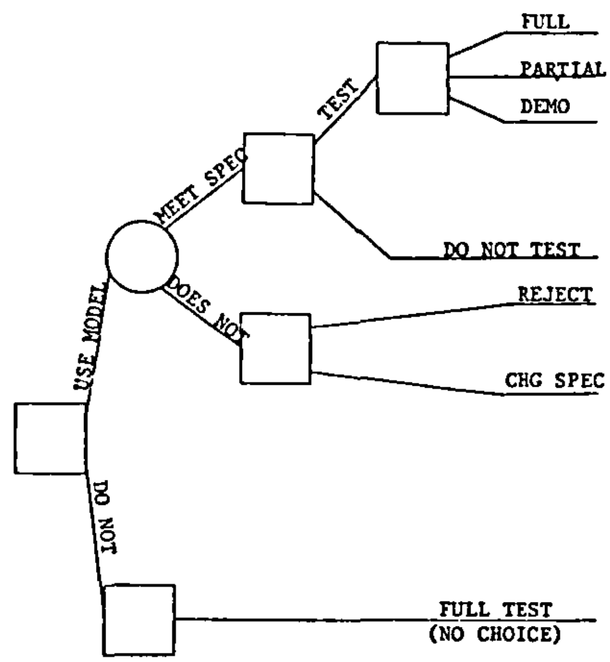

Figure 15. Engineering Model Decision Tree

FLIGHT TESTING THE DIGITAL ELECTRONIC ENGINE CONTROL (DEEC) — A UNIQUE MANAGEMENT EXPERIENCE

by

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Abstract

In 1979, NASA, Pratt & Whitney, and the Air Force individually concluded that the opportunity existed for a mutually beneficial flight test program to be conducted on the digital electronic engine control (DEEC) for the F100 engine. The management challenge was to develop a program that met each organization's needs. This was done with a minimum of contracts and other formal agreements typical of cooperative programs. The program included ground and altitude facility testing, flight engine modifications, and the ground and flight activities necessary to the flight test program. A four-phase, 18 month flight program was conducted at the NASA Ames Research Center's Dryden Flight Research Facility using an F-15 airplane. The DEEC system provided major improvements in F100 engine performance and operability, particularly for afterburner operation. The no-trim feature has been validated, with projected cost savings of $150 million. This paper describes the approach used for the flight test program, which resulted in major improvements in the DEEC system. In addition, the benefits of a strong and competent civil service technical capability, the absence of a "rigid" schedule, availability and application of related technical expertise from the Air Force, NASA, and the contractor, and a management approach built around mutually beneficial objectives are illustrated.

Background

The concept for the digital electronic engine control (DEEC) originated in the early 1970s, when it was recognized that the F100 engine performance, operability, reliability, and cost could be substantially improved by replacing the original mechanical/supervisory electronic control system with a full-authority digital control system. The engine manufacturer, under independent research and development (IR&D), designed and initiated testing on the DEEC system. The system was tested on bench test rigs, then on engines running at sea-level conditions, and eventually on research engines running in the altitude facilities at the Arnold Engineering Development Center (AEDC) and the NASA Lewis Research Center (LeRC). The computer and related control system hardware used during the initial testing was rack mounted and not suitable for flight testing. By 1978, the engine manufacturer had designed and initiated the procurement of flight-qualified control system hardware. During this period, the control logic and hardware had evolved to a point where further development was beyond the resources available in an IR&D project (Fig. 1). The engine manufacturer proposed to the Air Force that the DEEC system be included in the F100 engine model derivative program (EMDP I) for further development.

In 1976, the Air Force loaned NASA two F-15 airplanes to conduct research programs of mutual interest. The airplanes were used primarily for high-angle-of-attack flying qualities studies and propulsion system testing (Fig. 2). The propulsion tests involved engine-inlet compatibility, component improvement, and a propulsion system/airframe integration research program. NASA and the Air Force personnel had been discussing a follow-on program on integrated propulsion-flight controls for the F-15 airplane. As a precursor to the integrated controls program, a flight evaluation of the DEEC system on the F-15 airplane at the NASA Dryden Flight Research Facility was proposed, because the airplane was instrumented for propulsion testing and Dryden personnel were familiar with the airplane and the F100 engine.

Management

In 1979, the Air Force Deputy for Propulsion Office (ASD/YZ) requested that NASA Dryden conduct a cooperative, "mutually beneficial" flight evaluation of the DEEC and improved augmentor components. With concurrence from the NASA Headquarters Office of High-Speed Aircraft, the Dryden management agreed to conduct the evaluation on the F-15 airplane as part of a generic digital engine controls program. The memorandum of agreement (MOA) between the Air Force and NASA for earlier F-15 programs was deemed adequate to cover the DEEC tests. The Air Force (ASD/YZ) accepted the engine manufacturer's request to include the DEEC development activity in the EMDP I program.

The Air Force EMDP funds were not sufficient to provide a new engine for the flight evaluation, and no engines were available for loan. The engine manufacturer agreed to update, at his own expense, one of the prototype engines used in NASA's propulsion program to the production F100(3) configuration, and to install the DEEC system on it. A loan agreement was developed and implemented between NASA Dryden and the engine manufacturer so that the manufacturer could assume custody of the engine during modification. Responsibilities and obligations of both NASA and the contractor were delineated in the agreement with respect to data rights, engineering support, and ownership of the DEEC system.

The DEEC flight evaluation program was directed by middle management personnel within NASA, the Air Force, and the contractor, with all the benefits inherent in such management structures. The lines of communication were short and direct, with the majority of the technical deci-

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sions being made at the first-line management level. Nearly all program decisions were also made at the first or second level of management. The NASA Dryden F-15 project manager was responsible for the NASA program management, and the Chief of the Dryden Propulsion Section was responsible for coordinating and setting the technical direction of the program. A block diagram of the organizational structure is shown in Fig. 3. The arrows indicate the paths taken for almost all technical and programmatic decisions and the level of the communications. Also shown are the levels at which the formal agreements were executed.

The program was conducted with no exchange of funds between NASA and the Air Force or between NASA and the engine manufacturer. Limited funds were provided to the engine manufacturer by the Air Force. Essentially, each organization — NASA, Air Force, and the Contractor — provided the resources necessary for the discharge of their respective program responsibilities. A key element in the success of the program was the ability to divide the program responsibilities among the three participating organizations in a clear and unambiguous manner. The various organizational responsibilities were as follows:

NASA —

- Conducted the flight test program and reported results
- Provided altitude facility support as necessary
- Provided funding for the F-15 airplane and altitude facility
- Assured flight safety

USAF —

- Provided AEDC test support including funding
- Developed and implemented USAF flight clearance requirements
- Conducted program reviews

Pratt & Whitney —

- Conducted sea-level tests and supported altitude and flight tests
- Provided DEEC control hardware and software
- Updated F100(2 7/8) engine to F100(3) configuration
- Provided funding for hardware and software development and support

There was very little duplication or overlap. It should also be noted that the three organizations were in agreement on the broad technical goals of the program, even though each organization's motivation to support the goals was different.

Program Elements

The Air Force conducted a series of flight clearance tests on the DEEC system, including a test of the flight engine, at AEDC. The Dryden Flight Readiness Review Board, which addresses issues of technical readiness and flight safety, included a member with extensive digital flight control experience. Based on recommendations from Dryden personnel, the contractor's software verification and validation procedures were significantly modified, and a closed-loop real-time simulation was added to perform dynamic tests on each DEEC unit. Figure 4 shows the procedures that were used. The engine was installed in the F-15 airplane and the first flight was flown in June 1981. The initial flight test period (phase 1) was completed with seven flights flown and no significant problems encountered. A team of NASA and contractor engineers on site at Edwards and an extensive real-time data display and analysis capability expedited the flight evaluation.

During the flight clearance testing at AEDC, several improvements to the DEEC logic and augmentor were investigated. These improvements were incorporated into a second software package, which was flown in phase 2 from October 1981 to February 1982. In phase 2, the air-start envelope was investigated, the backup control operation was studied, and the augmentor performance in the high-altitude/low-speed region was determined. Significant problems were identified in the augmentor testing, including a nozzle instability. The instability had not been predicted from simulation studies and had not been observed in the altitude facility tests. To investigate the instability, the highly instrumented F100 engine at NASA Lewis was tested at the appropriate conditions. The cause of the instability was identified and various logic changes were evaluated. The availability of the engine and the flexibility of the Lewis test plan made it possible to rapidly generate a new software package for phase 3, which was flown in September 1982.

The final phase of the DEEC flight evaluation, phase 4, was flown in January and February 1983. This phase included logic changes in the control system to improve the engine acceleration characteristics, and the addition of an ultraviolet detector in the augmentor to detect combustion. The various ground and flight test phases are summarized in Fig. 5.

Technical Results

The F-15 DEEC evaluation has had several significant technical results. First, the DEEC system has been shown to eliminate the need for trim (periodic adjustment of the control system to keep the engine operating within limits). Figure 6 shows the test results for the DEEC no-trim mode, in which the closed-loop control is able to automatically compensate for changes in engine condition, control system tolerances, and fuel characteristics. The results of engine tests at sea-level static conditions, in the altitude facility, and in flight are compared, and all fall well within the allowable band of engine pressure ratio (EPR).

Another significant technical result was a large improvement in augmentor transient capability. Figure 7 shows a number of test conditions in the low-airspeed/high-altitude portion of the flight envelope at which idle-to-maximum power throttle snap transients were attempted. The lines represent the success boundaries, with points below the line successful and points above the line unsuccessful due to fan stalls or augmentor blowouts. The lowest line represents the standard F100 engine without DEEC. At the end of the second phase of DEEC testing, the line had been raised only slightly, much less than predicted. The management team determined that additional altitude testing, simulation, and analysis were required, and (as mentioned previously) altitude tests were run on another F100 engine at the NASA Lewis Research Center. The phase 3 test results showed further improvement, and finally, in phase 4, all tests were completed successfully. This step-by-step improvement shows the validity of an integrated ground test and flight test program that is not excessively constrained by a rigid schedule. The elimination of augmentor transient-induced stalls is a major improvement that will eventually benefit airplanes in the F-16 fleet.

An additional benefit of the DEEC is the improved air-start capability. Figure 8 shows the successful air-start boundary for the DEEC-equipped F100 engine and the air-start boundary for the standard F100 engine. The DEEC allows the engine to be successfully air-started at speeds 50 to 75 knots slower than the speeds at which the standard F100 engine can be air-started. The ability to air-start engines at a speed of 200 knots is a significant advantage to an airplane like the F-16, because its maximum lift-to-drag ratio occurs at about 200 knots.

Impact

The flight evaluation of the DEEC system has had a significant and far-reaching impact on afterburning turbofan controls technology and its use in and application to military aircraft. Just the elimination of the requirement to trim the engine is projected to save the Air Force $150 million in life-cycle costs for DEEC-equipped engines. In the summer of 1982, based on the DEEC performance and on the confidence generated by the results of the F-15 flight evaluation, the Air Force decided to proceed with full-scale development of the DEEC, with plans to incorporate the DEEC into production in 1984. Both the Air Force Deputy for Propulsion and the engine manufacturer acknowledged the importance of the timely F-15 flight evaluation in making this decision. One benefit of the NASA participation in the DEEC evaluation was the independent engineering assessment of the DEEC performance, which is not possible when tests are performed by the contractor.

NASA is now a recognized authority in the flight test of digital engine controls. This capability complements an existing strong capability in the digital flight control area, and establishes NASA as a potential leader for future propulsion-flight control integration projects. The F-15 DEEC flight evaluation has provided NASA with a fully instrumented DEEC-equipped F100 engine that may be used for future programs. In addition, NASA support to the Air Force in an important project provides a basis for cooperative efforts on future Air Force propulsion flight evaluations, such as the EMDP II (PW 1128 engine) program.

The technology for digital engine control has been advanced significantly by the F-15 DEEC flight evaluation. The DEEC system, with full-authority digital control and integral hydromechanical backup control, has significantly improved engine control performance, cost, and reliability. Through the participation of Dryden digital flight control specialists in the software verification and validation, the experience of the digital flight control community has been transferred to the engine control community.

Conclusions

Several lessons have been learned through the conduct of this program. Some are new; but others reinforce and reemphasize previous experience.

1. An experimental or research program produces the most effective results when the program is managed and directed at the lowest practical level. The ability to make management and technical decisions at the same organizational level also improved the productivity of the DEEC program.

2. A strong and competent civil service technical capability is an asset and provides effective technical direction to a program that depends heavily on contractor implementation. Many times during the course of the program, Dryden Propulsion Section personnel provided the technical leadership that ultimately resulted in the success of the program.

3. The absence of a "rigid" schedule allows the maximum technical knowledge to be extracted from a program. NASA, Air Force, and contractor engineers had time to evaluate data, identify problems, develop solutions, and validate the solutions through subsequent testing.

4. The application of multi-Center knowledge, skills, and facilities is a very effective method for the timely identification and solution of complex problems. For example, the timely availability of the F100 test engine and engineering expertise at the NASA Lewis Research Center made it possible to quickly diagnose a control system instability and develop a solution.

5. Technically excellent and cost effective programs can be implemented between various government agencies and private industry when mutually beneficial goals and objectives are established, and responsibilities are divided on the basis of the capabilities of the individual organizations.

6. The knowledge and experience gained in one technical discipline (such as flight controls) can be transferred to a related discipline (such as propulsion controls) when important issues such as flight safety are involved. The Controls Section personnel at Dryden have many years' experience in qualifying digital flight control systems. Through management action, this expertise became a valuable tool for the digital electronic engine control (DEEC) program.

7. The mission and management independence of NASA personnel allows them to provide unbiased technical judgments on the value of advanced technologies for use by the military in their decision-making processes.

Future Directions

The success of the DEEC program has spawned another cooperative propulsion program between NASA and the Air Force. The program is the flight evaluation of a new F100 derivative engine designated the PW 1128. The program is again characterized by agreement on the major goals and objectives, and by application of the resources of each organization in their respective areas of strength. An MOA that defines the responsibilities of the respective organizations has been signed by NASA and the Air Force. The flight evaluation of the PW 1128 engine was conducted in the spring and summer of 1983.

NASA Ames-Dryden and NASA Lewis are continuing their research into the fault detection and accommodation characteristics of digital engine controls with the endorsement of the Air Force. Flight evaluation of this research is planned after completion of the EMDP II (PW 1128) program.

Digital engine controls and digital flight controls have been proved in flight and are individually ready for application. Research engineers within NASA and the Air Force and in cooperation with industry are formulating a program to provide a flight evaluation of integrated engine and flight controls. To assure accurate results, the integration benefits must be assessed in the dynamic flight environment.

Event	1973	1974	1975	1976	1977	1978	1979
Configuration studies	▽						
USAF design review		▽					
Full-scale development proposal			▽				
Breadboard engine test				▽			
F100 configuration studies				▽			
F-15 DEEC proposal						▽	
Group I hardware received						▽	
NASA LeRC P072 engine tests						▽	
Group I engine test							▽

Fig. 1 Early history of the DEEC program.

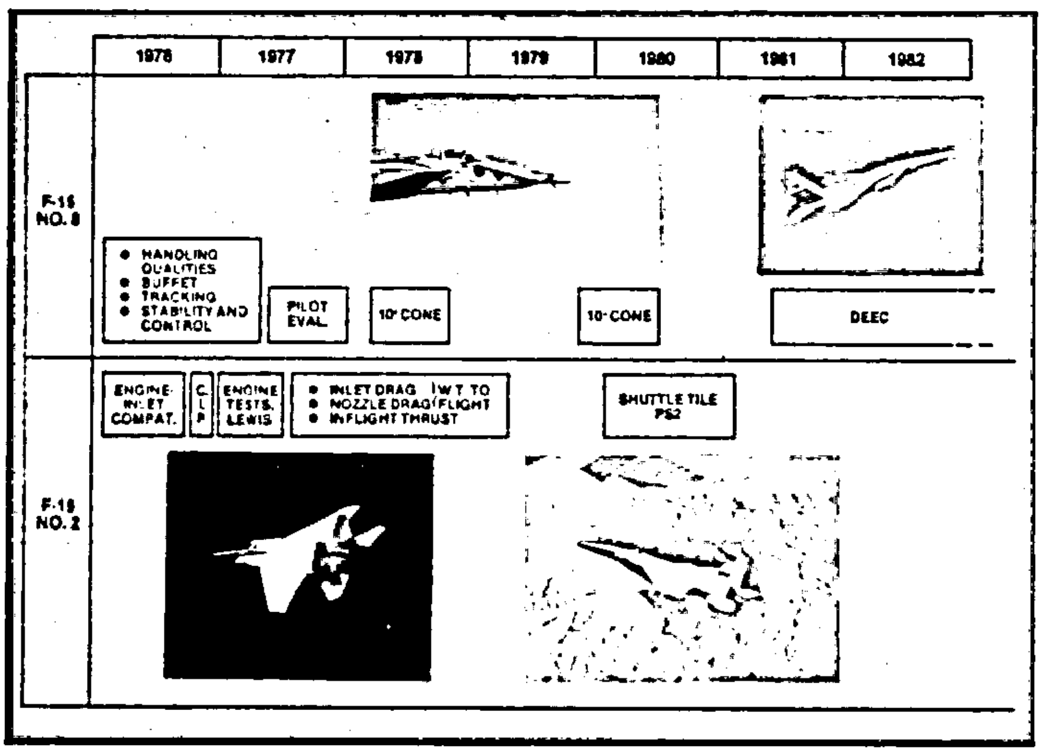

Fig. 2 Dryden F-15 programs. (C.I.P. = component improvement program; PS2 = static pressure at engine face; and W.T. = wind tunnel.)

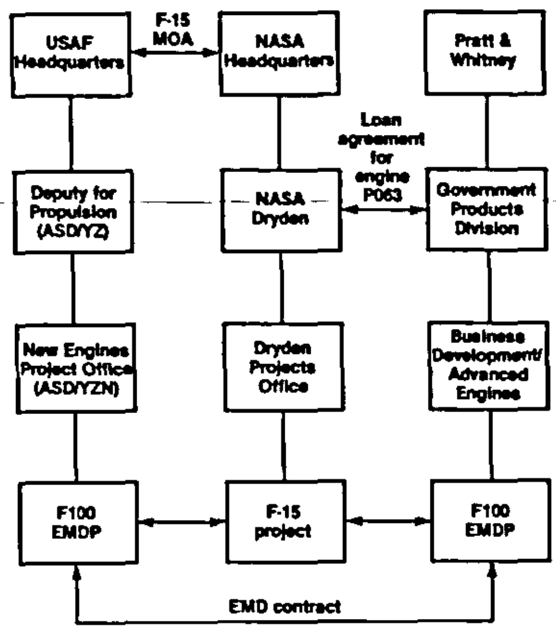

Fig. 3 DEEC flight program management.

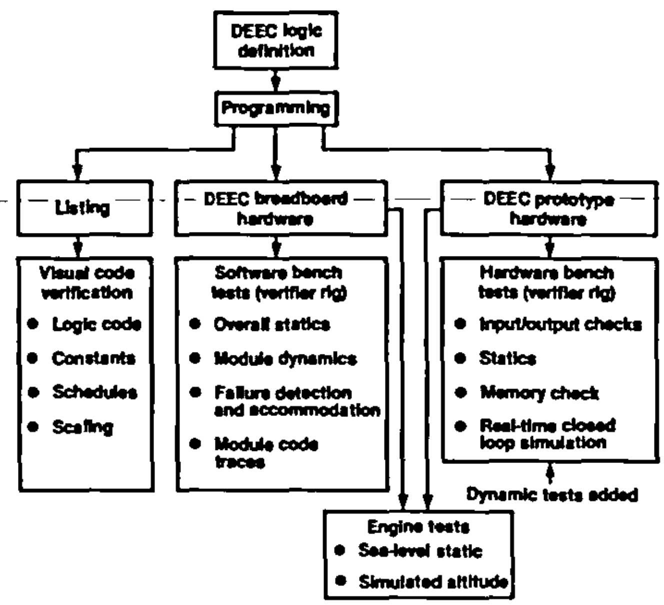

Fig. 4 DEEC software verification and validation tests.

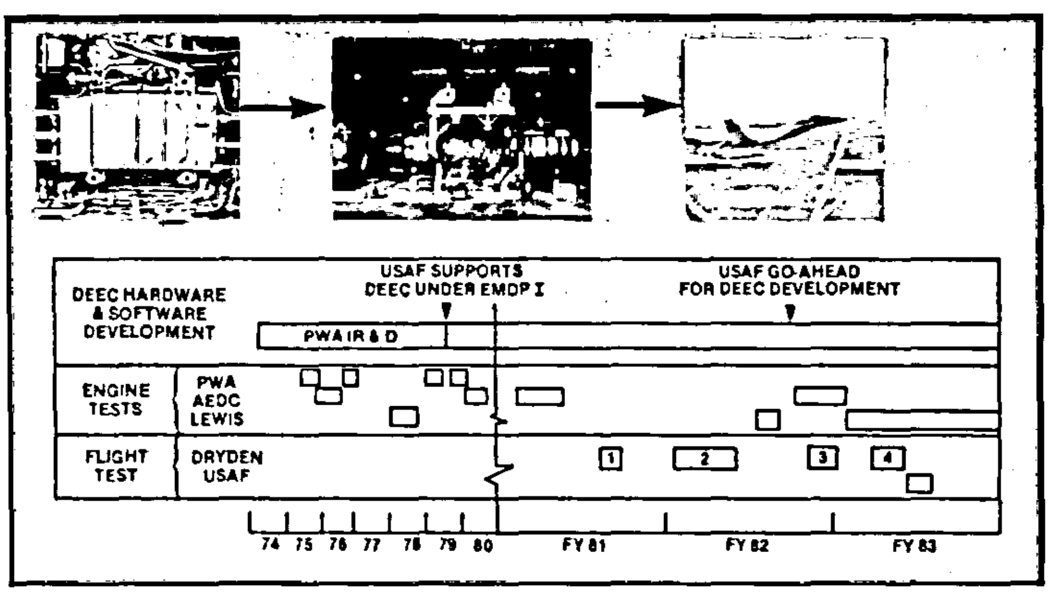

Fig. 5 DEEC ground and flight test elements.

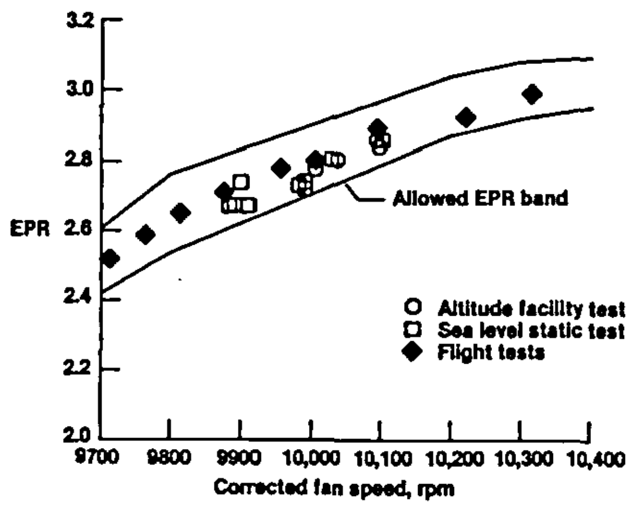

Fig. 6 Test results for DEEC no-trim feature.

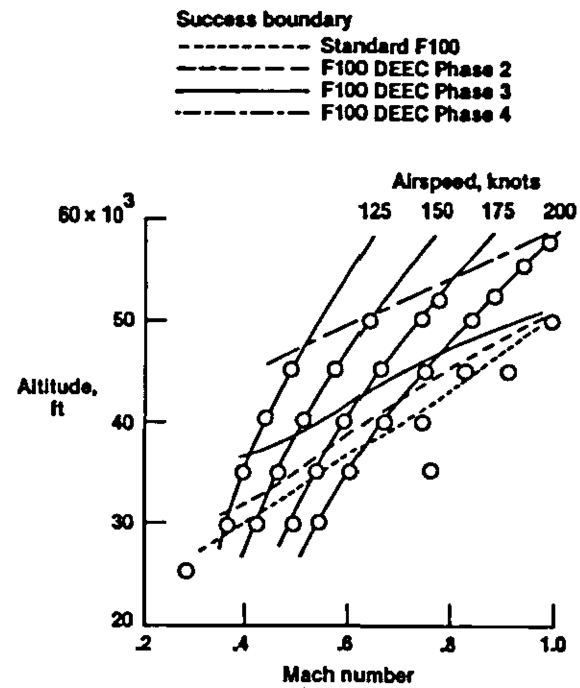

Fig. 7 Test results for idle-to-maximum power throttle transients.

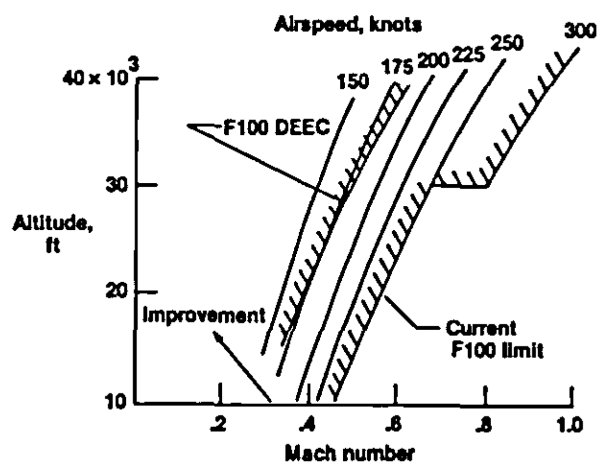

Fig. 8 Results of DEEC air-start tests.

767 FLIGHT TEST PROGRAM OVERVIEW

by
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ABSTRACT

The Boeing Model 767 certification flight test program which began in September 1981, was completed 10 calendar months later with the issuance of the FAA type certificate on July 30, 1982. This paper constitutes an overview of the 767 certification program presenting program statistics which show estimates, actuals, and comparisons to other Boeing flight test programs. The flight test instrumentation system, particularly the onboard system is discussed to outline the reliability, utilization, and capabilities which contributed greatly to the 767 program success.

For the most part the 767 flight test program went as planned and scheduled and any discussions of the normal day to day regime and the routine performance of required certification demonstrations would be uninteresting to even the most avid historical buff. Discussions of the testing therefore, include problem identification and resolution, configuration determination, and significant events unique to the 767 program. A certification summary is included to highlight significant requirements, especially those criteria changes from previous commercial transport. Finally a brief overview of the production status and derivative aircraft is presented to indicate the future of the Boeing 767.

INTRODUCTION

The Boeing 767 is a new technology two engine commercial jet transport. The 767 has a design range of between 2900 and 3475 nautical miles, carries 211 passengers in a mixed class two aisle configuration (up to 289 passengers can be carried in a 30 inch pitch configuration), and is powered by either the P&WA JT9D-7R4 or GE CF6-80A engines each developing 48,000 pounds SLST.

Flight Test was presented with its challenge of the 1980's, "to perform the unprecedented simultaneous certification test programs of the 767 and 757." The organization, manpower, and facilities required to perform this task were discussed by James Lincoln at the 1982 SFTE symposium and will not be addressed here. This paper is concerned with only the 767 test program except for the instrumentation system discussion. Both the 767 and 757 utilized the same equipment even though the physical arrangement was quite different due to airplane body dimensions.

767 PROGRAM STATISTICS

The program statistics included herein, are intended to present a broad brush overview of the 767 flight test program. Figure 1 provides individual airplane schedules and depicts the types of testing assigned to each of the 767 test airplanes. Figures 2 and 3 show the program estimates and actuals for flight hours, flow time, flight rates, and instrumentation requirements for the 767 Certification Flight Test Program. Flight hour comparisons of Boeing development, FAA certification, and non-test flight hours with the recently completed 757 program and all other model Boeing commercial jet transports is presented in Figure 4.

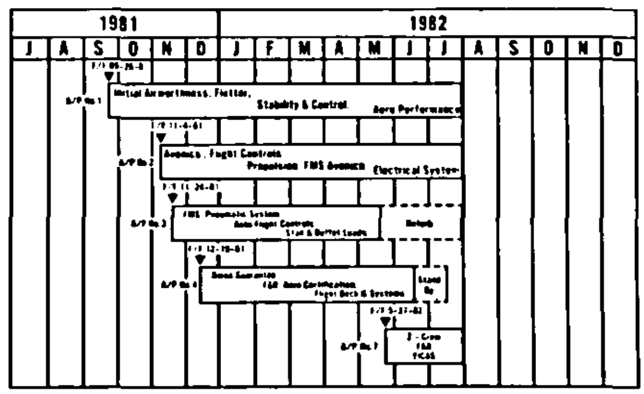

Figure 1 767 Flight Test Program

	VA001	VA002	VA003	VA004	TOTAL
Flight Hours	445	436	350	332	1563
Airplane Months	10	9	7 1/4	7 1/4	33.5
Flight Rate	44.5	48.4	48.3	45.9	46.7
Instrumentation Requirements	1679	2969	2535	601	7784

Figure 2 767 Flight Test Estimate

	VA001	VA002	VA003	VA004	VA006	TOTAL
Flight Hours (Total)	482	507	241	375	189	1794
Boeing	318	302	109	89	14	832
Certification	89	188	98	155	153	683
FCF & Ferry	18	3	11	16	2	50
Flight Hours (Technical)	425	493	218	260	169	1565
Flight Hours (Non-Technical)	57	14	23	115	20	229
Instrumentation Requirements	1918	3218	3175	960	137	9408
Flight Rate (Hours/Month)	48	56	42	52	94	53

Figure 3 767 Flight Test Actuals

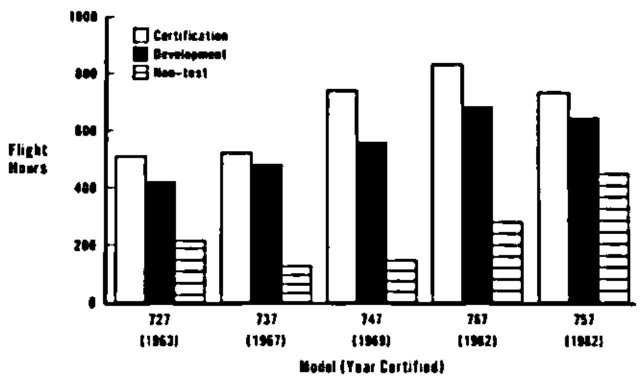

Figure 4 New Model Certification Programs

FLIGHT TEST INSTRUMENTATION

Initial Flight Test plans provided the 767 test program with the flexibility to perform high risk tests such as refused takeoffs, landing performance, brake system tuning, autopilot/flight director, flight management system, and thrust management system, on two or more of the test airplanes. This of course necessitated that the appropriate instrumentation be installed on primary and secondary airplanes. One of the cost savings initiated by engineering management was a scrub of the flight test instrumentation. The result of this scrub was reduced test program flexibility; the capability to move tests from one airplane to another was greatly reduced. Unfortunately the maximum cost savings possible from this exercise was not realized. The timing of the instrumentation scrub was late enough in the program that the engineering was well along, the purchased equipment was already on order (most had been delivered), and installations had been made. Specifically, the results of the instrumentation scrub negated the ability of accomplishing following testing on the airplanes identified:

Airplane 1 - Brake System, Antiskid System, RTO Performance, Landing Performance.
Airplane 3 - Brake System, Antiskid System, RTO Performance, FMS, TMS, AFDS.

Not to say "I told you so," but as test requirements were finalized it became apparent that tests had to be redistrbuted to keep estimated flight rates at a reasonable level. Following reassignment of tests, Airplane 3 received the full brake instrumentation package as well as approximately 1100 digital channels to support RTO performance, antiskid performance, FMS, TMS, and AFDS testing. The excellent reaction to the additional instrumentation requirements by the Flight Test Instrumentation groups and the Flight Test Manufacturing groups who installed the equipment and checked it out allowed testing to proceed on the original test schedules, meeting configuration freeze dates and handbook data requirements dates.

ONBOARD DATA SYSTEM (See Figure 5)

Any discussion of the 767 flight test program has to include a section concerning the Onboard Data System. Tom Smidt will present a paper dedicated to the Onboard Data System during the Thursday afternoon session of the symposium, but I want to also highlight the system and its contributions to the program.

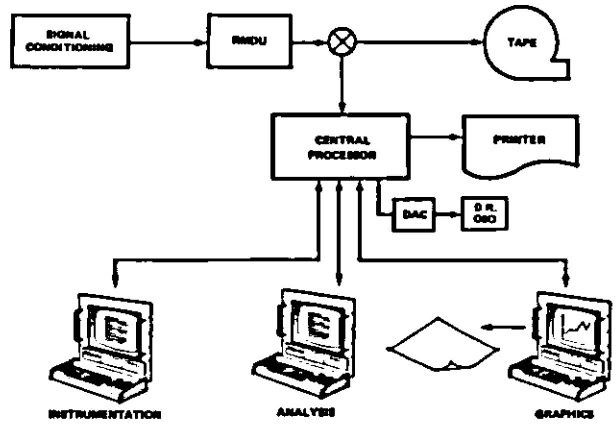

Figure 5 Onboard Data System

First of all the reliability of the current onboard system is unparalleled in Boeing history. The 767 Flight Test Program experienced very few flight delays of one hour or more and a total of only 3 test cancellations due to onboard data system problems during a total of 481 airplane flight days, for a reliability of 99.6 percent.

The Onboard Data System consists of the sensors, signal conditioning, Airborne Data Analysis and Monitor System (ADAMS II), and the recording system. ADAMS II, which is an airborne computing system, has been improved to provide better computer efficiency and additional capabilities. Improvements include new and refined analysis programs, graphics capability, and the ability to more efficiently preflight up to 2,000 measurements. ADAMS II utilizes a number of display methods to provide the test crew with the desired data. CRT displays can display up to 20 parameters with an update rate of once per second. ADAMS II can drive up to 10 digital displays located throughout the airplane. Each display can be programmed to show two measurements (one at a time) at an update rate of five times per second. Hard copies can be printed of 180 para-

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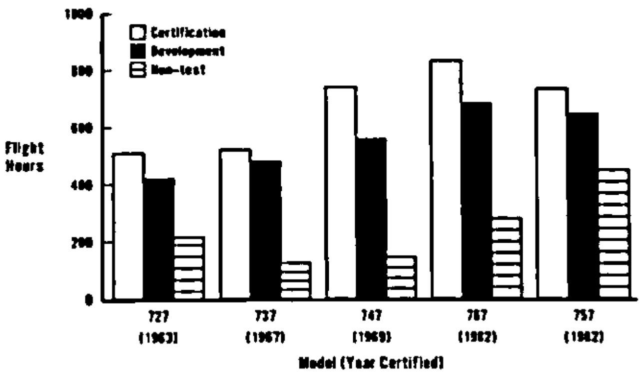

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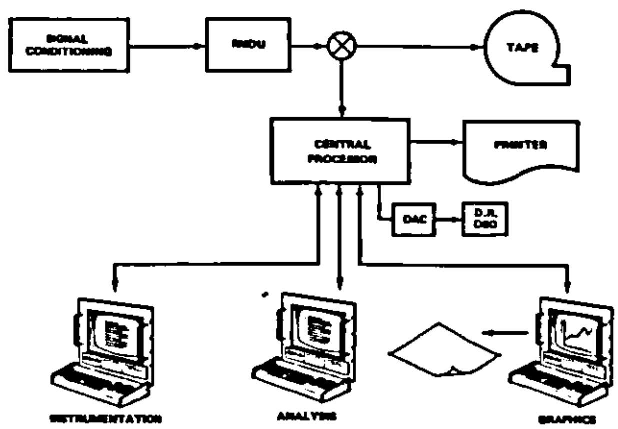

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The graphics system can provide either time histories or cross plots of stored data. Time histories can be displayed for up to six parameters on six y-axes plotted versus time. Up to six parameters can be separately cross plotted versus one horizontal axis. Two eight channel Brush recorders, one 16 channel oscillograph, and analog cockpit meters round out the means of providing the test crew with the information required for them to perform as an effective and efficient test team. All display media can display any of the parameters recorded on the flight tape whether raw data or a computed value.

Structural design studies indicated the need for establishing limit horizontal tail unsymmetrical loads and the requirement to monitor these loads during stall and increased load factor conditions which penetrate into buffet. The large number of tests which require test airplanes to enter buffet placed a manpower crunch on the structures analysis group as one of them was required to monitor tail loads on up to three test airplanes per day. To alleviate the requirement of the human tail loads monitor a "Black Box" was developed to provide visual indication when certain limit load percentage values were reached. This display after thorough checkout was mounted in the cockpit to alert the flight crew of reaching 70, 80, 90, or 100 percent of limit load. Seventy and 80 percent readings were advisory; whereas, the 90 percent indications were cautionary, i.e., discontinue that type of testing; and the 100 percent indication was an alert to discontinue all testing, return to base, review data, and if data confirmed that limit load had been exceeded then inspect the horizontal tail. The tail monitor system proved to be extremely reliable and certainly eased the structures analysis group workload to an acceptable level. By the way, horizontal tail limit loads were not reached during either development or FAA certification demonstrations.

Flight Test Instrumentation engineers developed a micro-processor which became an efficient Gross Weight/Center of Gravity (GW/CG) computer for use onboard the test airplanes. This system not only provided real time GW/CG information to the test crew, but by recording the GW and CG directly on the data tape, GW/CG data reduction time in the final data system was reduced. Initialization of the GW/CG computer requires loading the zero fuel weight, fuel tank quantities, fuel temperature and fuel density; other data required to make the system operate (fuel quantity burned, water ballast configuration, and fuel feed information), are input to the GW/CG processor via the onboard data system.

In addition to the two obvious advantages of real time display and final data system savings, gross weight is utilized in numerous calculations both onboard and in the final data system. Referred gross weight (W/δ) is used to normalize both airplane performance and certain airplane stability and control or handling qualities evaluations. W/δ is computed by ADAMS II and can be displayed in real time to aid the pilot in performing the required test. Data results of Coefficient of Lift (C_L), Nautical Air Miles Per Pound (NAMPP), Thrust to Weight (T/W), Standardized Fuel Flow (SSWF) are all computations which utilize gross weight and having a continuously updated value for the computer to work with allows for near real time onboard review of test results.

The water ballast system which was installed on each of the first three 767 airplanes could hold a total of 45,000 pounds of water and when required could allow center of gravity excursions from the forward limit 11% MAC to the aft limit - 36% MAC or visa versa, during a typical test flight. Water transfer time for full CG travel is approximately 7 minutes.

The 767 wide bodied airplane allowed design of a very efficient, roomy Instrumentation and Analysis area (see Figure 6). This area had seating for two Instrumentation Engineers, a Weights Engineer and six Analysis and/or Staff Engineers. This area housed the ADAMS II stations (2), the graphics terminal, two 8 channel brush recorders, the printer, and the GW/CG processor and water ballast control panel. The biggest advantage of this layout was the close proximity of the people and the direct communications that it allowed and encouraged.

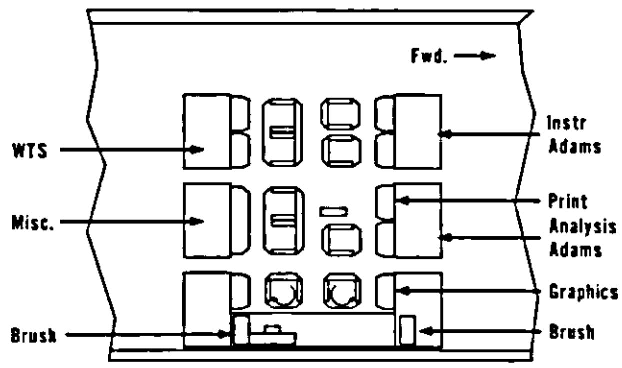

Figure 6 Instrumentation/Analysis Area

Each test airplane was equipped with dual interphone systems to provide test crew communications from one station to another. One interphone channel was designated as primary and was monitored by all test personnel including the pilot. The second interphone was utilized for ancillary discussions which did not require all test crew cognizance. All test crew stations are provided with the capability for communications on both interphone channels as well as all VHF or HF radios.

A video system was utilized for both onboard monitoring and post test review of video tapes during the configuration determination and slat development testing. The onboard monitors available in the cockpit and analysis areas allowed the test crew to observe behavior of the slats or tufts during flight maneuvers. Post test review allowed for stop action, slow motion, and multiple looks at the selected phenomena. Cockpit mounted video cameras were used for substantiation of crew work load in support of the 2-crew certification. Cockpit cameras trained on the various flight management system controls and displays provided the opportunity to monitor and record system performance and data entry for assessment of proper system operation. By having the video capability, observers were located in the main cabin rather than jamming the cockpit.

The data produced by the onboard data system allowed decisions to be made by the test crew on the airplane during flight rather than after data had been processed in the ground based data processing system. Onboard decisions were made during each flight concerning the health of the instrumentation system, goodness of the test condition, and test objective compliance. During the configuration development phase of the 767 program the onboard data was utilized to expedite the configuration determination process. If data showed the performance or characteristics of a certain configuration to be unacceptable; testing could be terminated prior to accomplishing a complete matrix of test conditions, thus saving costly flight time.

767 FLIGHT TEST PROGRAM HIGHLIGHTS

The 767 flight test program began as a four airplane success oriented program with forecast flight rates and instrumentation packages exceeding any major commercial airplane program in history (refer to Figure 3). In addition to the 767 program Boeing was also committed to a similar effort for the 757 which followed the 767 by approximately five months.

The President's Task Force on Aircraft Crew Compliance concluded that a 2-crew aircraft could be safely operated in the U.S. airspace environment, provided that proper design and FAA certification flights were accomplished. This report triggered a chain reaction of customer changes to 2-crew airplanes in lieu of 3-crew airplanes. In fact only one Boeing 767 customer will take delivery of 3-crew airplanes. At the time of the 2-crew decision 29 airplanes had reached the point of no return and either had been built or were being built as 3-crew airplanes. Boeing committed to certify the 2-crew configuration and maintain established delivery schedules with 2-crew airplanes.

The first four airplanes (those allocated to the test fleet) were tested in the 3-crew configuration while the seventh airplane off the line was the first airplane modified to the 2-crew configuration and joined the test fleet for 2-crew workload certification demonstrations and testing which required a fully equipped production airplane such as galley oven and chiller performance, cabin temperature and airflow survey, smoke evacuation and penetration, and passenger accommodations. Airplane number seven joined the test fleet on May 27, 1982, just two months prior to certification.

Another perturbation to the planned 767 Flight Test program was the deletion of the flight load survey. This was made possible because of conservative analysis and the results of static tests. One result of the flight load survey cancellation was that five airplane weeks were available for other flight testing. With this additional time available the decision was made to relieve the very high flight rate scheduled for the second airplane and off load some of the AFDS and TMS development testing to airplane three. Additionally RTO performance was rescheduled to Airplane 3 from Airplane 2.

The first flight of the 767 came on September 26, 1981, just one day after taxi tests were accomplished at speeds up to 120 knots. The maiden flight began at 1154 with the new red, white and blue 767 climbing to the north from Paine Field in Everett, Washington. Testing to evaluate handling characteristics and predicted initial buffet speeds at each flap position progressed in textbook fashion until the landing gear were retracted. During nose gear retraction the steering actuator dust cover interferred with and ruptured the center hydraulic nose gear retract return line, resulting in loss of the center hydraulic system fluid. This dust cover had been previously removed by rejection tag due to interference during gear swing testing, but was reinstalled by mistake when proper disposition of the tag was not accomplished in a timely manner. By the way, this was the first ever attempt to raise the landing gear on the first flight of a new model Boeing airplane. The next attempt made on the 757's first flight was a success.

Worse than usual Pacific Northwest fall weather moved in and kept the 767 on the ground for the next 10 days, except for one flight which was cut short due to deteriorating weather. The third flight provided the most excitement experienced during the entire program. During slat retraction the right side slats failed, apparently due to oscillatory loads acting on the slat linkage. See Figure 7 for a pictorial description of the slat failure.

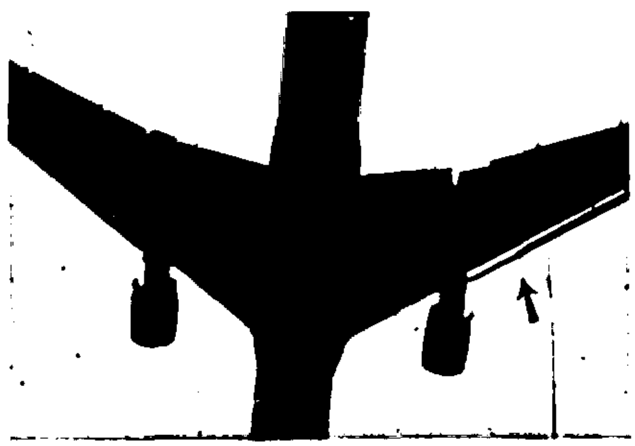

Figure 7 Inflight Slat Damage

Of course the 767 Flight Test Program now had a new top priority. The next several months were required to design, fabricate, install, instrument and evaluate slat improvements which strengthened the main tracks, revised the main track attach point, revised auxiliary track slat programming, revised slat rigging,, and added a fail safe stop to prevent the slats from rotating nose down in the event of auxiliary track failure (See Figures 8 and 9).

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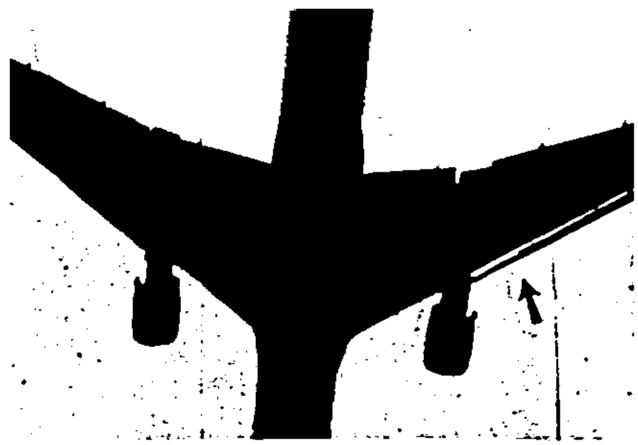

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In order to prevent unacceptable program delays while the problem was evaluated and a new configuration was designed, the slats were locked in the retracted position allowing high speed configuration and flutter testing to be accomplished. The top priority phase of the slat development testing was to define an acceptable takeoff slat configuration so that V_{mu} and takeoff performance testing could be accomplished, and maintain test schedule integrity. In fact the V_{mu} and initial takeoff performance testing was accomplished with the slats locked in the defined takeoff position. Including the final production slat configuration, 30 configurations of takeoff and landing slats were evaluated.

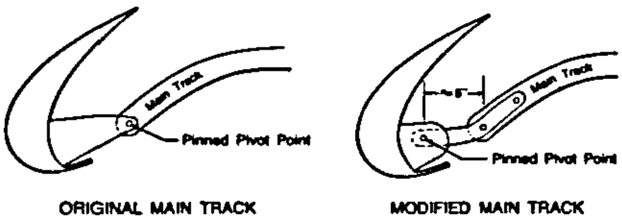

Figure 8 Model 767 Slat Track Geometry

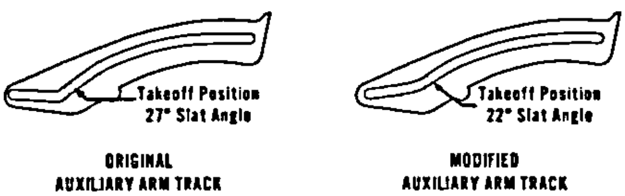

Figure 9 Model 767 Auxiliary Track Geometry

Typical tests of each slat configuration included slat loads at load factors from 0 to 2 g's and at speeds out to flap/slat design speeds (V_F); stalls were accomplished for evaluation of speeds and characteristics and low speed drag was assessed. During slat loads evaluations, in addition to utilizing the ADAMS II system for onboard data, two tracks of PCM data were telemetered for real time evaluation in the Telemetry Analysis Station (TAS). The TAS had four brush recorders that could display eight parameters each, thus providing a total of 40 (32 in the TAS and 8 onboard the airplane) parameters of analog data which could be reviewed in real time. Data obtained during each test condition were reviewed prior to clearing the airplane to increased speed or load factor.

Structural dynamic damping testing to prove the 767 was flutter free was completed in only 6 flights and 15 flight hours, compared to 184 flight hours required for 747 certification.

With the flutter clearance out of the way and the undetermined slat configuration preventing low speed configuration development testing, the program launched into high speed configuration development testing.

One of the few areas of concern going into the 767 test program was longitudinal stability at increased load factors at high Mach numbers. A Pitch Augmentation and Control System (PACS) was designed and installed on each of the test airplanes to correct the anticipated pitch up tendency. PACS philosophy was to provide sufficient nose down elevator to alleviate the pitch up thus providing positive maneuvering longitudinal stability. Testing confirmed that the basic airplane did have the pitch up as expected and that in order for the airplane to meet FAR certification criteria, a means had to be devised to correct this pitch up either aerodynamically or synthetically.

Boeing's bag of tricks to cure maneuvering longitudinal stability anomalies for the 767, contained three magical devices. As mentioned earlier, there was the PACS which was available for initial testing. Ready and waiting in the stores for installation were body vanes. The vanes were to be mounted on the fuselage just forward of the wing root and would be programmed to provide nose down pitching moment in order to maintain a positive stick force gradient. Imagine a canard on an otherwise conventional airplane .... The third and least costly of all the pitch up cures was Boeing's old friend, the vortex generator.

High speed configuration development testing consisted primarily of evaluations of maneuvering longitudinal characteristics and buffet boundary with each of eight vortex generator configurations. Vortex generator patterns varied from as many as 36 of the 3" long x 1½" wide x 3/4" high beauties per wing to the final configuration of seven per wing. The seven VG's are separated into a group of four located at approximately 35% semi-span with the other three located at approximately 60% semi-span. These vortex generators improved the maneuvering longitudinal stability of the 767 to certificable standards, and the PACS and body vane were put on the shelf to stay.

Flaps up stall characteristics exhibited a mild stick lightening prior to stall "g" break or deterrent buffet. This characteristic even though safe and controllable was not certificable and required corrective action. A synthetic stick force ramp activated upon reaching a certain angle of attack was input through the elevator feel system to overcome the unacceptable characteristic. All flaps/slats extended configurations exhibited acceptable stall characteristics.

Like all of its Boeing commercial jet transport predecessors the 767 rolled out with an operable Mach/Speed trim system. Estimates of the 767's static longitudinal stability, either in cruise or climb, concluded that a Mach/Speed trim system would be required. Boeing development testing demonstrated that static longitudinal stability was satisfactory, meeting the applicable FAR's without a Mach/Speed trim system installed. Thus, another system was sent to the bone yard.

The 767 is the first Boeing commercial jet transport equipped with a fully integrated digital Flight Management System (FMS). The FMS is common between the 767 and 757 airplanes and contributes greatly to the common cockpit type rating. The FMS includes the following major subsystems: Flight Management Computer, Autopilot/Flight Director, Thrust Management, Inertial Reference, Air Data, Navigational Sensors, Electronic Flight Instruments, and Caution and Warning. These integrated systems while providing performance optimization, flight management capabilities, and reduced airline operating costs; also increased test requirements due to system complexity, increased system capabilities, and system interaction. Four hundred hours of development and certification hours were required for certification of the FMS on the 767 compared to 200 hours for similar systems on the 747. Significant certification items include: automatic landings to Category IIIb criteria (zero decision height, 150 feet RVR and rollout to 60 knots) and vertical and lateral navigation with the capability of coupling with the autopilot to allow hands off flying from shortly after takeoff to 60 knots during landing rollout. It should be noted that FAA has not yet approved automatic landings to Category IIIb minimums even though Boeing feels that all required testing has been accomplished satisfactorily.

Those of you familiar with the 747 test program probably recall the development problems encountered with the new high bypass ratio JT9D engines. Prior to the 767 test program both engine manufacturers (P&WA and GE) contracted with Boeing to install and test their respective 767 engines on the Boeing owned 747 to provide early confidence in engine performance and operating characteristics. These tests and improved engine technology certainly paid off as the entire basic certification program (P&WA JT9D-7R4 engines) was accomplished without an engine change.

CONFIGURATION DEVELOPMENT SUMMARY

- o PACS is not required.
- o Body Vane is not required.
- o Mach Speed Trim system is not required.
- o 7 Vortex Generators per Wing Installed.
- o Stick Nudger for flaps1 stall characteristics is installed.
- o 2 position slats (T.O. and landing) are installed.

767 CERTIFICATION SUMMARY

The 767 was certified on schedule with the type certificate issued on July 31, 1982. Initially the airplane was approved to a gross weight of 300,000 pounds MTOGW and 270,000 pounds MLGW, even though testing was accomplished to obtain certification data for future certification to 335,000 pounds MTOGW. Engine and airplane performance testing was conducted to thrust levels of 50,000 pounds SLST by overboosting the 48,000 pounds SLST engines currently being delivered. Testing has been accomplished since the end of July to certify those two major items which remained open for basic certification; natural icing handling characteristics and recognition, and FMS vertical navigation.

Referring back to Figure 4 you can see the significant increase in certification flight hours from model to model, beginning with the 727 in 1963. FAA demonstration flight hours required to obtain the 767 type certificate are 163% of those required for the 727 and 122% of those required to certify the 747 just 13 years earlier in 1969. Interestingly, when you consider the 200 flight hour increase over the 747 for certifying the FMS or equivalent systems then the remaining certification testing required approximately 100 less hours than the 747, despite the large number of rules changes, more stringent rule interpretation, and compliance with FAA Issue Papers. Some of the more significant new or revised test requirements were:

1. Stall characteristics and stall warning during accelerated stall and maneuvering low speed flight (addressed by Issue Paper F-3).
2. Handling qualities with natural ice built up on the airplane, including evaluations of landing flap configurations and flap extensions after ice accretion.
3. Maneuvering stability, restatement of the rule resulted in a more thorough exploration of the airplane flight envelope.
4. TIA requirement to perform fuel cut takeoffs in addition to throttle cut takeoffs for performance.

PRODUCTION STATUS

As of August 1, 1983, 61 airplanes have been delivered to 11 customers. Current orders are for a total of 174 airplanes with five remaining customer introductions.

Two new derivatives of the 767, the 767-ER and the 767-300, have been announced. The 767-ER has increased fuel capacity for extended range to 4880 nautical miles (767-200 is currently advertised as a 3790 N.M. range airplane) and an increased gross weight to 345,000 pounds. Ethiopian Airlines is the first 767-ER customer with delivery scheduled for the first of two airplanes in May, 1984. The 767-300 is a stretched version of the 767-200 with two 110 inch plugs (donuts) added to sections 43 and 46 (both forward and aft of the wing). The -300 will also be a 345,000 pounds MTOGW airplane. Depending on receipt of airline orders for this version, the first delivery could be made in early 1987.

subsystems: Flight Management Computer, Autopilot/Flight Director, Thrust Management, Inertial Reference, Air Data, Navigational Sensors, Electronic Flight Instruments, and Caution and Warning. These integrated systems while providing performance optimization, flight management capabilities, and reduced airline operating costs; also increased test requirements due to system complexity, increased system capabilities, and system interaction. Four hundred hours of development and certification hours were required for certification of the FMS on the 767 compared to 200 hours for similar systems on the 747. Significant certification items include: automatic landings to Category IIIb criteria (zero decision height, 150 feet RVR and rollout to 60 knots) and vertical and lateral navigation with the capability of coupling with the autopilot to allow hands off flying from shortly after takeoff to 60 knots during landing rollout. It should be noted that FAA has not yet approved automatic landings to Category IIIb minimums even though Boeing feels that all required testing has been accomplished satisfactorily.

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TEN YEARS OF COMBINED TEST OPERATIONS AT THE AIR FORCE FLIGHT TEST CENTER (AFFTC) - IN RETROSPECT

By
Peter D. Kennedy
Air Force Flight Test Center, Edwards Air Force Base, California

PAPER NOT AVAILABLE AT TIME OF PRINTING

THE AUTOMATED KC-135R TEST PROGRAM

by
D. A. Fletcher*
J. T. Higgs**
Boeing Military Airplane Company
Wichita, Kansas

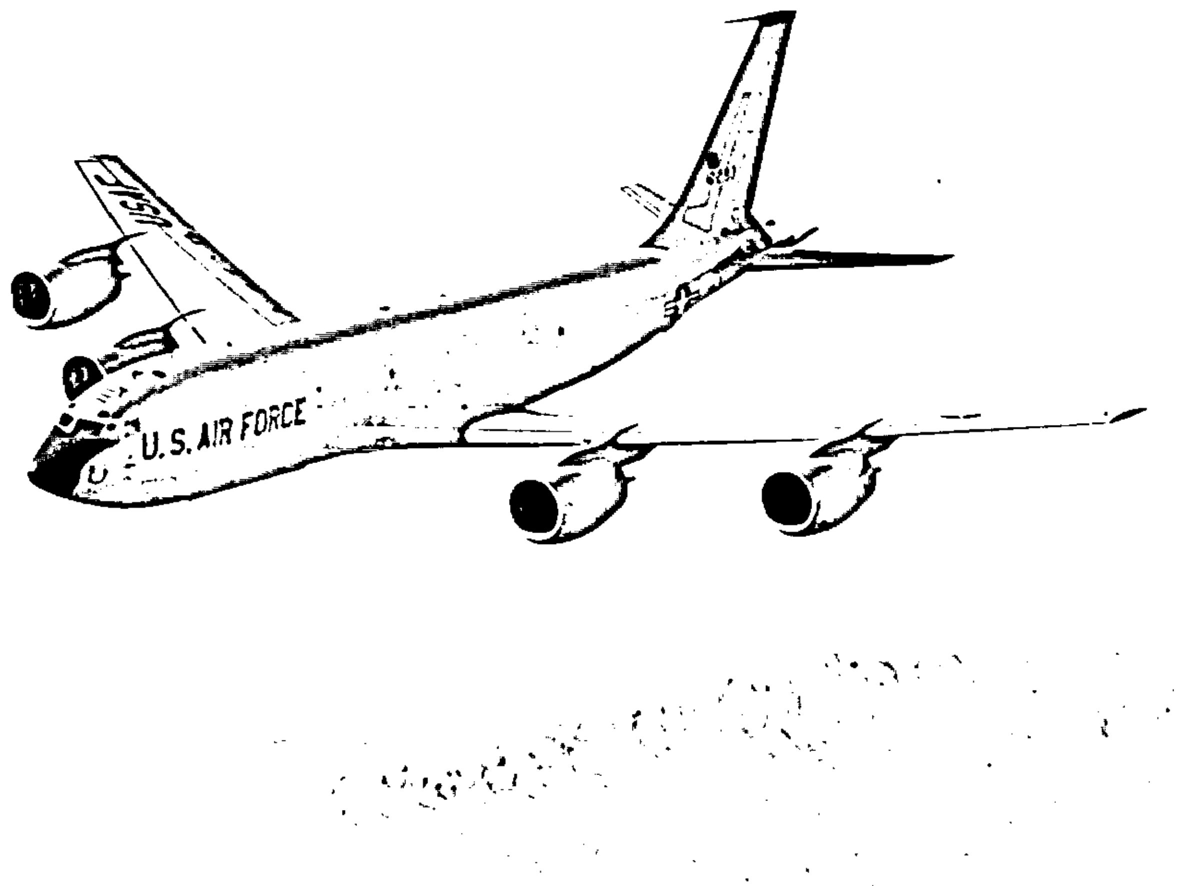


Fig. 1 The KC-135R

Abstract

The KC-135R Flight Test program required approximately 750 discrete instrumentation parameters, with over 50 different types of testing and 1748 discrete test conditions. The task of scheduling, dispositioning, and ordering data for the 1748 conditions presented a formidable obstacle in light of available resources and the contractual performance period of 131 days. In order to minimize manpower requirements and maximize flexibility, while remaining within strictly controlled limits, a computerized system of flight test planning, tracking, documenting, and data requesting was developed. The use of a master data base of test instrumentation and test conditions, accessed through interactive terminals and printers, provided a realtime decision making capability, minimizing errors and redundancy. This paper describes the evolution, capabilities, advantages and shortcomings, along with some recommended future enhancements, of this computerized flight test system.

*Deputy Test Director
**Lead Flight Test Operations Engineer

Background

The KC-135 Stratotanker (Figure 1) was originally built by the Boeing Company over twenty years ago. Since the original production, the advancement in engines and propulsion technology has been rapid and directed toward increasing thrust and decreasing fuel consumption and noise levels. In 1978, the Boeing Military Airplane Company (BMAC) in Wichita, Kansas, proposed to the Air Force a re-engining program to extend the useful life of the KC-135 fleet well into the 21st century. The engine selected for this re-engining program was the CFM-56-2B-1 (Figure 2), manufactured by CFMI, a consortium of General Electric Aircraft Engine Group of Evendale, Ohio, and SNECMA of Villa roche, France. The engine is a high bypass ratio turbofan weighing only 380 pounds more per engine than the J-57 engine it replaces; it produces 9,000 pounds more thrust per engine, while consuming approximately 25 percent less fuel and reducing the 90 EPNdB footprint by 98 percent.

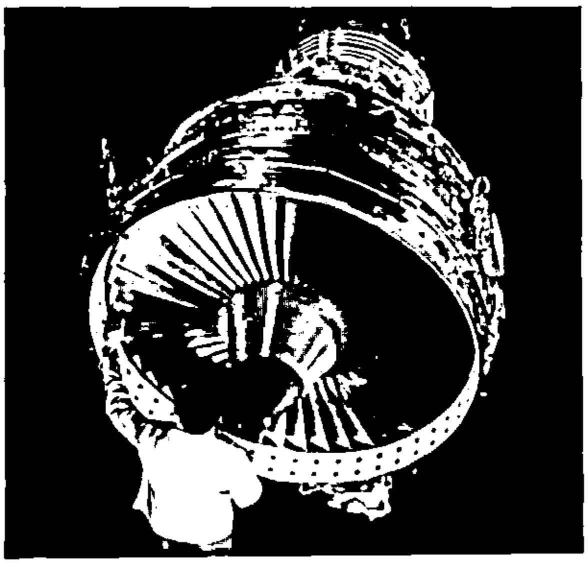


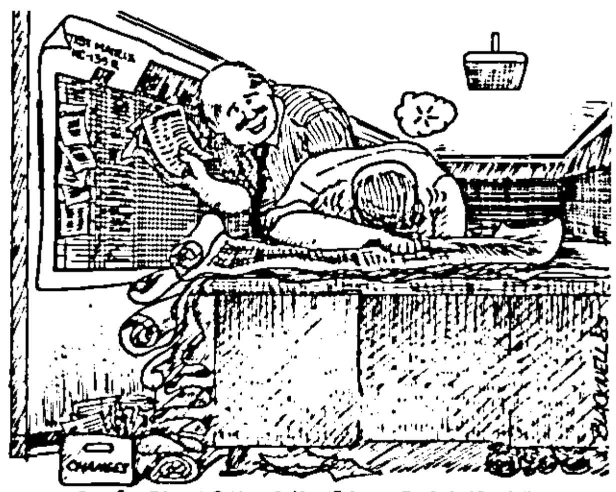
Fig. 2 CFM-56-2B-1

The first unit installation of the propulsion systems was accomplished during the spring of 1982. In addition to replacement of the engines, the modification included an increased horizontal stabilizer area, the deletion of water injection for increased takeoff thrust, and the addition of a Flight Control Augmentation System (FCAS), Series Yaw Damper (SYD), and rudder actuator improvements. Hydraulic, electrical and environmental control systems were also modified. The anti-skid and brake systems were altered, and a fuel temperature indicating system was added. The re-engining program necessitated a ground and flight test effort of sufficient magnitude to quantify the effects of the added and modified systems. This report deals with the Developmental Test and Evaluation/Operational Test and Evaluation (DT&E/OT&E) segment of this testing.

Testing was accomplished by a Combined Test Team (CTT), composed of test and engineering personnel from BMAC, the Air Force Flight Test Center (AFFTC) from Edwards AFB, CA, the Air Force Operational Test and Evaluation Center (AFOTEC) from Kirtland AFB, NM, Aeronautical Systems Division (ASD) from Wright-Patterson AFB, OH, and the Strategic Air Command (SAC) from Offutt AFB, NE. Representatives from CFMI, Northrop Electronics Division (for Turbine Engine Monitoring System testing) and Sundstrand participated in the system and subsystem testing at Edwards AFB.

Evolution

The size and scope of the ground and flight tests planned for the new KC-135R quickly identified the need for over 750 different test instrumentation parameters and over 1700 distinct flight test conditions in the test planning phase. The concept of using a computerized method to store and sort test information was recognized early in the test definition phase of planning. Due to projected costs involved to develop the system, the proposed computer system was shelved at first, with traditional matrix methods being substituted as the time-proven method for tracking test requirements and accomplishments. With 75 categories of Test Items (TIs) and 750 parameters, a wall chart matrix using one inch squares was estimated to be approximately eight feet tall and 90 feet long (Figure 3). The difficulties foreseen in constructing and maintaining this size matrix using a small number of test personnel at a remote location on a daily basis were immediately obvious. The computerized method of tracking was resurrected, and development began in the spring of 1982. Software development was assigned to software engineers from Boeing Computer Services (BCS) Company.



BILL?...REALLY, BILL, WE AREN'T GOING TO CHANGE YOUR TEST MATRIX ANY MORE....JUST THIS ONE TIME...O.K. BILL?....

Fig. 3 Manual Wall Chart Method

The system required simultaneous development of both hardware and software. The hardware choices immediately available included (1) the purchase of a mini-computer with hard disk and magnetic tape backup, (2) use of the AFFTC computer system at Edwards AFB, the location of most of the flight testing, or (3) the BMAC VAX computer facility in Wichita. Funding and contractual problems precluded the first two options, so the BMAC system was chosen for the development.

The BMAC system at Wichita consisted of a Digital Equipment Corp. VAX/VMS 11/780 CPU with 4-megabyte random access memory (RAM). Peripheral devices included two RK07 26-megabyte discs, one RM05

300-megabyte disc, three digital tape drive units, a D-900 (900 lines/min) printer, a Versatec plotter printer, standard interface for eight terminals, floppy disc interface, and a card reader. A VT131 terminal with a LA100 letter printer was located at the remote site and linked to the mainframe via a 1200 Baud modem.

The software developed for the test parameter tracking was a combination of FORTRAN 77 and assembly language, the choice being dependent on hardware compatibility for remote terminals and on speed of preparation. This latter requirement was a result of the rapid expansion of the original scope of the project.

What was originally intended to be a test instrumentation status tracking aid quickly ballooned to include almost every phase of the testing process. The software that was to sort the instrumentation operability status of each of the 750 parameters was developed just prior to the start of ground testing at BMAC-Wichita. While the ability to list the required instrumentation for each test was achieved, a much broader scope of software was becoming evident. If the instrumentation status could be tracked, then why not every test condition contained in the flight test program? The lack of time prevented inclusion of the ground test conditions, but there remained approximately three months to the scheduled first flight of the modified airplane. The go-ahead was authorized by the BMAC Test and Evaluation Manager, and the development of the test matrix data base was initiated; but the limits were not yet reached, as the BMAC test personnel added suggestions to the project. The data base was input according to test condition number, airplane weight, center of gravity location, altitude, airspeed, configuration and special notes, and the project expanded to include preparation of mission flight plans and postflight summaries, test condition completion status, data processing requests, and program visibility outputs. Each suggestion was evaluated and added to what was quickly becoming an automated test program for the KC-135R. The evolution of these segments was dictated primarily by the flight test phase in development.

Capabilities

At the completion of the KC-135R Flight Test Program at Edwards AFB, the automated test program consisted of the following segments:

- A test condition and status data base (Figure 4) with the ability for users to load or change the base after entering a limited access code.
- A data base manipulation software module which enabled test personnel to accomplish the following functions:
  - Construction of a mission profile (Figure 5), comprised of a list of test conditions and alternate profiles based on weather, facility restrictions, instrumentation, etc. The profile was reviewed by the Engineering Staff and aircrew for compatibility and feasibility.
  - Construction of a kneecard or detailed mission flight plan (Figure 6) to be used to brief and conduct the flight. The Test Conductor prepared the profile and mission kneecard from the test conditions listed in the data base through a search/sort process that could manipulate the conditions according to completion status, gross weight, altitude, etc., with up to four criteria able to be specified for the search. Hard copies were available for search/sort or the kneecard construction process. This kneecard segment was the key element to the automated process, as this segment had to be extremely "user friendly." It combined elements of engineering testing and text processing, since it required enumerating the selected test conditions, adding the text contained in the Detailed Test Information Sheets (DTISs), as well as other Test Conductor instructions such as disposition of the trailing static pressure cone. This kneecard construction segment was still in the software development iteration well into the flight test phase. Once constructed on the remote terminal CRT, hard copies were printed, copied and distributed at the preflight briefing. The CTT Test

KC135R DT&E/OT&E TEST CONDITIONS DATA BASE

TEST ITEM	CT	GW	ALT	K-M	CG	TY	E	GS	FS	FLT	RUN	PRI	COMMENT
1.06.001.015	3	323	0	0	0	TO	3	DN	30	1	0	/1/	3 ENGINE TRT T.O.
1.06.001.016	3	323	0	0	0	TO	3	DN	30	1	0	/1/	3 ENGINE 85% N1 T.O.
1.06.001.017	3	323	0	0	0	TO	3	DN	30	1	0	/1/	3 ENGINE 60% N1 T.O.
1.06.001.018	3	323	0	0	0	TO	3	DN	30	1	0	/1/	3 ENGINE 75% N1 T.O.
1.06.001.[illegible]	3	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	DN	[illegible]	1	0	/1/	MPGM FAIL [illegible]
1.06.001.													
1.06.001.													
1.06.001.													
1.06.001.													
1.06.001.													
1.06.001.													
1.06.001.													
1.06.001.													
1.09.001.													
1.09.001.													
1.11.001.													
1.11.001.													

KC135R DT&E/OT&E TEST CONDITIONS DATA BASE

TEST ITEM	CT	GW	ALT	K-M	CG	TY	E	GS	FS	FLT	RUN	PRI	COMMENT
1.01.001.001	10	0	300	.80	0	CR	4	UP	UP	1	0	/1/	PITOT STATIC SYS. CAL
1.01.001.002	10	0	300	.70	0	CR	4	UP	UP	1	0	/1/	PITOT STATIC SYS. CAL
1.01.001.003	10	0	300	.60	0	CR	4	UP	UP	1	0	/1/	PITOT STATIC SYS. CAL
1.01.001.004	10	0	100	0	0	CR	4	UP	UP	1	0	/1/	1.3VS PITOT SYS. CAL
1.01.001.005	10	0	100	250	0	CR	4	UP	UP	1	0	/1/	PITOT SYS. CAL
1.01.001.006	10	0	100	350	0	CR	4	UP	UP	1	0	/1/	PITOT SYS. CAL
1.01.001.007	10	0	100	0	0	CR	4	UP	30	1	0	/1/	1.6VS PITOT SYS. CAL
1.01.001.008	10	0	100	0	0	CR	4	UP	30	1	0	/1/	1.4VS PITOT SYS. CAL
1.01.001.009	10	0	100	0	0	CR	4	UP	30	1	0	/1/	1.2VS PITOT SYS. CAL
1.01.001.010	10	0	100	0	0	CR	4	DN	30	1	0	/1/	1.2VS PITOT SYS. CAL
1.01.001.011	10	0	100	0	0	CR	4	DN	50	1	0	/1/	1.4VS PITOT SYS. CAL
1.01.001.012	10	0	100	0	0	CR	4	DN	50	1	0	/1/	1.2VS PITOT SYS. CAL

Fig. 4 Sample Output - Test Condition Data Base

```
                    KC135R DT&E/OT&E TEST CONDITIONS DATA BASE
                                MISSION PROFILE

****** MISSION:     49   C5REFUELING                       TOTAL # OF RUNS: 18.0
                                                           CUMULATIVE TIME: 283

0                1   2   3   4   5  6  7 8  9  10  11  12  13   14
TEST ITEM        CT  GW  ALT K-M CG TY E GS FS FLT SEQ  P  RUN  COMMENT
---------------- --- --- --- --- -- -- - -- -- --- --- --- ---- ------------
2.23.001.001.00  15   0   0   0  0 ES 4 ON 00  11   1 /2/  1.0 MISC ENG START
2.23.002.001.00  15   0   0   0  0 ES 4 ON 00   6   2 /2/  2.0 ENG HEALTH MONITOR
2.24.001.001.00  30 265 250 255  0 AR 4 UP 00   0   3 /1/  3.0 C-5 RND2
2.24.001.002.00  10 265 250 255  0 AR 4 UP 00   0   4 /1/  4.0 C-5 MJDAR
2.24.001.003.00  10 265 250 255  0 AR 4 UP 00   0   5 /1/  5.0 C-5 MJDAR NO CONT
2.24.001.004.00  10 265 250 255  0 AR 4 UP 00   0   6 /1/  6.0 C-5 MJDAR WET CONT
2.24.001.005.00   5 265 250 255  0 AR 4 UP 00   0   7 /1/  7.0 C-5 DISC/PRE CONT
2.24.001.006.00  15   0 250 255  0 AR 4 UP 00   0   8 /1/  8.0 C-5 APR LSP TRIM
2.24.001.007.00  15   0 250 255  0 AR 4 UP 00   0   9 /1/  9.0 C-5 APR MSP TRIM
2.24.001.008.00  15   0 250 255  0 AR 4 UP 00   0  10 /1/ 10.0 FAST APR MSP-TRIM
2.24.001.009.00  15   0 250 255  0 AR 4 UP 00   0  11 /1/ 11.0 FAST APR LSP-TRIM
2.24.001.010.00  20   0 250 255  0 AR 4 UP 00   0  12 /1/ 12.0 C-5 M&LSP TRIM BKAWAY
```

Fig. 5 Sample Output - Mission Profile

```
          G E N E R A L   F L I G H T   I N F O R M A T I O N
                      F L I G H T   N U M B E R   M23

TAIL NUMBER     : 61-0293

TACTICAL CALL SIGN: RICK 52

FLIGHT DATE        : 14 DEC 19
LANDING TIME       : 23:10:00
TAKEOFF TIME       : 16:20:00
TOTAL TIME         :    6.5

TOTAL PROJECT TEST TIME          : 146.9

TEST FREQUENCIES  -  PRIMARY     : 349.6
                     SECONDARY   :
                     REFUELING   :

 AIRCRAFT CONFIGURATION
 ******** *************

 THE TEST AIRPLANE WILL BE A KC-135R CON
D455-40045-1, "CONFIGURATION DESCRIPTION
D458-40071-27, "INSTRUMENTATION INSTALLA
AF 61-0293, CLASS II MODIFICATION".  DET
INFORMATION WILL BE CONTAINED IN D3-1218
STATUS AND PLANNING DOCUMENT - AIRPLANE
RE-ENGINE FIRST UNIT PROGRAM".

 PURPOSE OF FLIGHT
 ******* ** ******

THE PURPOSE OF THIS FLIGHT IS TO COMPLET
FLIGHT TESTS, AND ACCOMPLISH THE LOW GRO
STALL TESTING, SELECTED ENGINE PERFORMAN
PRE-BOOSTER CHANGE TAKEOFF,CLIMB, AND BO
```

```
      FLIGHT NO.   : M23                                      PAGE   :     1
      FLIGHT DATE  : 14 DEC 19                                DATE   : 16-MAY-83
*******************************************************************************

RUN   1
      ACCOMPLISH A PREFLIGHT WEIGHING OF THE AIRPLANE.
      ACCOMPLISH ALL CHECKLIST ITEMS UP TO BUT NOT INCLUDING THE
      "START ENGINES" CHECKLIST.
      INSTRUMENTATION -- ON.
      ADAMS I         -- ON.
      RECORD              -- FUEL CONFIGURATION AND TIME.
      RECORD          -- START TIME AND MAX EGT FOR ALL ENGINES.
      INSURE THAT A TIME CODE SYNCRONIZATION WAS ACCOMPLISHED.

      TI COND #        GW  ALT K-M CG TY E GS FS COMMENT
      ---------------- --- --- --- -- -- - -- -- ---------------------
      2.23.001.001.00   0    0   0  0 ES 4 ON 00 MISC ENG START
      ENGINE START

      TAXI TO LAST CHANCE.

RUN   2
      A LAPSE RATE TAKEOFF IS NOT REQUIRED.
      ACCOMPLISH  A NORMAL TAKEOFF WITH A/C-ON, AND PMC-ON.

      TI COND #        GW  ALT K-M CG TY E GS FS COMMENT
      ---------------- --- --- --- -- -- - -- -- ---------------------
      2.04.001.001.00   0    0   0  0 TO 4 ON 30 59F,A/C ON,PMC ON
      ENGINE PERFORMANCE
```

Fig. 6 Sample Output - Mission Kneecard

Conductors became so facile with the machine that their preparation of the kneecards was often accomplished the morning of the pre-flight briefing. Changes to these kneecards were manually recorded on the Test Conductor's copy and filed in the test records to reflect planned deviations. The kneecard also contained a standard approval signature page, aircraft configuration, crew list, and weight and CG data, all of which simplified the test historical record tracking.

- Construction of a Request for Instrumentation Preflight (RIP) (Figure 7). This segment was based on the mission profile built previously and listed each Test Item to be addressed during the flight versus each test parameter, reflecting a "C" or "D" if the parameter was determined to be "critical" or "desired" for accomplishing the test.

- Preparation of a Brief Flight Summary (Figure 8). Although they were part of the postflight segment that follows, these summaries by the Test Conductor and Pilot in Command provided a quick synopsis of the flight results that were telefaxed to appropriate program managers across the country the morning following the flight.

- Preparation of a Full Flight Summary (Figure 9). Containing the Brief Flight Summary above, the remaining required information was input by the Test Conductor by means of a "prompt-response" software module. This enabled him to respond to computer prompts

LISTING DATE: 16-MAY-83 KC-135R TEST DATA MEASUREMENTS LIST/TEST MATRIX

ADAMSI NO	PARAMETER NAME	STATUS	1.06.001 AE-FT1 TIS 9.2. 1.1.1	1.15.001 AE-FT1 TIS 9.2. 1.1.1	2.02.002 PP-FT1 TIS 9.2. 1.8.2	2.04.001 PP-FT2 TIS 9.2. 1.8.3
5	A005	5		C		
7	A007	9	C	C		
26	B003	9	C	C	C	C
753	B003A	9	C	C	C	C
28	B007	9	C	C	C	C
754	B007A	9	C	C	C	C
29	B009	9			C	C
33	B013	5	C	C		
38	B019	9	C			
39	B020	9	C	C		
40	B022	9	C	C	C	C

Fig. 7 Sample Output - Request for Instrumentation Preflight (RIP)

```
             FLIGHT TEST MISSION SUMMARY  -  KC-135R
                 F L I G H T   N U M B E R  49

TAIL NUMBER    : 61-0293

TACTICAL CALL SIGN: RICK 74                         CHAS

FLIGHT DATE        : 15 MAR 83
LANDING TIME       : 00:16:34
TAKEOFF TIME       : 20:46:12
TOTAL TIME         :     3.6

TOTAL PROJECT TEST TIME             : 279.8

TEST FREQUENCIES  -  PRIMARY     : 358.4
                     SECONDARY   :
                     REFUELING   : 358.4

CREW
****

  PILOT IN COMMAND     :   GEBHARDT
  PILOT                :   JONES
  PILOT                :   ENGSTROM
  TEST CONDUCTOR       :   HIGGS
  TEST ENGINEER        :   BIROS
  TEST ENGINEER        :   CHAFFIN
  INSTR ENGINEER       :   BITZEGAIO
  INSTR ENGINEER       :   SEPPANEN
  BOOM OPERATOR        :   PARRISH
```

```
               FLIGHT TEST MISSION SUMMARY  -  KC-135R
                   F L I G H T   N U M B E R  49

 TEST CONDUCTOR SUMMARY
 **** ********* *******

A .295 LAPSE RATE TAKEOFF WAS MADE AND A CLIMB TO FL250 WAS ACCOMPLISHED
TO RENDEZVOUS WITH THE C-5 RECEIVER (KITE 02).  AFTER THE RENDEZVOUS
WAS ACCOMPLISHED ALL REFUELING CONDITIONS WERE SUCCESSFULLY ACCOMPLISHED.
THE AUTOPILOT HIGH SPEED TRIM WORKED QUITE WELL FOR ALL CONDITIONS.
AFTER REFUELING A DESCENT TO 10,000FT WAS ACCOMPLISHED AND THE PITOT
STATIC SYSTEM UTILIZING THE PILOTS PRODUCTION PROBE WAS CHECKED AGAINST
THE TRAILING CONE.  NO SIGNIFICANT DIFFERENCES WERE NOTED.  THE CONE
WAS RETRACTED AND FOUR APPROACHES WERE FLOWN AT REDUCED APPROACH
SPEEDS (-5KTS AND -10KTS).  THE -5KTS APPROACHES APPEARED TO BE THE BEST.
COPILOT AIRSPEED INDICATORS AVERAGED ABOUT 5 KTS LOWER IN THE TRAFFIC
PATTERN.  THE PILOTS SYSTEM WAS HIGHER THAN THE COPILOTS.

                         SIGNATURE :
                                    -----------------------------
                                    JOHN T. HIGGS BMAC


 PILOT SUMMARY
 ***** *******

ALL SCHEDULED CONDITIONS WERE ACCOMPLISHED EXCEPT FOR THE 30 DEG
FLAP LANDINGS.  THE C-5A INFLIGHT REFUELING CONDITIONS WERE SUC-
CESSFULLY COMPLETED, AND THE HIGH SPEED AUTOPILOT TRIM WORKED DURING
```

Fig. 8 Sample Output - Brief Flight Summary

```
                         R U N   S U M M A R Y
      FLIGHT NO.   : 49                            PAGE    :    1
      FLIGHT DATE  : 15 MAR 83                     DATE    : 17-MAY-83
*************************************************************************

 RUN   1      ENGINE START

     TI COND #          GW  ALT
     ---------------    --- ---
     2.23.001.001.00     0   0

     START SEQUENCE 3,4,2,1

     TI COMPLETION CODE
        ( C/P/N )
            C
            C
            C
            C

 RUN   2      ENGINE HEALTH MON
```

```
                      T E S T   I T E M   S U M M A R Y
          FLIGHT NO.   : 49                                PAGE    :    1
          FLIGHT DATE  : 15 MAR 83                         DATE    : 17-MAY-83
*******************************************************************************

TEST ITEM          RUN NUMBER  TI COMPLETION CODE  START TIME  STOP  TIME
                                   ( C/P/N )       (HH:MM:SS)  (HH:MM:SS)
1.01.001.025.00        21              C            23:20:45    23:21:45

TEST ITEM          RUN NUMBER  TI COMPLETION CODE  START TIME  STOP  TIME
                                   ( C/P/N )       (HH:MM:SS)  (HH:MM:SS)
1.01.001.026.00        22              C            23:19:21    23:20:13

TEST ITEM          RUN NUMBER  TI COMPLETION CODE  START TIME  STOP  TIME
                                   ( C/P/N )       (HH:MM:SS)  (HH:MM:SS)
1.01.001.027.00        23              C            23:17:06    23:18:38
```

Fig. 9 Sample Output - Full Flight Summary regarding test condition completion status (C = complete, P = partial completion, and N = not accomplished), condition start and stop times, and any notes applicable to the condition. The output provided by this module included both a run summary, in numerical order of run number, according to the knee-card (and adding extra conditions at the end) and a Test Item summary, in ascending order of TI number. This module provided an invaluable historical record of test condition accomplishment and was used extensively even after completion of the test program.

- Preparing Automated Data Requests (Figure 10). This module alone saved many hours of effort when compared to manually completing data requests. The time segments requested for data processing were taken from the Post-flight module, while the output formats were derived from a standard formats document.

- Preparation of an Instrumentation Arrangement Sheet (Figure 11). The status of each instrumentation parameter was ascertained subsequent to each flight time in order to provide a historical record, for review and analysis of the data, and to resolve data discrepancies. Since the Airborne Data Acquisition and Monitoring System (ADAMS I) was used to pre-flight instrumentation and monitor its status during airborne tests, tapes of the Pulse Code Modulation (PCM) data could be replayed on the ADAMS I to, in effect, refly the test condition and status the parameters.

```
                                               PRODUCT REQUEST FORM

  date today: 16/ 5/83   originator: _______________  phone: ________   project name: KC135R    Jo

  test date: 14 DEC 19  test no.: 065      tail no.: 293      request approval: ________________

  data library rec. date: ___________   complete/return to project dpt date: ___________

  tech. name: _______________ action date: ___________     start time: ________    stop time: ___

  MEDIA: L = LIST; P = PRINT TAPE (MICROFICHE); S = STRIPOUT; J = PLOT; D = DUB TAPE; M = MISC.
                          1     1    2  2                    3                    4    4     5     5
1      5                  2     7    2  4                    5                    6    8     2     8
----------------------------------------------------------------------------------------------------
PROD  MEDIA             DIGIT  PRINT- *  START TIME          STOP TIME          I    TAIL  TEST   C
CODE                    RATE   PLOT      (DECIMAL            (DECIMAL                NO.   NO.
                          $    RATE       REQUIRED)           REQUIRED)
       L P S J D M             $
----------------------------------------------------------------------------------------------------
 ZZZZ 2.23.001.001
 MPE1   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPS1   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPE2   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPS2   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPE3   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPS3   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPE4   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MPS4   1                      1.000     154245.0            154345.0           M    293   06523  UN
 MD4A   1                      1.000     154245.0            154345.0           M    293   06523  UN
```

Fig. 10 Sample Output - Automated Data Request

```
A/P __________                                    4-APR-83 009:55:04

KC-135 RE-ENGINE                              INSTRUMENTATION ARRANGEMENT

TEST NO: __________

TEST DATE: __________                                 PCM SYSTEM


           NO OF  MEASURE                                            CALIB      ARU   XDUCER     S/C
 WD    FR  OCCUR   I. D.        MEASUREMENT DESCRIPTION             NUMBER     GAIN    S/N    CARD S/N
 32     5    4    A005    VERTICAL-BODY STATION 825 (C.G.)         81247-01     5.     205      202
 34     5    4    A006    LATERAL-BODY STATION 825 (C.G.)          81249-01     5.     206      K16
 36     5    4    A007    LONG-BODY STATION 825 (C.G.)             81253-01     5.     207      K15
 38     5    4    B003    DIFF PRESS NOSE BOOM/CONE STATIC         82446-03            101      061
 40     5
 38     5    4    B003A   AIRSPEED (NOSE BOOM)                     82447-03            101      061
 40     5
116     9    1    B004    ADC AIRSPEED TO FCAS                     82399-02     1.     N/A      511
124     9
  6     1    4    B007    STATIC PRESS (BOOM OR CONE)              82322-02            102      061
  8     1
  6     1    4    B007A   ALTITUDE (BOOM OR CONE STATIC PRESSURE)  82323-03            102      061
  8     1
 10     1    4    B009    ADC ALTITUDE (AAU-19) #1 PILOT           82428-03            N/A      001
124     1    1    B010    ADC ALTITUDE (AAU-19) #2 COPILOT         82431-04            N/A      002
```

Fig. 11 Sample Output - Instrumentation Arrangement Sheet

- A Program Summary by Flight (Figure 12), which provided a by-flight, first-time completion status of test conditions. Partial completions and repeated test conditions were not included in this data, so it provided management a cumulative flight record at a glance.
- A Program Summary by DTIS (Figure 13). This summary provided a more detailed breakdown by category of test, depicting the completed, partially completed, and remaining conditions, by number and percentage. This information was directed primarily toward engineering management in order to reveal those areas of primary interest that remained to be addressed.

K C 1 3 5 R   P R O G R A M   S U M M A R Y

as of 16-MAY-83
(through flight 56)

Flight Number	(date)	Number of TI's Completed this flight	to date
1	04-AUG-82	8	8
2	18-AUG-82	15	23
3	25 AUG 82	45	68
4	30 AUG 82	2	70
5	02 SEP 82	34	104
6	03 SEP 82	1	105
7	10 SEP 82	43	148
8	14 SEP 82	235	383
9	20 SEP 82	41	424
1C	21 SEP 82		427
	14 OCT 82		481
			501
54		4	
55	30 MAR 83	8	
56	5 APR 83	37	1748

Total Number of TI's Completed to date: 1748

Fig. 12 Sample Output - By-Flight Summary

K C 1 3 5 R   P R O G R A M   S U M M A R Y

as of 16-MAY-83
(through flight 56)

DTIS	TI	Description	Total	Complete	CONDITIONS Partial	Remaining	%Complete
9.2.1.1.1							
	1.01.001.	PITOT-STATIC	39	39	1	0	100%
	1.06.001.	TAKEOFF PERFORMANCE	27	27	6	0	100%
	1.13.001.	V SO	32	32	10	0	100%
	1.15.001.	LOW SPEED PERFORMANCE	41	41	4	0	100%
	1.17.001.	CLIMB PERFORMANCE	38	38	7	0	100%
	1.18.001.	DESCENT PERFORMANCE	8	8	2	0	100%
	1.19.001.	CRUISE PERFORMANCE	78	78	0	0	100%
	1.20.001.	LANDING PERFORMANCE	21	21	6	0	100%
	1.31.001.	BUFFET CHARACTERISTICS	6	6	0	0	100%
	1.09.001.	SIMULATED RTO'S	2	2	0	0	100%
	Totals for this DTIS:		292	292	36	0	100%
9.2.1.13							
	7.13.003.	COCKPIT EVALUATIONS	6	6	1	0	100%
	Totals for this DTIS:		6	6	1	0	100%
9.2.1.13.7							
	7.01.003.	INTERIOR NOISE LEVELS - FLT TEST	9	9	0	0	100%
	Totals for this DTIS:		9	9	0	0	100%

Fig. 13 Sample Output - DTIS Summary

TI 3.12.002. with 13 uncompleted Conditions

TI	CN	CT	GW	ALT	K-M	CG	TY	E	GS	FS	FLT	PRI	CMNT
3.12.002.	013	20	225	350	285	0	DS	4	UP	00	0	/1/	02 ANTI ICE
3.12.002.	025	10	0	0	250	0	CR	4	UP	00	0	/1/	REPEAT MAX CONDT
3.12.002.	030	10	0	0	250	0	CR	4	UP	00	0	/1/	REPEAT MAX COND
3.12.002.	031	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .2 ICE 1/4
3.12.002.	032	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .2 ICE 1/2
3.12.002.	033	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .2 ICE 3/4
3.12.002.	034	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .2 ICE 1
3.12.002.	035	10	0	0	200	0	CR	4	UP	00	0	/1/	REPEAT MAX COND
3.12.002.	036	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .7 ICE 1/4
3.12.002.	037	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .7 ICE 1/2
3.12.002.	038	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .7 ICE 3/4
3.12.002.	039	10	0	0	200	0	CR	4	UP	00	0	/1/	LWC .7 ICE 1
3.12.002.	040	10	0	0	200	0	CR	4	UP	00	0	/1/	REPEAT MAX COND

Fig. 14 Sample Output - Conditions Remaining Summary

- A Conditions Remaining Summary (Figure 14). While not strictly management visibility or tracking, this output proved invaluable near the completion of the program. It provided those conditions remaining, not only by TI number, but also by type; i.e., takeoff, cruise, maneuver, landing, taxi, etc. It aided scheduling of these conditions, particularly when a program completion date estimate was required two months prior to the contracted completion date. This summary was used to construct mission profiles for all remaining flights, including contingencies due to weather, cancellations, or inoperable systems. The planning flexibility was greatly enhanced when hard copies were available to the Test Conductors for every remaining flight.

Overall, the capabilities of the entire automated system were developed during the entire ground and flight test program. User friendly software was continually stressed to the two BCS software engineers who were assigned to the development. The automated system resulted in the ability of six test engineers from BMAC and one from AFFTC to effectively plan, conduct and provide control over a test program normally requiring four to five times that number.

Shortcomings

As with any new program, several shortcomings in the use of the system were identified. Most were corrected on the spot, but some were unable to be rectified during the development.

The initial development costs of writing, revising and debugging software were significant, but were considered cost effective when the flight test program was initiated. Approximately 900 manhours and $5,000 in computer costs were spent in developing the capabilities described above. Much of the software is considered applicable to future test programs.

The ease of changing the planning for specific missions became a mixed blessing when the very limited number of operators was tasked to change missions constantly in order to keep up with changing weather or program directives. Six separate profiles were developed in one day to meet varying contingencies, and none were eventually used. The flexibility of the system, while an asset, could be and was seemingly used capriciously at times.

Communication of hardware and software problems by telephone between the users at the remote site and the software development engineer at BMAC was unsatisfactory. The benefits of having a software development engineer on-site while testing at a remote facility would far outweigh the cost of locating him with the test team.

Time sharing on the mainframe computer was a problem at the initiation of remote testing. The ability to develop a kneecard was given priority on arranged days and at other coordinated times in order to preclude the waste of time and unnecessary expense.

The use of a 1200-baud line between the mainframe and the remote terminal was a handicap during the operations at Edwards AFB. Because a dial-up dedicated phone line was utilized, random noise and interference on the line were experienced to the extent that extraneous commands and inputs were invariably received during usage. Several times the printer was commanded OFF during a print sequence of a kneecard.

Enhancements

The initial use of the automated system highlighted improvements that could, or should, be incorporated in future uses. The integration of text processing capabilities of the system is an enhancement that would ease the problem of combining engineering test conditions and the required notes, instructions and comments in the kneecard construction module. This availability and the use of trained text processing personnel would increase the value of the capability.

The project history file, or collection of instrumentation calibrations and effectivity dates, was maintained separately from the automated system during the test program, but integration into the computer data base would be desirable.

A dedicated mini-computer with hard disk and the availability of a tape backup, as originally envisioned, remains a desirable method for automating test planning and tracking. If a phone modem is used, a dedicated high speed connection between the modem at the remote facility and the mainframe at 9600 baud, dedicated software and hardware support, and a large printer buffer at the remote facility should be requirements for the system.

Transmission of processed data from the remote facility to the analysis engineers via the dedicated hardlines would be advantageous compared to shipment of microfiche or data tapes.

The use of multiple terminals while remote testing is a requirement that was immediately recognized after attempting to operate from a single terminal. Flying two or more test flights per week revealed a priority problem among knee-card and profile preparation, instrumentation arrangement and visibility production. The addition of another user terminal would have proven invaluable.

Conclusions

The automated test planning, tracking and visibility system developed for and used on the KC-135R Flight Test Program demonstrated a significant increase in test engineering productivity. The conduct of an extremely time and budget limited flight test program at a remote facility placed demands on manning and personnel effectiveness that necessitated this system. The computer provided the data base sort and search capability required to effect sufficient planning, status reporting and tracking of a major flight test program, utilizing a small fraction of the manpower normally associated with a program of this size. A significant cost savings will be realized, and the potential for future cost avoidances is essentially in hand.

The automated program can be easily adapted for use on other test programs and is a valuable asset in test program management.

GROUND SUPPORT FACILITIES
THE WAY TO EFFECTIVE AVIONICS FLIGHT TESTING

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Abstract

This paper proposes uses for an avionics ground support facility and addresses specific methodology for avionics integration and radar system ground tests in support of flight testing. Simulation has become a normal process in developing and troubleshooting integrated avionic systems. It is an economical alternative to the old fly-fix-fly technique. The number and magnitude of the functions performed with new avionic systems makes it very difficult and expensive to achieve successful systems integration on board a flying vehicle. Avionic simulation is a technique for operating avionic systems in a ground support facility. A support computer provides artificial flight parameters to the test facility and controls the simulation. Software models for the earth, atmosphere, airframe dynamics, etc., are provided. The productivity of test flights can be significantly improved by providing the capability to prefly and troubleshoot integrated avionics on the ground. Each avionic system can be thoroughly tested as an integrated system in a realistic simulated environment before being flight tested. System changes can be completely investigated using preprogrammed mission scenarios or past flight data.

Introduction

The tendency of modern avionic systems to become more complex and sophisticated has had major impacts on flight tests. First, the number of system integration test requirements and related problems has increased dramatically. Because avionic subsystems are becoming more dependent on each other, simply testing each subsystem individually and putting it on board an aircraft does not often result in a workable system. Therefore, much in-flight time is spent determining whether the system performs its intended mission. Because of the high expense and relatively long turnaround time associated with flight test, the fly-fix-fly method of troubleshooting is not a very efficient way to solve system integration problems.

Second, as changes are made to a system during its development, the modified system must be checked prior to resuming the test sequence. Reflying previous missions to accomplish this is not practical.

Third, when problems are discovered during flight test, they must be adequately described to the developer along with the specific conditions under which these problems occur. Since in-flight instrumentation is limited by numerous technical considerations, the amount of information derived from the flight is frequently not sufficient to do this.

Fourth, learning to use the system has become more difficult. It is not reasonable to expect flight crews to learn how to effectively use the system simply be reading the manuals, and flight time dedicated for this purpose is costly. Consequently, much of the crews' in-flight time is spent learning to use the system rather than on evaluating it.

Recent avionics flight test programs have required an additional three to four times as many test flights as initially planned to develop one or more new avionic systems. One of the prime drivers for this significant increase in test flights is the flexibility that avionics embedded software provides and the relative ease in which software changes can be made. These changes are often made and flight tested before sufficient checkout is performed to determine all effects on the modified system and other systems.

Government and contractor ground support facilities (GSFs) are being developed to handle the integration and testing of these avionic systems prior to evaluation in flight. A good example of one of these facilities is the Integration Facility for Avionic Systems Testing (IFAST) which is being developed and acquired by the Air Force Flight Test Center (AFFTC), Edwards Air Force Base, California. IFAST will be an integration and simulation facility for both government and contractor personnel to support Development Test and Evaluation (DT&E) and Initial Operational Test and Evaluation (IOT&E) of airborne avionic systems. IFAST is designed to accommodate four separate programs concurrently. It will reduce or eliminate recurring costs to modify AFFTC flight line buildings and will reduce the duplication of Automated Data Processing Systems (ADPS) presently needed to support multiple flight test programs. IFAST will provide the capability to operate and integrate complete avonic systems in real-time simulated flight. These activities will be performed in individual test bays that are specifically designed for dynamic ground operation and troubleshooting of avionics. This ground support facility will include common databases and automated support tools. Therefore, Combined Test Force engineers will have the tools needed to ensure quantitative assessment of avionics to permit timely consideration of test results for avionics acquisition decisions.

Modern aircraft weapon systems undergo major changes from time to time to keep pace with technological advances and to respond to changes in operational requirements. These major changes require flight test verification of both hardware and software. A significant portion of the enhancements and modifications will be implemented in software; therefore, one of the objectives of a GSF will be to verify functional operation of the new Operational Flight Programs (OFPs). Another objective can be to perform sensitivity analyses for flight test planning.

The productivity of test flights can be significantly improved by providing the capability to prefly and troubleshoot integrated avionics on the ground. Each avonic system can be thoroughly tested as an integrated system in a realistic simulated environment before it is flight tested. System changes can be completely investigated using preprogrammed mission scenarios or past flight data. Problems which occur during flight can be thoroughly investigated in the ground support facilities where instrumentation may be more comprehensive than on the aircraft. Pilots and flight test engineers will be able to operate avionic systems before flight to become totally familiar with them and to resolve any questions not covered by documentation. Flight test personnel will then be able to concentrate on mission related evaluations which will be performed by thoroughly trained flight crews.

Description of Ground Support Facilities

Ground support facilities for avionic systems have been in existance for some time. These facilities have undergone much development and numerous refinements over the years due to changing requirements and evolving technology.

Ground support facilities provide numerous functions based on the operator requirements. Prime airframe contractors (or prime avionics integration contractors) will require such a facility for design, development, and integration of major new avionics systems. The Navy uses ground support facilities for the integration of avionics and for all logistics support after development of the overall weapons system. Examples of these are the F-14 facility at the Pacific Missile Test Center, the F/A-18 facility at the Naval Weapons Center, and the AV-8B facility at the Naval Air Test Center. The Air Force is developing the IFAST at Edwards Air Force Base to support DT&E and IOT&E of airborne avionics systems. The Air Force also is developing facilities at the various logistic centers to provide support to the avionic systems after development is completed. Examples of these are the F-15 facility at Warner-Robins AFB and the F-16 facility at Ogden AFB.

The GSF should be located at the test site and be capable of supporting related (e.g., competitive flyoff) and non-related (e.g., tactical and strategic) test programs. The facility should be located in a relatively isolated area to provide security, electromagnetic radiation isolation, and airspace for conducting ground-to-air tests.

A typical GSF should contain a time-shared computer complex. This capability is provided to allow the development of real-time software, support of test scenario generation, and remote communications on a concurrent basis with an ongoing simulation. Portions of the time-shared computer complex may also be diverted to real-time support when necessary. This complex should provide the following:

(1) A simulation development computer; disc storage units; magnetic tape units; a printer/plotter; and a raster graphics display system.

(2) A graphics development computer; disc storage units; a magnetic tape unit; a hardcopy terminal; a color picture system with work stations; a graphics system; and an electrographic film plotter.

(3) A general purpose computer; dual-port disc units; magnetic tape units; a scientific character line printer; a letter quality printer; and remote device communications interfaces.

The time-shared computer complex should include a complement of operator's consoles, interactive terminals, real-time clock modems and inter-computer communications.

The GSF should provide multiple independent program test areas with controlled access in order to preserve program security and contractor proprietary requirements. Each program test area should contain an electromagnetic radiation shielded test bay, an antebay with a radome for radiating to the outside into an open area, an engineering work area, and a maintenance area.

The primary real-time simulation capability should be provided by the Automated Data Processing Equipment (ADPE) in the test bay and may be used to interconnect directly with the avionic system under test. The following is a typical list of ADPE to support each test bay:

(1) Real-time simulation control computers; disc storage units; magnetic tape units; line printer/plotter units; and a raster graphics display system for data analysis and presentation.

(2) A real-time graphics control computers; disc storage units; a hardcopy terminal; a color picture system with work station; a graphics system; and an electrographic film plotter.

(3) A simulation data monitoring computer; disc storage units; and interface units to support avionics system interconnection.

Additionally, each of the above computer system should contain an operator's diagnostic console, interactive terminal, a real-time clock with 1 microsecond resolution, a modem for telephone interconnect, and an intercomputer communications device. Alphanumeric terminals with data plotting capabilities should be available for data review from any of the facility computers.

The graphics equipment available in the bays and central complex will serve three primary requirements: generation of 3-dimensional color environment scenarios, generation of 2-dimensional cockpit displays (e.g., HUD), and graphic presentation of data for review and analysis. The graphics systems should interface with the plotter to produce graphic hardcopies.

Each of the computers should be equipped with a high-speed parallel interface, which will allow direct memory access data transfers. The interface may be used for module-to-module communications during real-time operation.

The roof of the GSF should be designed and structured to test antennas. Provisions for electrical and electromagnetic connections between the roof and the test bays should be incorporated in the design. A large freight elevator should service all test bays and the roof. Since the antebays in the test bays will be well suited for testing normal-sized avionic antennas, it is expected that larger antennas, such as the E-3A AWACS antenna, will be the primary users of the GSF roof structure.

Utilization of Ground Support Facilities

The use of ground support facilities for airborne avionic systems is becoming a more integral part of most flight test programs. Historically, this support has been solely in the areas of avionics repair and maintenance. With the tremendous advances in technology in both hardware and software, current and future avionic systems present a much more complex system to be tested. This testing complexity is further amplified by the need for integration testing of various combinations of these systems in order to evaluate particular capabilities.

Current radar systems are excellent examples of the trend in all avionic systems toward increased use of embedded digital computers and supporting software, i.e., Operational Flight Programs (OFPs). Relatively speaking, the digital computer hardware plays a minor role, compared to the OFPs, in improving the capabilities of a given radar system (or most other avionic systems). For example, most upgrades to current radar systems occur in the areas of digital signal processing and data processing. This normally requires increased computer and/or processing capacity (which may be as simple as replacing or adding a memory board) plus substantial software development and modification and more software (OFP) testing.

Primary GSF operations should be in support of flight test programs. For a typical operation, the test force personnel can develop, maintain, and operate the program-peculiar equipment and perform the avionics testing, analysis, and evaluation. The test force will be supported by GSF personnel who will develop, maintain, and operate the generic test support systems.

Each test program to use the GSF will develop the software package required for the test program. This package will be based on available GSF, government, and contractor developed software/models. There will be several options for developing simulations for avionics testing in the GSF. Three of these are:

(1) The test force can modify generic software models and executive routines to operate on GSF hardware.

(2) The test force can develop the simulation software and hardware to interface with GSF supplied hardware and software.

(3) The test force is provided only a test area (test bay, work area, power, cooling, etc). The test force provides the hardware and software for real-time simulation, interfaces, and data processing.

The GSF can be used both prior to the flight test and in conjunction with flight test. Before avionic flight test begins, simulations of various test missions can occur. Any problems encountered can be analyzed, corrected, and re-evaluated prior to flight test. These simulated missions may use data recorded on previous flights to stimulate the avionic sensors. Once problems discovered during ground testing in the GSF are corrected, the avionics systems can be flight tested. Problems discovered during flight test can be duplicated in the GSF (if possible), and after appropriate corrections are implemented, the modified system can be retested in the GSF, prior to flight. This procedure can be followed to the extent practical. For example, if a flight problem cannot be duplicated in the GSF, one would not expect to verify the correction in the GSF. Nevertheless, the GSF can be used to determine if the correction of that problem has caused another problem elsewhere in the integrated system.

The GSF can fulfill other objectives in support of the overall flight test program. These are:

(1) The GSF can support development, integration and testing of avionic systems.

(2) The GSF can be used for avionic system familiarization for air crew members and engineers. Man-in-the-loop dynamic simulation of integrated avionic systems can help both crew members and engineers understand system operation and characteristics.

(3) Man-in-the-loop simulations can also aid in an early evaluation of man-machine interface characteristics.

(4) Flight test mission planning can be conducted.

(5) The GSF can be used to aid in the development, test and validation of avionic instrumentation systems.

The ability of ground support facilities to support many different avionic systems has been greatly enhanced by the implementation of various avionics standards. MIL-STD-1553 defines the digital multiplex bus used to communicate between avionic subsystems. MIL-STD-1750 defines the Air Force 16-bit computer instruction set architecture. From a software point of view, MIL-STD-1589 or JOVIAL J73 is the Air Force standard higher order language for avionics embedded computers, and is used in applications associated with MIL-STD-1750. Another important standard being developed for Air Force avionic systems is MIL-STD-1760. This standard defines the electrical interface between an aircraft and weapon stores. These avionic standards enhance the accessibility of test data from the digital buses, promote the use of standardized test instrumentation systems, and permit the use of developed software on multiple programs, thus reducing test costs and time.

Test Methodology for the GSF

Primary Roles and Applications of the GSF

The test methodology associated with avionic system and integration testing in the GSF is presented below as applied to a specific system, an air-to-air attack radar.

The four primary roles and applications, which follow, consider the most likely relationships between radar systems to be tested and the total weapon system avionic suite.

The primary emphasis of the GSF is dynamic testing of integrated avionic systems. Airborne radars will be tested as one part of an integrated avionic suite. It is assumed that, in this case, radar systems will have undergone sufficient development engineering testing and subsystem integration testing to be ready for avionic suite testing prior to arrival at the test site. Also, the emphasis in the GSF should be on dynamic testing using actual avionic line replaceable units (LRUs). This emphasis on dynamic testing using actual hardware represents testing more closely aligned to the real operational environment.

GSF operations will be supportive of, not competitive with, flight test programs. The GSF will simulate the flight environment to the maximum extent practical. The simulated flight environment provided by the GSF will permit avionic tests to be "pre-flown" in the simulated environment. This will result in a significant reduction of flight hours being dedicated to "check out" of fixes, including software patches, to avionic problems.

Instrumentation systems used in the GSF will exhibit considerable commonality with flight test instrumentation systems. The GSF can be used to perform a thorough checkout of avionic instrumentation systems prior to flight. The determination of whether to use identical instrumentation systems for ground and flight tests will be a test force responsibility. The existence in the GSF of a bus monitoring system will permit the collection of the same data parameters during both the GSF and flight testing. Radar parameters which are not transmitted on the MUXBUS, such as inter-LRU bus parameters and digital signal processor data which are collected in flight, can also be recorded in the GSF. This means that the same analysis tools can be used for both ground and flight test.

A key capability of the GSF will be the ability to dynamically exercise OFPs and assess the effects of OFP changes on integrated avionic systems. The existence of a dynamic integrated avionics test simulator with extensive parameter monitoring capability will permit software changes to an OFP to be evaluated not only for the changed OFP but for the effect of the change on any associated avionic system(s). For example, the effect on the HUD of a change in the radar OFP can be determined. Similarly, the effect that a change in the central computer OFP has on the radar system can be evaluated.

The GSF should be designed to facilitate troubleshooting avionic system design problems. Routine hardware maintenance activities are assumed to be conducted outside of the GSF. However, the capability to perform integrated hardware - software troubleshooting of avionic subsystems in stand alone configurations is inherent in the GSF design. This implies that, except for hardware maintenance capabilities, the same capabilities of a radar system test bench will be present in the GSF, although the test bench itself may not be in the GSF. Extensive avionic simulation capability in the GSF will enable all interfacing avionic systems to be simulated while the actual radar system is exercised.

GSF Radar System Test Environment

Two basic environments are necessary for ground testing of a radar system. The integration environment creates the "test aircraft". It provides the capability to simulate aircraft motion, aircraft location and the physical environment. Additionally, it creates a system which provides the means to generate other avionic data, provides the control of communications among the avionic subsystems and provides the communication interface between the integration environment and the system under test. It should be noted that the integration environment is applicable to testing any avionic subsystem. The other environment, called the Radar Test Stand (RTS), is a physical structure to support the actual radar hardware and to provide power, cooling fluid and pressurized air necessary for radar operation. Included in this environment is a support computer which allows standalone testing.

That ground testing in the GSF environment is subject to the same test planning, scheduling and procedural disciplines as actual flight test. Thus, in the GSF test environment, flight tests would be limited by the ability of the simulation to reflect the desired test situation. In this regard, an initial step is the establishment of baseline configurations which are verified and validated using real world data. This is analogous to making sure the flight test aircraft is operational. Scenario generation is somewhat more complex than actual flight test mission planning in that more data must be specified. In addition to aircraft and target locations, detailed data for simulation initialization must be delineated.

The type of stimulus exciting the radar system dictates the kinds of tests that can be conducted on the ground. There are three basic stimuli: RF, IF and digital. RF signals are inserted through the antenna or in the receiver front end, IF signals are inserted before the A/D converters and digital signals are inserted after the A/D converters.

Integration Environment. The ground support facility must create a real time, dynamic environment. Functions to be performed by the simulation include:

(1) System and simulation control

(2) Scenario generation

(3) Computation of airframes kinematics

(4) Environment simulation

(5) Weapon and sensor simulation

(6) HUD simulation

(7) Data processing

A key element in this configuration is a Command and Monitor Device (CMD) which creates and monitors the MUXBUS. This device is a special digital interface device which performs the function of bus controller, bus monitor and multiple remote terminals as defined in the MIL-STD-1553B. The integration environment requires a dual CPU minicomputer; a medium size minicomputer in conjunction with graphics systems to generate out-of-cockpit displays; a medium size minicomputer for monitoring the MUXBUS data and performing real time engineering unit conversion; two small minicomputers for cockpit interface; and intercomputer communications devices. The dual CPU minicomputer hosts the software models necessary to create the dynamic test environment.

Integrated Avionics Test Software Requirements. The simulation system should provide the capability to exercise the total avionic system in a simulated operational environment. The system simulates the aircraft dynamics, environment and appropriate avionic equipment so the software in the central avionic computer and radar computer can be exercised through the various system modes and functions. Basic modules and functions are summarized below:

(1) Executive (Simulation Control and Timing) - In the full-up system simulation mode, the GSF would perform the real time simulation and exercise control of the total complex. The radar test stand support computer would also be under control of this executive.

(2) Aircraft Dynamics - This software would perform the computations of the 3 DOF and 6 DOF aircraft equations of motion and derive the aircraft attitude, attitude rate, position and velocity information. This simulates the Flight Control System in automatic and manual operation.

(3) Environment - Environment models would include standard atmospheric models, gravitational models, wind profiles, etc. This simulates the Air Data Computer (ADC) and its sensors.

(4) Other Avionic Subsystems - The other avionic subsystems would include the inertial navigation system and the central computer. The functions of these two would be simulated to facilitate system testing without utilizing the actual hardware.

(5) HUD Simulation - The HUD simulation would provide the data and interface with the graphics system to display the HUD data in lieu of utilizing the actual HUD hardware and provide out of the window background display.

(6) Weapon and Sensor Models - Weapon models would simulate missile trajectories and bomb scoring. Sensor models would include the radar, infrared sensors and laser ranging devices. Simulation of these functions would allow one to exercise the other elements of the avionic system without the actual hardware being utilized.

(7) Stores Management System Simulation - The stores management function would stimulate the interface between the aircraft system and the weapons. These would include the computation of safe release zones, alignment of missile seekers, launch initialization data and weapon release discretes.

(8) Data Processing - This software module would support the compilation and analysis of the test data. The function would include data formatting, engineering unit cOnversion, statistical analysis, etc.

Due to the volume and detail of data needed to generate test scenarios, an interactive scenario generator is necessary to insure completeness, consistency and accuracy. This off-line program allows the mission planner to input data, via a terminal, to define modes of operation, geometry and characteristics of target and test aircraft and change system parameters.

Radar Test Stand. Testing a physical radar in a ground facility requires the capabilities of current radar test benches:

(1) LRU mounting, electrical interfaces, distribution of air and cooling fluids for proper operation of the system.

(2) Provide measurement access (interface and breakout boxes, etc.) for performance testing and evaluation.

(3) Operation of the radar as a system or operation of selected LRUs.

In addition, the test system should include a target generator with the capability to generate the RF and digital target signature data. Target parameters for the target generator should be remotely programmable from the support computer. In addition to static targets (clutter), the generator must have the capability to stimulate Doppler frequencies representative of a moving target, and to simulate the effects of ground clutter and jet engine modulation.

Interface and control of the radar system is accomplished by a general purpose support computer. The support computer provides control of the test operation, controls the target generator functions and also provides the capability to capture data from the radar computer during the test operation. Interfaces with other avionic subsystems (such as the INS and central computer) are provided by either using actual hardware or by simulating the data interface.

Software Required to Support the RTS. Software required to operate the radar system utilizing the RTS in the standalone mode should include the current version of the radar system operational flight program(s) and software residing in the support computer.

In the RTS standalone mode, the support computer should provide the real time simulation control and monitoring function for total RTS operation. This includes the functions of:

(1) Simulation timing and control

(2) Avionics simulation

(3) Simulation scenarios

(4) Target generator control

(5) Dynamic target generation

(6) Data display and acquisition

(7) Data processing and analysis.

Data Handling Capabilities. The performance monitor system (PMS) allows the selection and recording of the MUXBUS traffic for data analysis. This system is comprised of a medium size minicomputer, a MUXBUS monitor device, disk storage and a tape transport. The minicomputer provides control of the hardware devices and performs real time engineering unit conversion of MUXBUS data. Additional information required for analyses are internal radar parameters, TSPI data and recorded video from the radar displays.

Two data handling capabilities are required: real time monitor capability and post mission analysis capability. The real time monitor capability allows considerable time savings in the following areas: initial operational checkout of the baseline configuration, initial checkout of the system with the radar installed, verification of mission scenarios and monitor of selected test data during actual testing.

The post mission analysis capability allows the quick reaction checkout of parameter time histories and the production of report quality plots. This interactive capability would include the generation of titles, legends, grids, grid marking, legends and comments for single or multiple plots. The quick reaction capability would require a graphic CRT with copy capability. Also included in this system would be tape and disk data handling routines, data scaling, data smoothing and standard analysis tools, such as regression and least squares analysis routines. The above capability, coupled with a word processor, could aid in the efficient and timely production of reports.

Test Methodology

Radar testing in the GSF will generally be performed to fulfill one or more of the following purposes:

(1) To determine if the radar system, operating as part of the integrated avionic system and in the ground test environment, performs consistent with values specified for inflight performance.

(2) To prefly integrated avionics flight test missions and familiarize aircrew members and engineers with the system.

(3) To troubleshoot radar design problems discovered during ground testing or flight testing.

(4) To checkout OFP changes in a dynamic integrated avionic system environment.

Effective testing in the GSF requires carefully planned test scenarios. The scenarios will generally be prepared by the test force personnel. These scenarios will be system unique to the extent that individual test trials are dependent on end item specifications and specific system performance values. However, much of the scenario preparation task will involve the appropriate test force personnel interacting with common GSF software in a question-response fashion to input information such as ownship altitude, way points, radar fix points, target altitude, range, velocity, relative bearing, RCS, etc. Scenarios, once constructed, can be retained by the GSF for future use or for modification and subsequent use on the same program or a different program. Frequent use of these "canned" scenarios will aid in insuring test repeatability, confirming satisfactory radar system operation after an OFP change, or duplicating standard flight test profiles in the GSF. Also, standard scenarios permit adjusting one variable through the full range of values while holding other variables constant. For example, target characteristics can be repeated as the radar is cycled through the automatic acquisition modes of HUD field of view, vertical scan, boresight scan, and slewable scan.

The test force radar engineer can help ensure test program continuity and completness by setting up a simple matrix of radar ground test requirements versus the scenario(s) to be used to fulfill the requirements. The completed matrix can be used to determine the need to generate new test scenarios, the potential to improve test efficiency by modifying scenarios to accommodate more test events, and to ascertain that all ground test requirements are met.

Radar ground test methodologies and flight test methodologies should be as similar as is reasonably possible. This includes both the test scenarios and test configurations mentioned above. The immediate benefits of this similarity are:

(1) The ability to determine the correlation between flight test data and ground test data.

(2) Preflying flight missions in the GSF will be more easily accomplished and more meaningful.

(3) Duplication of flight anomalies in the GSF will be more readily achieved.

(4) Similar data processing and analysis processes can be used for both ground test and flight test data.

In the longer term, the use of similar test methodologies will provide empirical data to support determinations of the GSF test validity, flight tests which can be supplemented by the GSF tests, and the potential to use the GSF data to demonstrate specification compliance.

The basic data analysis method common to all the radar test methodologies is to compare data from the radar with a reference system and determine the differences. The differences are then compared with required performance values as delineated in specifications and/or other requirements documents. This generally requires data time tagging and interpolation of data. Reference system data may be obtained from the GSF test environment system.

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FLIGHT TEST AND EVALUATION OF THE A-10 SINGLE-SEAT NIGHT ATTACK AVIONICS

by

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Abstract

This paper describes the major A-10 single-seat night attack avionics systems, the test procedures used to evaluate each system, the results of the test program, and some recommendations. The test program satisfied all major avionics test objectives by developing and employing new and innovative test procedures. The results of the evaluation provide an insight into the effectiveness of the test procedures and the A-10's night attack avionics systems. This paper is also intended to serve as a partial guide for testing future night attack avionics sysems.

Introduction

The objective of the Calibration and Resolution Phase of the A-10 Single-Seat Night Attack (SSNA) program was to determine and document the capabilities of an avionics suite designed for single-seat night attack operation. This evaluation was conducted from 4 Aug to 1 Dec 82 on an avionics suite configured for single-seat operation from the front cockpit, while the rear cockpit was configured as a safety observer station. The avionics systems used during the evaluation are listed in Table 1. This paper, however, concentrates on the test and evaluation of three of the major avionics systems used for navigation and targeting. These systems include the Terrain Following/Terrain Avoidance (TF/TA) radar, the Radar Altimeter (RA), and the Forward Looking Infrared Receiver (FLIR). The purpose of this paper is to briefly describe each of these subsystems, the test procedures and methods used to evaluate each system, and the results of the evaluation. The Recommendations section of this paper should serve as lessons learned for future night attack avionics testing.

Table 1 SSNA Avionics System

System	Manufacturer
WX-50 Terrain Following/ Terrain Avoidance (TF/TA) Radar	Westinghouse
AAR-42 Forward Looking Infrared Receiver (FLIR)	Texas Instruments
Head-Up Display (HUD)	Kaiser
LN-39 Inertial Navigation System (INS)	Litton
AN/APN-194 (V) Radar Altimeter (RA)	Honeywell
Multifunction Displays (MFDs)	Westinghouse
Model 105D Laser Ranger	Ferranti
Electronic Moving Map Display (EMMD)	Astronautics
Low Light Level Television (LLLTV)	Edo Western; Marconi
Malcolm Horizon	Garrett

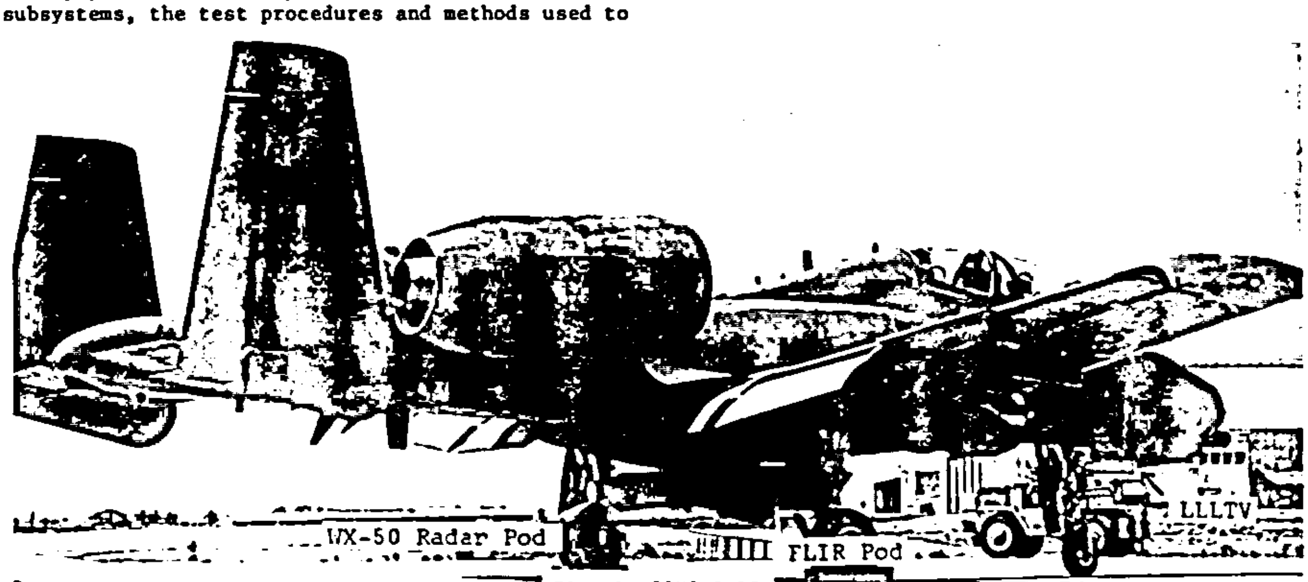

Fig. 1 SSNA A-10

Background

The SSNA A-10 Program was a follow-on to the Night/Adverse Weather (N/AW) Test Program conducted by the Fairchild Republic Company (FRC) , the Air Force Flight Test Center (AFFTC) , and the Tactical Air Warfare Center (TAWC) from 1 May 79 to 30 May 80. The YA-10B (USAF Serial Number 73-01664) in Figure 1 was designed for two-seat operation, though the N/AW Test Program had focused on the single-seat capability of the aircraft. That attempt to evaluate the single-seat capability of the aircraft to perform the N/AW mission was hindered because the cockpit design was not optimized for single-seat operations. For that reason only limited conclusions were drawn pertaining to the conduct of single-seat operations. Although the aircraft showed some potential in the single-seat role during the N/AW testing, judgement was reserved until the SSNA evaluation was conducted with a design better optimized for single-seat operations. The flight test objectives satisfied during the SSNA evaluation were extracted from the SSNA A-10 Basic Test Plan.

Test and Evaluation

Terrain Following/Terrain Avoidance Radar

The Westinghouse WX-50 multimode radar (Figure 2) was modified for use as a TF/TA radar for the SSNA A-10. This radar provided the pilot with a Terrain Following Box (TFB) and two contour lines projected on the Head-Up Display (HUD) simultaneously with either a Ground Map (GM) or Ground Moving Target Indicator (GMTI) displayed on a Head-Down Display (HDD). This simultaneous dual mode operation was achieved by using 2000 pulses per second during each 60 degree per second scan or approximately 16 pulses per one-half degree. One of 16 pulses were used for the TF/TA mode and 15 of 16 pulses were used for the GM or GMTI modes. Using 1 of 16 pulses provided 125 azimuth looks to the TF/TA processor during each 60 degree scan. For the other 15 of 16 pulses, the sum receiver was switched to carry returns from the main antenna to the GM and GMTI processor. The TF/TA radar was constructed with a 15-inch main antenna stabilized to ±45 degrees in roll. Furthermore, the radar operated at 35 GHz and produced a 1.6 degree cosecant squared elevation radio frequency (RF) radiation pattern for uniform ground illumination.

The flight test and evaluation of the TF/TA radar centered primarily around four objectives. These objectives were:

1. Verify the TFB commands proper terrain clearance.
2. Determine the accuracy of the contour lines.
3. Determine the effects of land-water transitions on all radar modes.
4. Evaluate the GM and GMTI modes of operation.

To satisfy these objectives, specific test procedures were followed. These test procedures could also be used to evaluate other night attack/adverse weather TF/TA radars. For instance, these procedures will be used to test and evaluate the Low Altitude Navigation and Targeting Infrared for Night (LANTIRN) Texas Instruments Terrain Following Radar (TFR) beginning in October 1983.

To accomplish the first objective, the pilot was instructed to "fly the box" by rigorously keeping the Total Velocity Vector (TVV) in the TFB. This was done to verify the TFB commanded proper terrain clearance. Specifically, this objective was accomplished by flying the A-10 over the TFR Edwards Surveyed Route (ESR) (formally B-1 TFR Route #2). The route extended eastward from the intersection of Runways 07 and 35 to Haystack Butte for 13 NM with a heading of 070. From west to east, the terrain was a gradual upslope. Haystack Butte, at the easternmost portion of the profile, protruded approximately 350 feet above the surrounding terrain. Flying the ESR served two purposes. First, the terrain produced good radar returns for the developmental testing of TFRs. Second, there was adequate radar coverage (FPS-16) and photo coverage (cinetheodolites) throughout the course to determine the position of the aircraft in three axes. This Time Space Positioning Information (TSPI), onboard bank angle, pitch angle, radar altitude, and barometric altitude were later merged to evaluate the WX-50 TF/TA radar. The aircraft was first flown straight and level along the ESR, then tested and evaluated during turning flight. During the turns, the aircraft was flown over Haystack Butte to determine what altitude the TFB commanded the pilot to fly.

The accuracy of the TF/TA contour lines (objective 2) was evaluated by flying toward known terrain

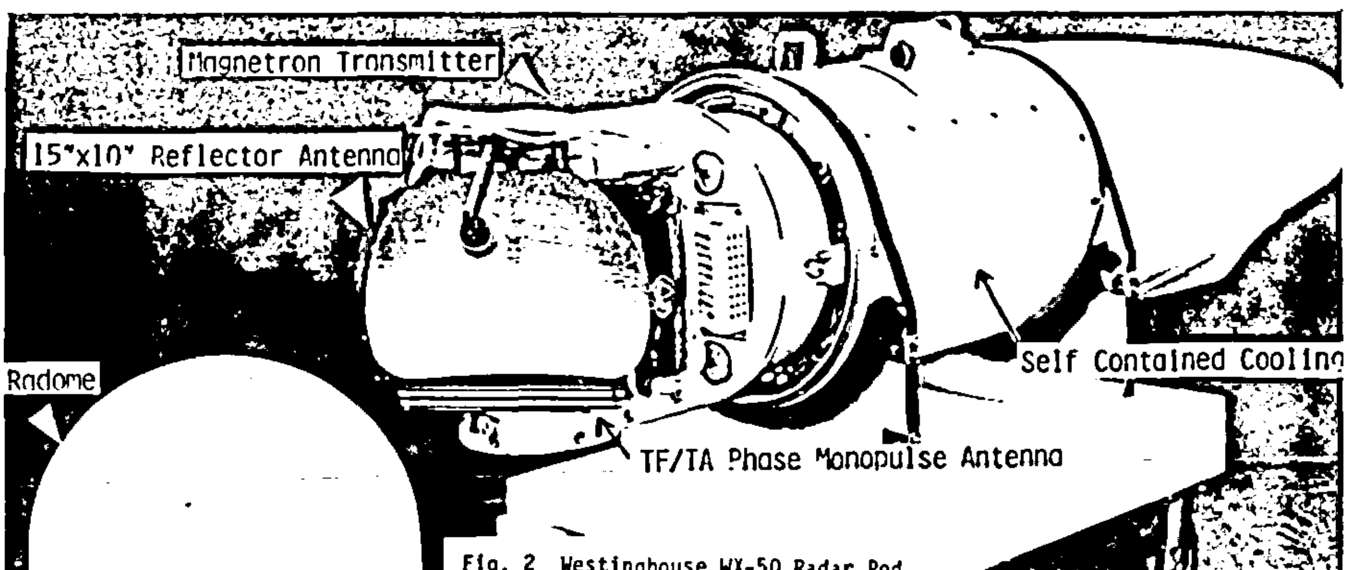

Fig. 2 Westinghouse WX-50 Radar Pod

configurations such as hills and other natural obstacles. These hills were located in the Precision Impact Range Area (PIRA) and north of the Edwards main base complex in the vicinity of California City. The HUD video was recorded on a recorder, and the contour lines were compared with the known terrain profiles to determine their validity. Known terrain was terrain that had been surveyed by helicopter and was overflown during day missions prior to the SSNA A-10 avionics evaluation.

The Edwards local area also provided a unique opportunity to test and evaluate TF/TA radars during land-water transitions. Since the dry lake beds appeared to be water-bearing bodies to radars, the WX-50 TF/TA was tested over Rogers Dry Lake. The performance of the TF/TA was also substantiated by flying to Lake Isabella, about 65 NM northwest of EAFB.

The final objective, evaluating the GM and GMTI modes of operation, was limited to observing appropriate targets of opportunity (e.g., known terrain features for GM and moving vehicles for GMTI) to verify their capabilities.

The TF/TA radar demonstrated acceptable performance to accomplish the SSNA flight test program. The TFB provided the pilot with terrain following commands to fly the low altitude flight profile. For example, Figure 3 illustrates a profile when the pilot flew the TFB over the ESR at an average speed of 275 KIAS and Set Clearance Plane (SCP) of 300 feet above ground level (AGL). The uppermost thick line is the aircraft barometric altitude with respect to the elapsed time after the start of the run. The thinner line which closely follows the barometric altitude line is the A-10 position as tracked by the FPS-16 TSPI radar. The line below the FPS-16 aircraft tracked profile is the barometric altitude minus the radar altitude. That profile closely resembled the terrain which the aircraft overflew. Haystack Butte was on the easternmost portion of the ESR and is represented by the two peaks on the terrain profile. The bottom line is the radar altitude with respect to elapsed time. It represents the aircraft height above the ground and is primarily used to determine how well the TF/TA radar kept the aircraft at the SCP.

As shown in Figure 3, the TF/TA radar commanded the pilot to fly the aircraft between 370 and 380 feet AGL or about 70 to 80 feet above the SCP. During the overflight of Haystack Butte, the radar altitude dropped slightly below the SCP over the butte and ballooned or increased past the butte. For example, in this run, a 700-foot AGL radar altitude ballooning effect was observed east of Haystack Butte. This particular run was representative of the TF/TA performance during straight and level runs at different SCPs and airspeeds.

This cresting or SCP violation and subsequent ballooning characteristic resulted from two anomalies. First, the radar receiver was blanked for

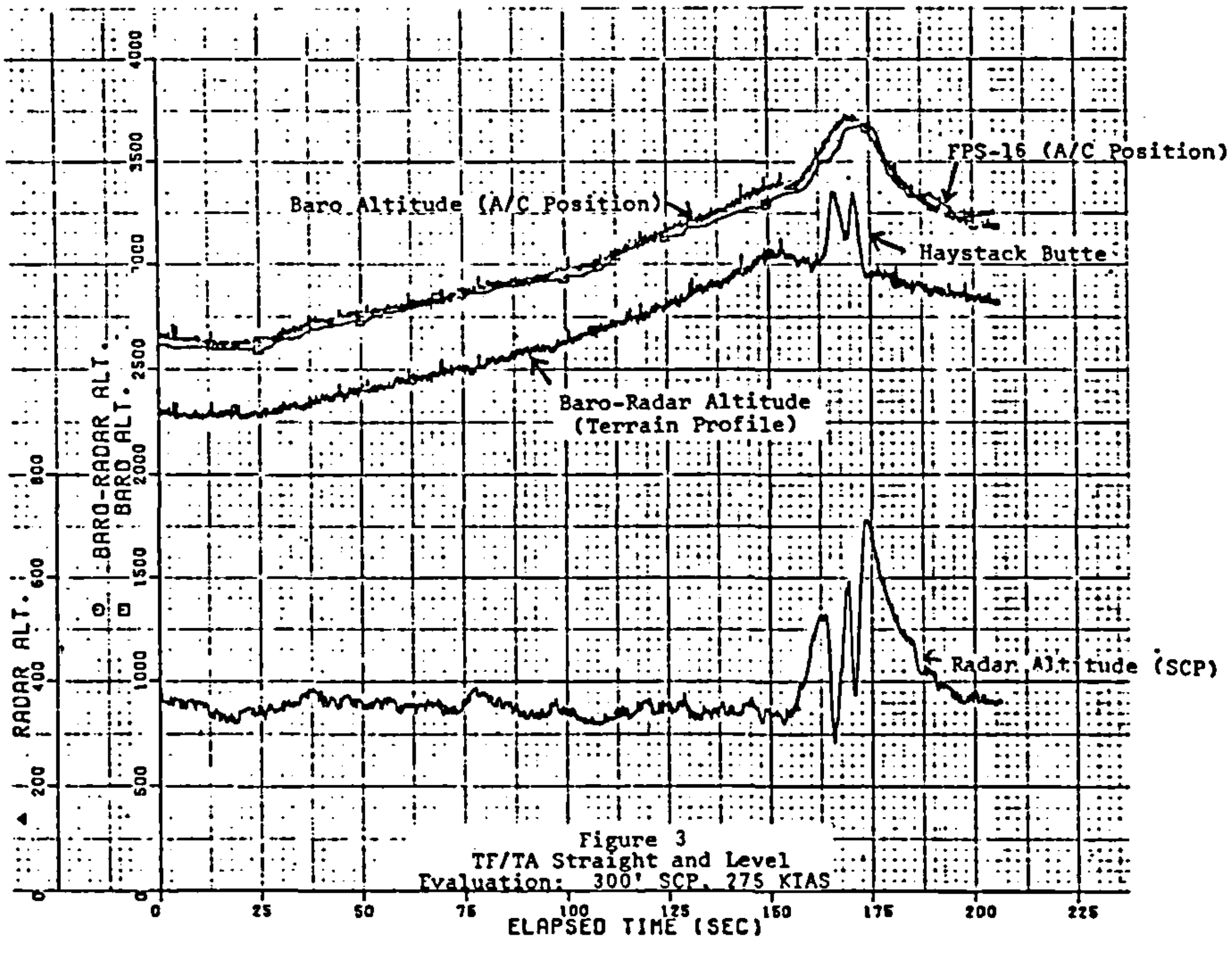

Figure 3
TF/TA Straight and Level
Evaluation: 300' SCP, 275 KIAS

3.6 microseconds to avoid processing false returns. This blanking period caused an 1,800-foot dead zone or blind area, and the TFB never cued on the highest point on the butte once that point entered the dead zone. The result was a violation below the SCP. Second, with the aircraft in a nose up attitude when cresting the peak, the TFB disappeared off the HUD because the terrain beyond the peak was below the maximum look-down angle of 10 degrees. Figure 4 diagrams what happened when the terrain beyond a ridge or peak was below the 10 degree look-down angle. The peak was within or inside the 1,800-foot dead zone, but since there was no terrain for the TF/TA radar to cue on, the TFB would disappear before the aircraft crossed the highest point.

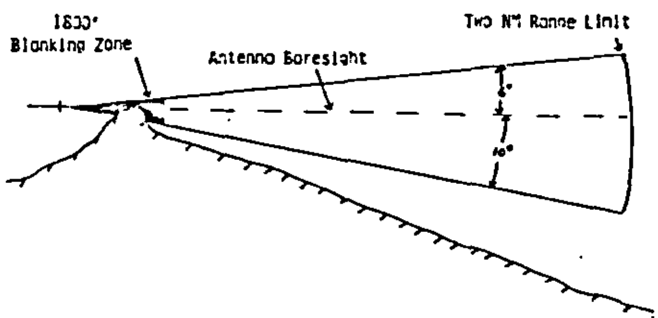

Fig.4 Lack of Ground Returns in Nose Up Attitude

On the other hand, if the highest point was within the 1,800-foot dead zone and the TF/TA radar processed radar returns from terrain within its 2 NM range limit as in Figure 5, the TFB would actually command the pilot to fly down to the terrain the radar had detected prior to flying over the highest point. To compensate, the pilot held the aircraft at the pitch attitude last indicated by the TFB before the TFB disappeared and counted until an appropriate amount of time (ie., 5 seconds) had elapsed or when the radar altimeter showed a large increase. This resulted in the aircraft ballooning east of Haystack Butte.

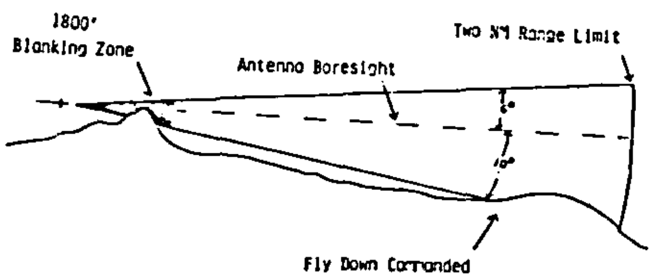

Fig. 5 TF/TA Radar Response to Terrain While Ridge is Within Dead Zone

These SCP violations and ballooning deficiencies were recognized early in the SSNA test program. As a result, pilots and engineers agreed the minimum SCP would be 500 feet over mountainous terrain, 300 feet over obstructed terrain, and 200 feet over flat to rolling terrain. Haystack Butte was typical of obstructed terrain and the portion of the ESR before Haystack Butte was rolling terrain.

The turning flight evaluation was conducted over Haystack Butte, and the A-10 was vectored over the butte by Space Positioning and Optical Radar Tracking (SPORT). In Figure 6, the line with the solid box is the aircraft position derived from the cinetheodolite coverage. The jagged line is the barometric altitude, and the smoother line is the FPS-16 aircraft tracking line. The line below these two is the barometric altitude minus the radar altitude and roughly approximates the terrain which the aircraft overflew. The line with the "+" sign through it is a new parameter which reveals the aircraft bank angle. Positive bank angles are for right turns and negative bank angles are for left turns. The bottommost line is the radar altitude. In Figure 6, the aircraft was flown with a 300-foot SCP. The minimum altitude was about 200 feet AGL while flying over the highest point. The TFB did not command the pilot to fly the SCP over the butte for two reasons. The first relates to the 1,800-foot dead zone mentioned previously. The other reason stems from the fact the TFB was cued to terrain beneath the A-10 ± 2.5 degrees about the instantaneous ground track. Initially, the butte was outside of the azimuth coverage of the TF/TA radar. As the aircraft moved closer in, the butte was either only momentarily within the azimuth coverage of the TF/TA radar or never within the azimuth coverage before moving inside the 1,800-foot dead zone. Figure 7 shows the ground track profiles for right bank angles of 15, 30, and 45 degrees. At 15 degrees the butte was outside of the 1,800-foot dead zone. At 30 degrees, the butte was only momentarily within the TF/TA coverage, whereas the butte was not seen at a 45 degree bank angle.

The usefulness of the TF/TA contour lines was questionable although the idea behind the contour lines was good. The WX-50 TF/TA contour lines exhibited large inconsistencies from sweep to sweep. The contour lines were based on radar energy received at the radar receiver with a preassumed signal to noise ratio (SNR). If the actual SNR was lower than the preassumed value (i.e., smooth, flat terrain) the contour lines would actually be placed higher than the terrain at 1 and 2 NM than was originally intended. However, if the radar received a higher SNR from a portion of the terrain (i.e., vertical development), the contour lines were accurately placed over the terrain at 1 and 2 NM for that portion of the terrain. The result was a perceived terrain inversion. For example, Figure 8 shows representative contour lines over hilly terrain which can be seen in the background with the FLIR. The contour lines gave the pilot a false indication a valley existed where there was actually a mountain. The contour lines did not provide consistently valid information to the pilot.

A deficiency was also discovered when the TF/TA radar was tested over Rogers Dry Lake and Lake Isabella. When the TF/TA radar was emitting RF energy toward the lake during the land-water transition or while flying over the water, the amount of energy reflected back to the receiver decreased substantially and the pilot was given a gradual fly-down command. The pilot compensated by following the horizon line instead of the TFB. As the aircraft approached land during the water-land transition, the pilot was given a proper fly-up command.

The final objective was testing and evaluating the GM and GMTI modes of the TF/TA radar. A good GM display was obtained out to eight to nine miles.

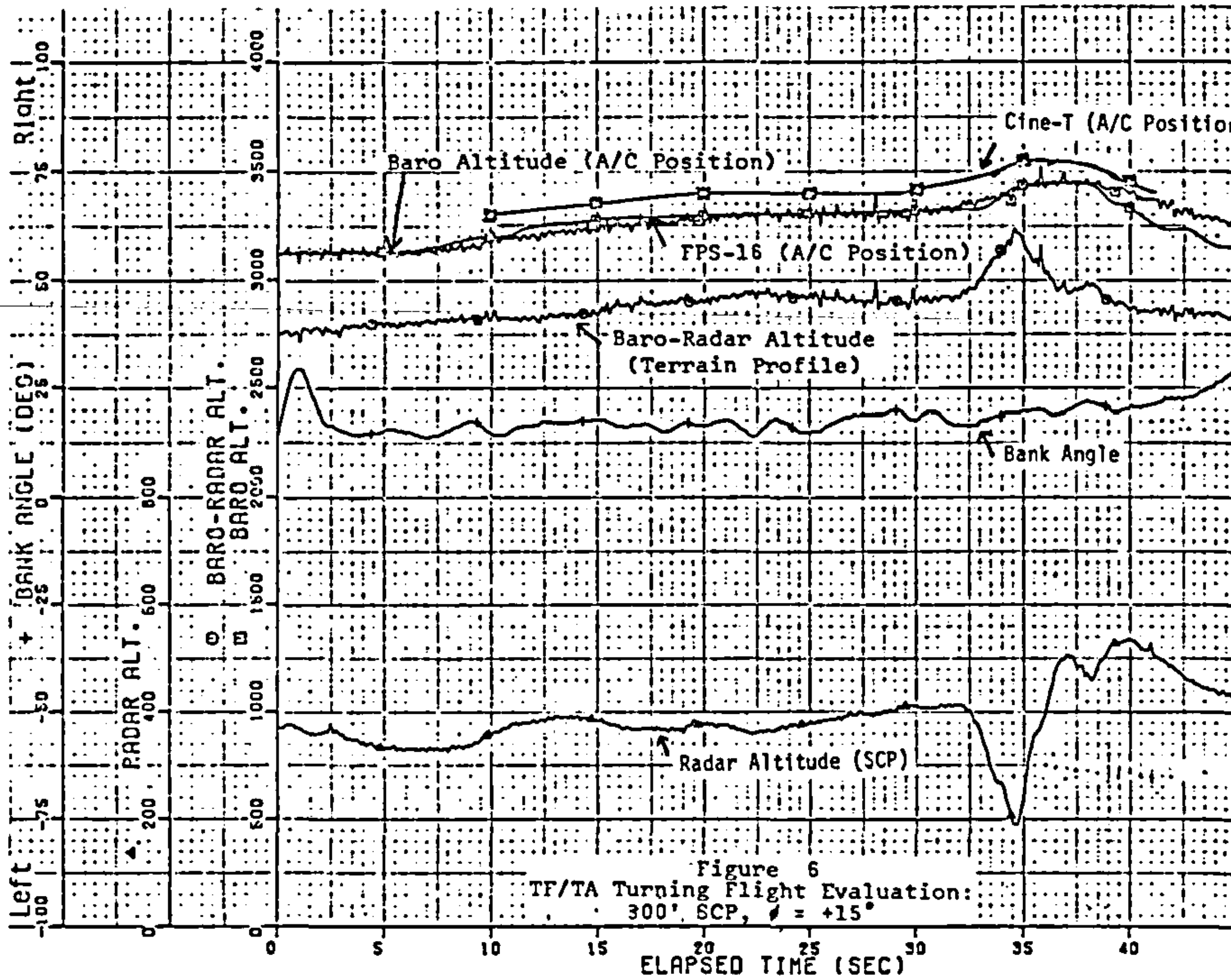

Figure 6
TF/TA Turning Flight Evaluation:
300' SCP, ϕ = +15°

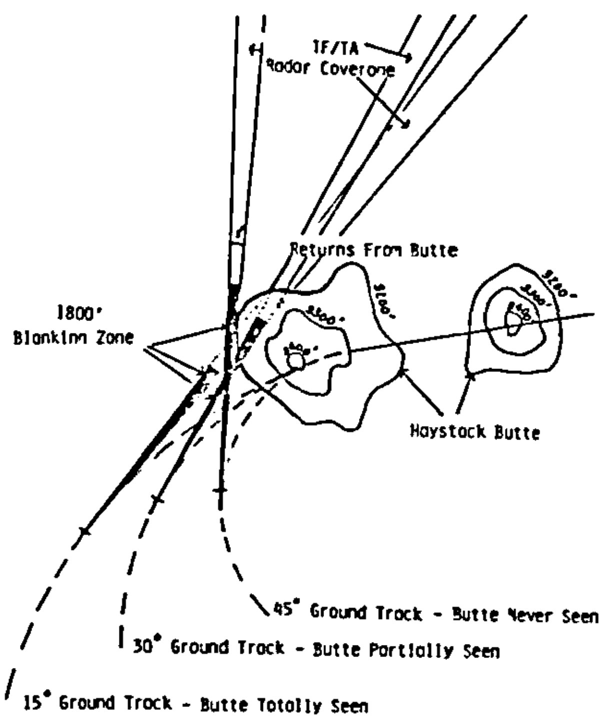

Fig. 7 A-10 Ground Track for Varying Bank Angles

Exact distances varied with the terrain. The GM function worked well for highlighting mountains, valleys, and cultural features within the radar's 60 degree azimuth scan. The GMTI, however, did not perform as well. For example, vehicles moving along paved highways were not consistently detected. The GMTI was also not reliable for detecting vehicles in the PIRA. The targets did not show up on every sweep, so if the pilot did not look at the GMTI during one of the sweeps that displayed a target, the target would not be noticed. Furthermore, terrain returns were also displayed in the GMTI mode. The final deficiency discovered with the GMTI was in the aural alert. The aural alert was overly sensitive and would be set off by false targets or targets too weak to be easily detected on the video display. To compensate, the pilots flew with the aural alert off.

Testing the TF/TA radar depended a great deal on the accuracy of the Radar Altimeter (RA). This was primarily because the height of the aircraft above the ground over all types of terrain was recorded off the RA to determine how well the TF/TA radar commanded the pilot to fly the SCP. Furthermore, the RA provided valuable situational awareness information to the pilot at night, as well as a Low Altitude Warning (LAW). Therefore, the accuracy of the RA was an essential system to test and evaluate.

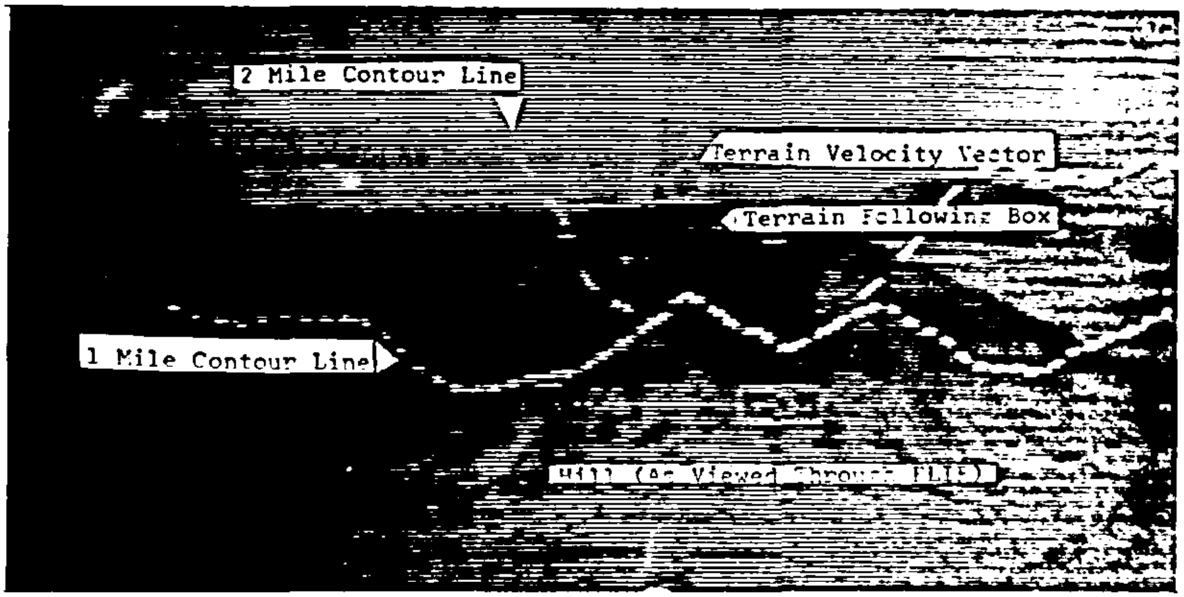

Fig. 8 Inverted Two-Mile Contour Line

Radar Altimeter

The Honeywell AN/APN-194 (V) pulse radar altimeter was installed on the SSNA A-10 to provide AGL altitude to the crew and associated avionics. The system consisted of a receiver-transmitter (RT-1015/APN-194(V)), two antennas (AS-2741/APN-194(V)) located in the horizontal stabilizers, and a height indicator (ID-1879/APN-194(V)). The radar altimeter provided height information between 0 and 5000 feet AGL. The RA provided altitude information in two forms: analog and digital. Analog output signals drove the height indicator mounted on the instrument panel in the rear cockpit. Analog signals provided altitude information to the HUD, to the WX-50 radar, and to the data recorder. The WX-50 radar used the analog radar altitude signal to adjust antenna tilt in the pitch axis. The digital data was reformatted into MIL-STD-1553A format and transmitted to the INS for calculating update positions using the FLIR and to the Programmable Display Processor (PDP) for writing the radar altitude data on the HDDs.

The RA also had the capability for the pilot to set a LAW. When the actual terrain clearance was below the desired value, the low altitude audio generator sent a signal to the HUD and PDP which caused the radar altitude displays to flash.

The evaluation of the radar altimeter was conducted with four objectives in mind. These were:

1. Determine altimeter accuracy.
2. Evaluate performance during land-water and water-land transitions.
3. Determine the bank and pitch limits.
4. Verify the accuracy of the LAW cue.

To satisfy the first objective, the RA was first flown past a tower used for airspeed and altimeter calibration and then over the south end of Rogers Dry Lake with FPS-16 radar and cinetheodolite tracking. The dry lake was ideal for a RA evaluation due to its nearly constant known mean sea level altitude and good RF reflectivity. The known dry lake altitude was then subtracted from the tracking data which is the aircraft altitude above sea level to give an aircraft altitude AGL.

RA performance during land-water and water-land transitions was also evaluated during flights over Rogers Dry Lake and Lake Isabella. The aircraft was flown over relatively level transition points at a constant altitude while monitoring altimeter performance.

The aircraft was flown at fixed bank angles up to 45 degrees over the dry lake with 1500 feet per minute (FPM) descent and ascent rates between 1000 and 100 feet AGL. Increasing bank, pitch, and dive angles were flown to determine the attitude limits of the RA.

The accuracy of the RA LAW cue was verified by setting the cue to 150 feet AGL and then descending until the LAW aural warning was activated. The altitude at which the warning went off was determined from altitude data obtained from the RA and cinetheodolites.

Results from the tower flybys indicated that generally the displayed radar altitude on the HUD was lower than the actual altitude. The maximum error in the front cockpit was about 30 feet low at actual altitudes of 90 feet AGL and 440 feet AGL. The minimum error was 7 feet high at an actual altitude of 343 feet AGL for both the front and rear cockpits. The difference in altitude between the front and rear cockpits was due to software errors in the HUD where the front cockpit radar altitude was displayed. The accuracy of the RA was also verified over the south end of Rogers Dry Lake. The difference between the cinetheodolite-derived aircraft MSL altitude and the terrain altitude of 2270 feet was the aircraft height above the ground, and that altitude was compared with the radar altitude to determine the RA accuracy.

Figure 9 is the altitude profile of the aircraft over Rogers Dry Lake. The initial altitude was 1000 feet AGL and the aircraft was flown with a 30 feet per second descent rate at 275 KIAS. The descent was terminated when the LAW was activated and the aircraft was flown back up to 1000 feet AGL. In Figure 9, the narrow line with the filled-in box

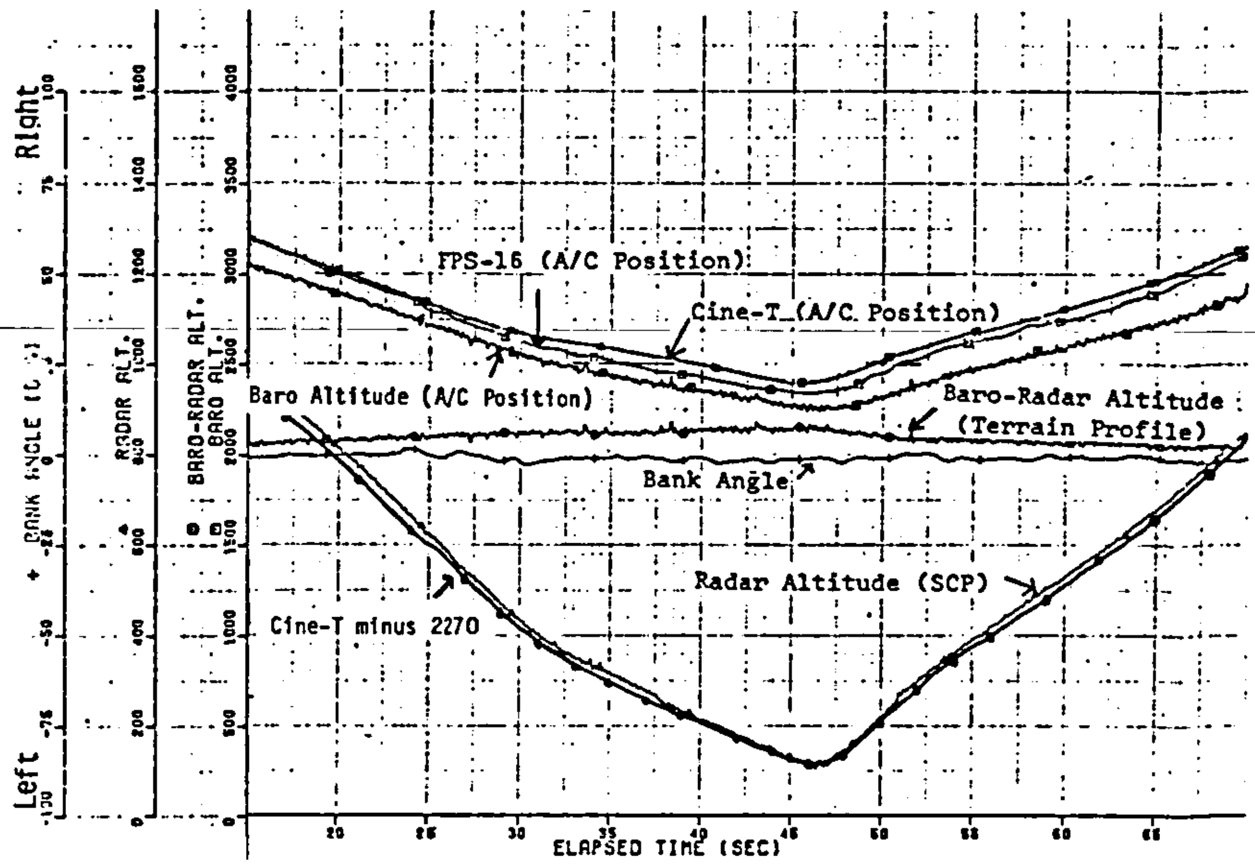

Fig. 9 Radar Altimeter Evaluation: 275 KIAS, $\phi = 0^{0}$

symbol (-●-) placed throughout the line is the cinetheodolite derived aircraft altitude MSL. The heavier line close to the cinetheodolite line is the aircraft barometric altitude. The terrain profile is represented by the next line which is the barometric altitude minus the radar altitude. The bottommost line is the radar altitude. The figure reflects radar altitude accuracies typical of the RA over level terrain with no bank angle.

RA accuracy in turning flight was verified over the south end of Rogers Dry Lake with various bank angles. Figure 10 shows the altitude profiles for a run performed at 275 KIAS initially at 500 feet AGL and the run was terminated when the LAW was activated. Since the bank angle was approximately 20 degrees left, the A-10 flew a spiraling profile first downward then upward at about 1200 FPM. The "-" sign denotes the aircraft was in a left turn. The data revealed in a 20 degree left bank the RA accuracy was within 10 feet. Additional tests indicated as bank angle increased, the accuracy in the RA did not appear to be a function of bank angle up to 45 degrees.

During the simulated land-water/water-land transitions over Rogers Dry Lake and the actual land-water/water-land transitions over Lake Isabella, the RA accuracy did not appear to be degraded. However, the actual height above the Isabella Lake surface was not known since cinetheodolite coverage was not available.

The RA bank and pitch limits were also verified over the south end of Rogers Dry Lake. The right bank limit results are shown in Figure 11. The bank angle was slowly increased until the radar altimeter broke lock. The maneuver was repeated to obtain additional data. Figure 11 reveals breaklock occurred at a bank angle of about 70 degrees. In a left bank, the breaklock angle was close to that of the right bank angle. However, for left bank angles the altitude error was about 260 feet at breaklock, and only 80 feet for positive bank angles. In both cases the deviation was within specifications up to 45 degrees.

Tests to determine the pitch limits of the RA revealed that in a pitch up attitude breaklock occurred at 75 degrees and the altimeter provided a correct reading for pitch angles all the way up to the breaklock point. A breaklock could not be induced for negative pitch angles up to 35 degrees.

The RA LAW accurately generated an aural tone at 150 feet AGL for both straight and level and turning flight. Since the radar altitude display on the HUD was somewhat lower than true altitude it was not uncommon to see altitudes of 130 to 140 feet AGL when the tone was activated. That discrepancy was not seen in the rear cockpit since the height indicator error was not as pronounced as in the front.

Generally the RA performance was found to be both accurate and reliable. The RA interacted well with the rest of the aircraft avionics systems with the exception of the radar altitude on the HUD.

The TF/TA radar and RA were used with the Forward Looking Infrared Receiver (FLIR) system to enable the pilot to fly at low altitude at night. The FLIR was also used for target detection and recognition. Therefore, the FLIR system was an important item to test and evaluate.

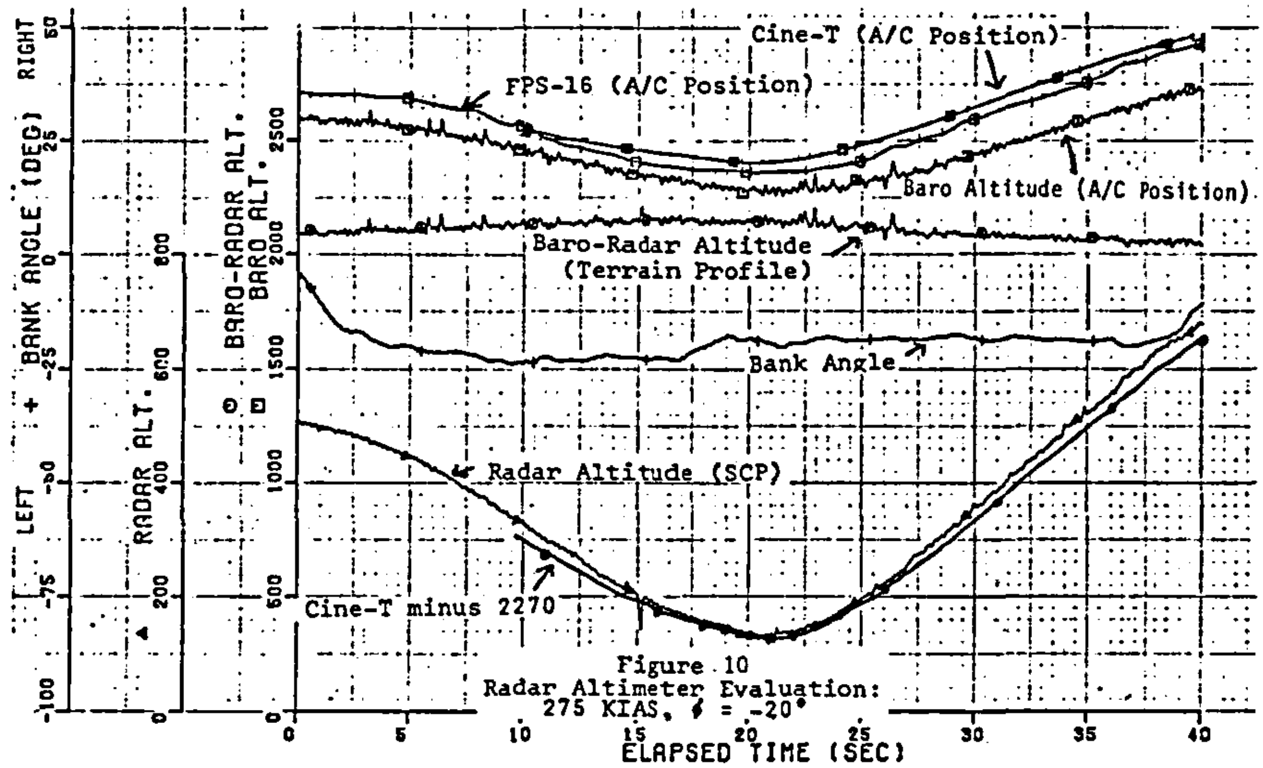

Figure 10
Radar Altimeter Evaluation:
275 KIAS, ϕ = -20°

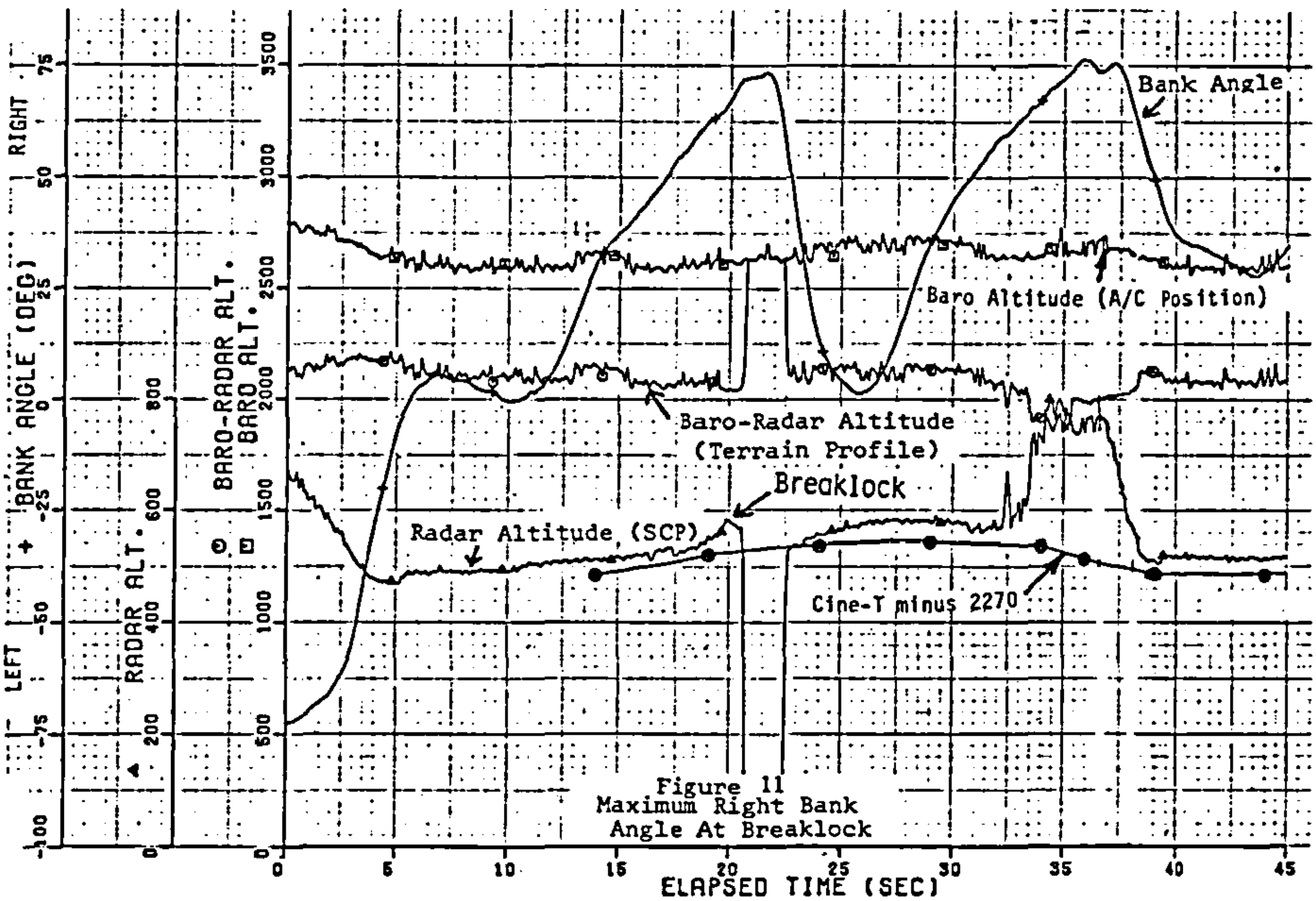

Figure 11
Maximum Right Bank
Angle At Breaklock

Forward Looking Infrared Receiver System

The Forward Looking Infrared Receiver (FLIR) system was a modified AN/AAR-42 Infrared Receiver Set (IRRS). The passive FLIR sensor operated within the 7.6 to 11.75 micrometer wavelength spectrum and provided a raster video presentation to the pilot.

The output from the FLIR was a differential composite video signal conforming to EIA-Standard RS-170 with an aspect ratio of 4:3. The FLIR video was displayed on a 525 line MFD or HUD and provided a real-time image of the terrain and targets being scanned. The FLIR had two selectable Fields Of View (FOV): a 12 by 16 degree FOV used for navigation and target acquisition, and a 3 by 4 degree FOV used for target recognition and identification. The FLIR had a Field Of Regard (FOR) of 20 degrees in azimuth and +5 to -35 degrees in elevation with respect to the aircraft Armament Datum Line (ADL).

The FLIR system operated as a parallel processed scanner. Infrared radiation (IRR) entered through the polycrystalline germanium front window. The focal and FOV-focus optics section focused and magnified (narrow FOV only) the IRR and projected it onto a two-sided scanner mirror. There were 180 detectors arranged in a straight line with a space between each detector. Pre and postamplifiers and Light Emitting Diodes (LEDs) were associated with each detector. The preamplifiers amplified the detected energy and the postamplifier controlled the intensity of the LEDs. The LEDs' emitters were arranged in a similar manner to the detectors. After the first sweep, the mirror reversed direction and the angle changed slightly corresponding to the space between the detectors and LEDs, so that another 180 lines were detected and reproduced. These 180 lines filled the spaces produced by the arrangement of detectors and LEDs on the first sweep. Thus 360 total lines were impressed onto the video camera. The video camera was capable of producing a standard 525 line RS-170 video output with 480 lines of useful information.

The flight test and evaluation of the FLIR system centered primarily around three objectives. These objectives were:

1. Determine the FLIR detection distance as a function of spatial frequency under flight conditions.
2. Determine the degradation in detection, recognition, and identification range from HUD video due to the HUD resolution and other related factors.
3. Estimate target detection, recognition, and identification based on the detection distance curves from number 1 above.

To satisfy these objectives, specific test procedures were followed. FLIR detection distance, as a function of spatial frequency, was determined by flying against a four bar active IR target on the PIRA. The target was configured with a four bar pattern, each bar four feet wide and 30 feet high, for a 7.5 to 1 aspect ratio. The spacing between the bars was also four feet. The temperature of the bars was varied so that the temperature difference (ΔT) between the heated bars and ambient conditions ranged from about 2 to 30 degrees C. The temperature of the IR target was determined by using a calibrated AGA 780 Thermovision FLIR System which had approximately the same spectral response character-

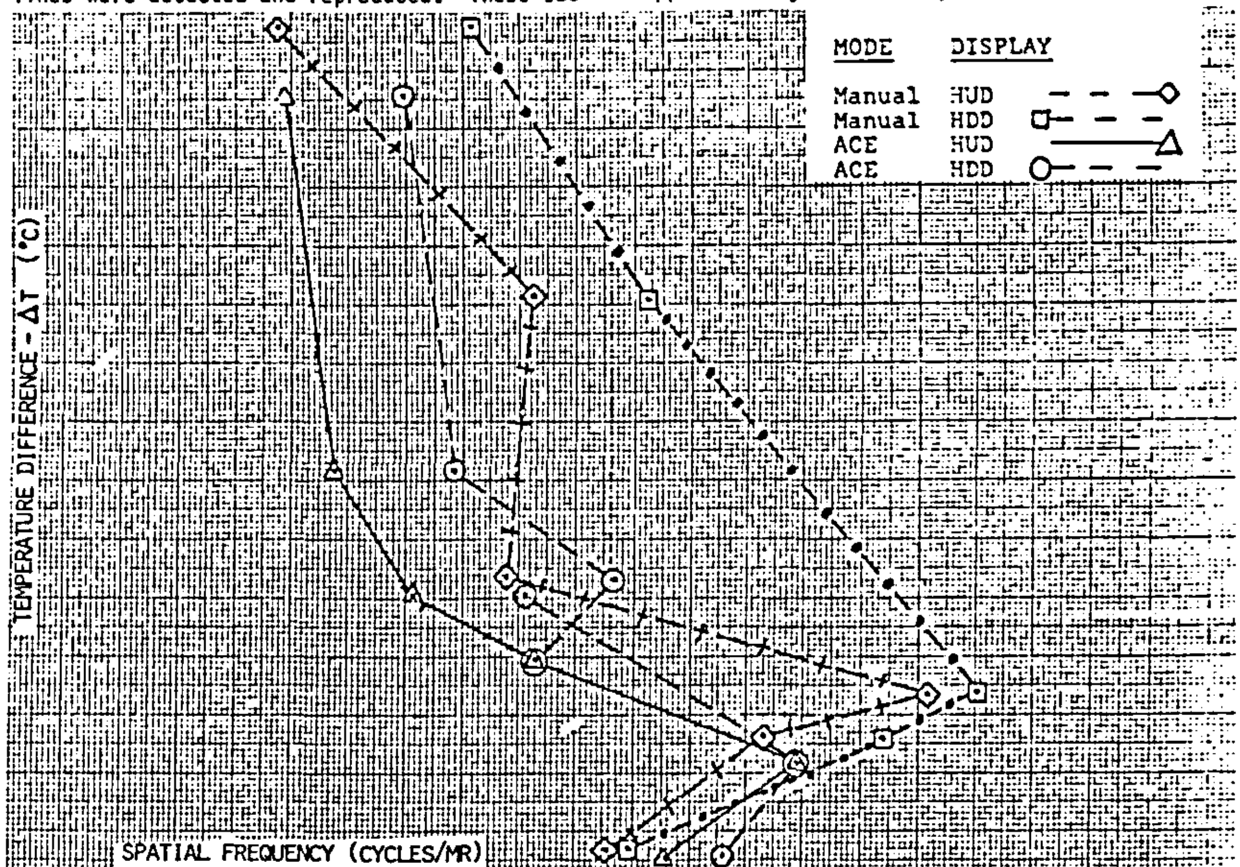

Fig. 12 FLIR NFOV Resolution

istics as the aircraft FLIR. As the aircraft flew towards the IR target the pilot called the time when the four bars could be resolved in the HUD. The safety observer did the same when he resolved the four bars on the HDD in the rear cockpit. TSPI was used to determine the slant range between the aircraft and the IR board at those times when the four bars could be resolved. The target temperature then was raised several degrees on each succeeding pass. The FLIR gain and level settings and HUD contrast and brightness were recorded. Performance between the manual setting and Automatic Contrast Enhancement (ACE) modes was compared. Display quality was qualitatively evaluated by the pilots. Comparisons were made between the HUD and MFDs and rear cockpit displays. The aircraft altitude was maintained at approximately 500 feet AGL and airspeed was 250 KIAS. The detection, recognition, and identification distances for a given target were computed by using the resolution test results and the FLIR transfer function.

The resultant spatial frequency (fs), as a function of ΔT for the FLIR, is shown in Figures 12 and 13 for the NFOV and WFOV, respectively. Since the resolution through the HUD in the front cockpit was different from the resolution through the Hartman HDD in the rear cockpit, the curves are annotated between the head-up and head-down displays.

The curves have been corrected for the loss of IR energy through the atmosphere via the LOWTRAN 5 computer program . Since the front and rear seaters resolved the four bar targets at different times, the atmospheric correction factors were computed for both the front and rear seats.

The scales have been removed from the graphs so as to declassify them. The results from the FLIR resolution test generally contradict theory. The flight test results revealed that the fs slightly decreased as the ΔT increased. This difference in the shapes of the curves between the theoretical results and flight test results was attributed to the FLIR gain and level and HUD and HDD brightness and contrast settings under flight test conditions. For the laboratory-generated FLIR curves, the gain and level was readjusted for each ΔT to optimize the FLIR detectors' capability to differentiate thermal energy. That optimization was not attainable by the pilots during flight test. The FLIR picture was manually tuned for each ambient thermal condition without the target board in the FOV. As the target temperature was increased, this provided a less and less suitable gain and level setting for resolution of the target pattern.

Other results indicated that the spatial resolution was better in the NFOV than in the WFOV. The difference in spatial resolution between the manual and ACE modes in the WFOV was minimal, but in the NFOV manual adjustments proved a slight improvement over ACE.

The spatial resolution was slightly better through the HDD than the HUD due to losses associated with the HUD optics. There were some occasions, however when the resolution distances were almost the same.

The FLIR modulation transfer function was used to generate the probability of detection, recognition, and identification curves for both the HUD

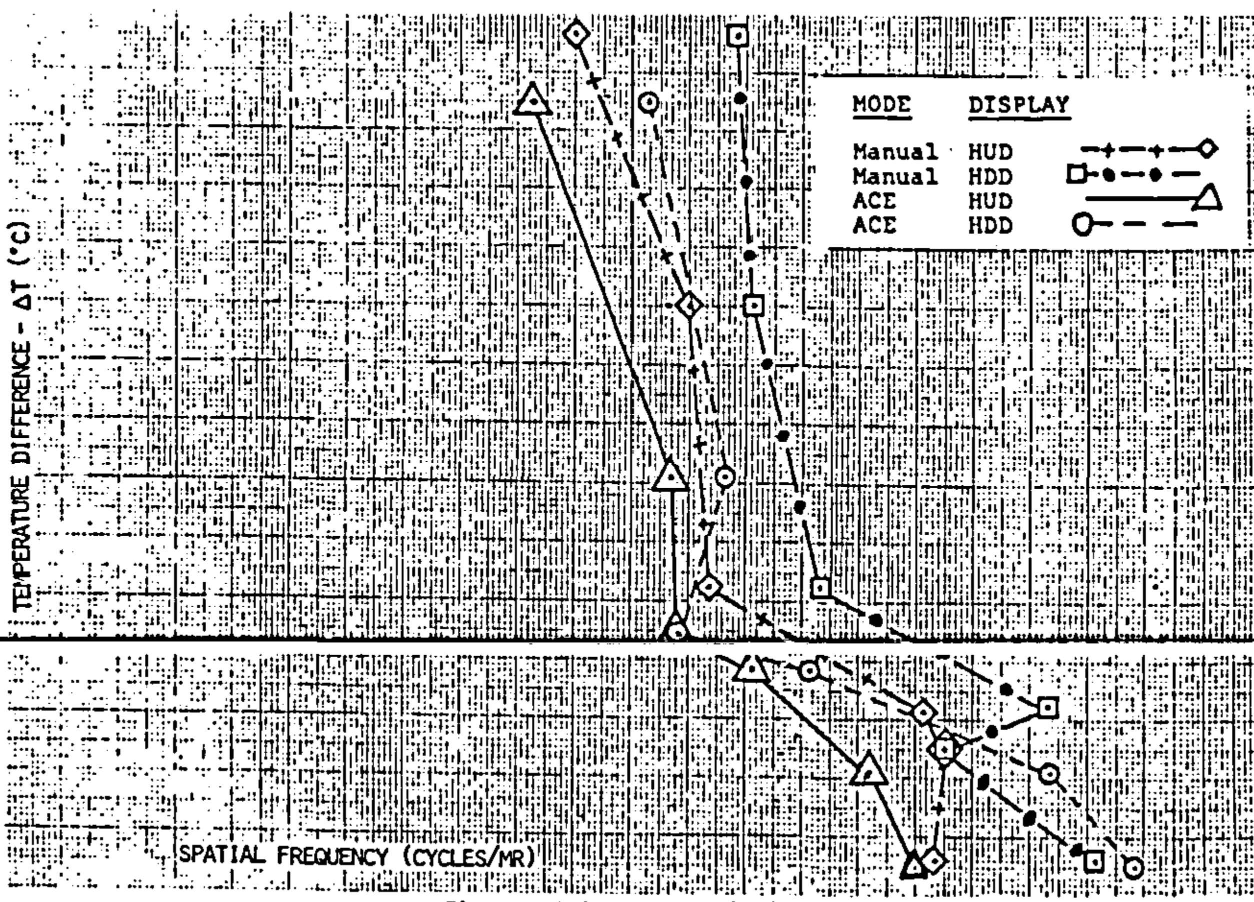

Fig. 13 FLIR WFOV Resolution

HDD for the two FOVs.

The horizon stabilized snap-look function was useful for providing look-into-turn capability. The slew rate was approximately 21 degrees per second. There were two deficiencies with the FLIR snap-look: a momementary dip in elevation when returning from snap-look and pitch nulling. When returning to the center FOV, the FLIR momentarily dipped in elevation. The pitch nulling occurred when the snap-look was performed and a change in aircraft pitch attitude followed. The snap-look would keep the pitch attitude shown when the snap-look was initiated and defeated the natural movement of the horizon in the picture during pitch changes. This situation was found to be undesirable and was overcome by frequently alternating between boresight and snap-look views.

By using the point track function the pilot designated a point in the FLIR and the FLIR would maintain track as long as the point stayed within the FLIR's FOR. The pilots found little use for this feature.

In the boresight mode the FLIR presentation on the HUD did not register correctly with the real world. The FLIR presentation appeared slightly elongated in the vertical axis with respect to the real world. This registration difference was approximately 10 mr at the top and bottom of the HUD FLIR presentation with respect to the real world. However, registration with respect to the real world was adjusted to be accurate at the FLIR center for gunfire.

The FLIR presentation also had a tendency to bloom during banks. This was attributed to the inability of the FLIR detectors to respond to changing thermal conditions between the sky and ground as the detectors were scanned throughout the FOV.

Recommendations

The calibration and resolution of the SSNA A-10 night attack avionics produced several recommendations. Generally, the SSNA A-10 avionics suite demonstrated sufficient capabilities to conduct the human factors evaluation. All the test objectives were met for the TF/TA radar, the radar altimeter, and the FLIR subsystems. The specific recommendations resulting from this evaluation are presented for these three subsystems only. These recommendations should serve as lessons learned for future night attack avionics testing.

The recommendations are divided below into two categories. The first is a mandatory correction category for items with serious deficiencies that adversely affect critical operational capabilities or safety of flight under uncontrolled conditions. The second is a desirable correction category for items that would improve the operational capability of future night attack aircraft.

Mandatory Corrections

1. Future TF/TA radars should be mechanized to reduce or shorten the aircraft blind zone.

2. TF/TA radars should have large azimuth coverage to accurately overfly the terrain in turns.

3. TF/TA radars should incorporate a wide elevation coverage or have some type of pitch compensation to help prevent ballooning over peaks.

4. TF/TA systems should provide automatic control from the radar altimeter over areas of poor radar returns.

Desirable Corrections

5. TF/TA contour lines are a useful feature but must be reliable.

6. GMTIs should consistently detect and retain display of moving targets from sweep to sweep.

7. While in the GMTI mode, TF/TA radars should detect only moving targets.

8. FLIR snaplooks should be smooth with no dips in elevation during snaplook.

9. Snaplooks should show actual movement of the horizon.

10. FLIR presentations should register correctly with the real world.

11. FLIR detectors should have compensation for changing thermal conditions between the sky and ground to prevent blooming.

LEAR FAN MODEL 2100 EMERGENCY EGRESS SYSTEM

by
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Abstract

This paper describes the flight crew emergency egress system developed for the Lear Fan Model 2100 developmental flight test program. Because of the extensive use of composite materials for primary structure, a radically unconventional propulsion system and unique "Y" tail, pusher propeller configuration, conventional general aviation provisions for flight crew escape from a damaged or out-of-control airplane were considered to be inadequate. After establishing design objectives and criteria and considering several alternative systems, a rocket extraction system was selected, designed, tested and installed in prototype airplanes.

Introduction

The Lear Fan Model 2100 is a pressurized, corporate, turboprop airplane designed to carry a maximum of ten people including flight crew. The airplane will be capable of operating at altitudes as high as 41,000 feet and cruise at speeds up to 350 knots. Twin PT6B-35F turboshaft engines flat rated to 650 horsepower each are imbedded in the aft fuselage and drive a single, four bladed Kevlar pusher propeller via independent, graphite-epoxy drive shafts and sprag clutches connected to an aft fuselage mounted transmission. The airplane is configured with a low wing and "Y" empennage (refer to Figure 1). Extensive use is made of graphite-epoxy lay-ups for the primary structure.

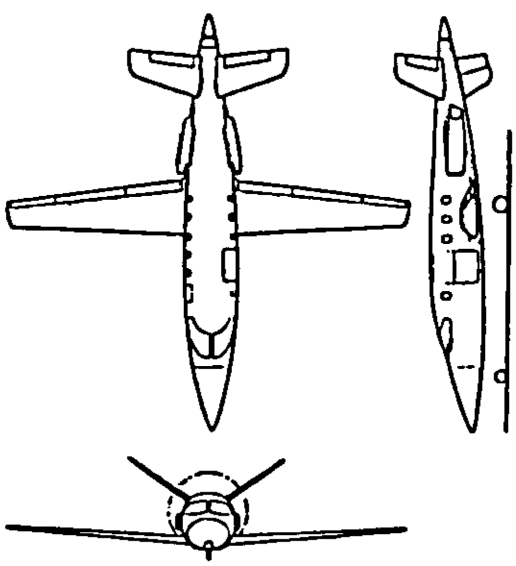

Figure 1 - Airplane Three View

*Manager, Flight Operations

Because of the use of a radically unconventional propulsion system coupled with the extensive use of state-of-the-art composites and bonding for primary structural assemblies, the likelihood of an emergency egress situation arising was considered to be higher than that associated with the testing of more conventional airplanes. Furthermore, the airplane's "Y" empennage and aft mounted propeller would impair a clean conventional "jump" bailout by the crewmember. Accordingly, a program to develop a reliable egress system was earnestly pursued starting in December, 1978.

Design Philosophy and Criteria

The design philosophy adopted was to provide a realistic system which would cover reasonably foreseeable egress situations as opposed to a system which would hypothetically cover any conceivable egress scenario. Thus a low level of calculated risk was deemed acceptable to permit the realization of a reliable and practical system rather than demand perfection at the risk of ending up with nothing. The following design criteria were established consistent with this philosophy:

- High degree of reliability and safety.
- Independence from ship systems.
- Capacity of up to 3 crewmembers.
- Reasonably light weight.
- Reasonably low cost.
- Transferability among prototype airplanes.
- Minimum impact on airplane structure.

Alternative Systems Considered

Ejection seats were considered and rejected because it would have been difficult to create suitable openings in the airframe structure for up to three seats. A possible requirement for seat translation and ejection sequencing would have resulted in a complex system reducing relability. Finally, the seats would have been unacceptably heavy and expensive.

An explosive system capable of removing the cabin door, propeller and left side of the empennage was considered. This scheme was rejected because a loss of a major portion of the empennage could easily render what might have been a stabilized escape situation uncontrollable and because inadvertent actuation of the system could be catastrophic.

Explosive severance of the fuselage aft of the flight crew stations was also considered, but was rejected because of the problems associated with cutting the control cables, wiring and plumbing which are routed through the fuselage. This concept, like the previous one, could also result in an unstable escape situation and inadvertent actuation of the system would be catastrophic.

By March, 1979, interest had focused on a version of the Yankee system originally developed by Stanley Aviation and now owned by Stencel Aero Engineering Corporation. Stencel proposed the Aircrew Walk Around Yankee (AWAY) system, an uncomplicated system which had been previously developed for the A300B Airbus flight test program. The system met all the previously mentioned criteria and was clearly capable of adaptation to to the Lear Fan.

AWAY System Description

This system consists of a crewmember parachute, torso harness and helmet assembly, a tractor rocket and pendant lanyard assembly, an initiator assembly, a seat restraint release system, and a launcher stanchion structure (refer to Figure 2). A seat mover system is also installed for high risk testing. The system may be configured with up to three rockets.

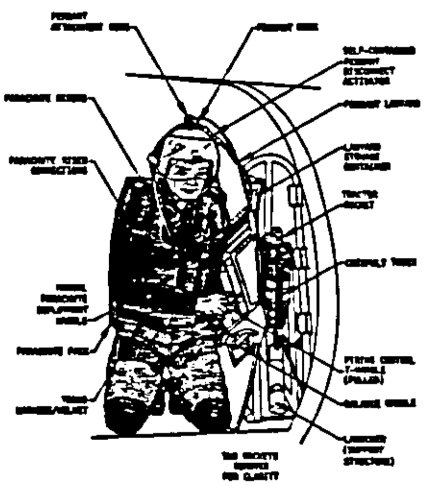

Figure 2 - AWAY System Components

The parachute is a back pack of French design (Etudes Et Fabrications Aeronautiques, Type 384P). The canopy is constructed of nylon, is 22.3 feet in diameter and capable of opening at speeds up to 270 mph. The parachute is stabilized by varying gore (panel) porosity among the 24 gores. Forward motion is provided via canopy drive slots and the chute is steerable using control lines attached to turn slots in the canopy. The parachute is designed to produce a landing rate or approximately 19 feet per second with a 176 pound load.

The parachute is attached via two risers to a Navy MA-2 Torso Harness suit manufactured by Pioneer Recovery Systems and modified by Stanley Aviation, Inc. The harness distributes parachute canopy loads to the crewmember's torso uniformally via straps and webbing. The harness comes in 12 sizes to ensure proper fit. Quick releases connect the parachute risers to the shoulder torso straps permitting quick canopy release after landing.

The helmet assembly consists of an APH-3 shell manufactured by Sierra and modified by Stanley, a visor with an automatic lowering provision, an integral radio headset and a pendant ring assembly. Three torso harness pendant lanyards pass through alignment clips attached to the sides of the helmet shell and connect to the pendant ring which is located on top of the helmet. The pendant ring is designed to separate from the helmet shell when a force of approximately ten pounds or greater is applied to the ring. Another lanyard, attached to the solid rocket motor case, also connects to the pendant ring and transfers the rocket extraction forces to the torso harness via the three harness pendant landyards. The helmet alignment clips permit the harness pendant lanyards to apply side loads to the helmet to prevent whiplash. The separation of the pendant ring from the helmet causes the visor to be lowered if it hasn't already been manually lowered.

The tractor rocket is a solid fuel rocket manufactured by Stanley and is capable of producing 2,250 pounds of thrust for 0.5 second decaying to zero thrust in another 0.5 second. The two rocket nozzles are oriented so as to cause the rocket to spin about its longitudinal axis to provide spin stability. The rocket is stowed in a tubular truss stanchion launch assembly which includes deflector shields to prevent the launch blast from contacting the crewmember. The rocket is catapulted from the launcher via an M-53 initiator cartridge that generates gas which pressurizes two catapult tubes attached to the rocket motor case. The gas is also used to automatically unlock the crewmember's five point restraint harness buckle when the system is configured with the seat mover installed. Rocket ignition occurs when the nine foot lanyard connecting the rocket to the crewmember becomes taut and generates approximately 1,000 pounds of force. Stabilizer tubes extend behind the rocket after it leaves the launcher so as to orient it into the relative wind. A self contained pendant disconnect actuator automatically releases the pendant lanyard from the helmet lanyard ring after rocket burn-out.

The total weight of the rocket stanchion assembly with two rockets installed is approximately 110 pounds. The combined weight of the parachute, harness and helmet is 43 pounds per crewmember.

The seat-mover system is in the final stage of development. The system has been designed, parts have been fabricated and assembled and an airplane fit check has been completed. The system has also been bench tested, but some redesign and an end-to-end functional check aboard the airplane remain to be accomplished. The seat mover system is intended to only be installed for high risk flight testing such as flutter and stall characteristics and restricts airplane occupancy to single-pilot when installed. The system consists of a pilot's seat, seat tracks, pilot restraint system, pneumatic cylinder, compressed air bottle**, pulley-cable system, and an actuation control.

**The bottle will be filled with Nitrogen gas.

The pilot's seat and seat tracks were designed and fabricated by Lear Fan. The seat includes a head rest to prevent whiplash and foot stirups are provided to enable the pilot to avoid flailing of the legs. The seat is articulated such that it automatically rotates 90 degrees about its vertical axis as it translates aft thus placing the pilot in a sitting position facing the egress opening at the end of seat travel.

The pilot restraint system consists of crotch, lap and shoulder belts which engage a 5 point buckle manufactured by Kin-Tech. A gas operated automatic release mechanism is incorporated in the buckle and is connected to the rocket catapult initiator system.

The pneumatic cyclinder is manufactured by the Origa Corporation, is of rodless design and has a stroke of 6 feet and incorporates provisions for end-of-stroke damping. Seat translation time intervals very with seat loading, but are nominally in the 1 to 3 second range and may be tailored by varying the operating pressure and volume of the compressed air bottle, neither of which have been finalized at this time.

The stroke of the Origa cyclinder is converted into seat translation via a closed loop cable and pulley system. The Origa cylinder, compressed air bottle and pulley-cable system are installed along the floor of the airplane's cabin adjacent to the pilot's seat and seat tracks and requires the removal of the copilot's seat.

System actuation will be initiated by a control handle located on the seat which will mechanically open a valve on the compressed air bottle.

The weight of the seat mover system has not yet been finalized, but it should not exceed the combined weight of one crewmember and a crew seat.

System Operation

Prior to operation of the AWAY system the pilot actuates a pyrotechnic system which cuts an opening approximately 4 feet square in the left side of the fuselage behind the cabin entrance door and ahead of the wing leading edge. The next steps, time and situation permitting, include notifying air traffic control, setting the transponder to the emergency code and heading the airplane in the direction of sparsely populated terrain and engaging the autopilot if installed.

Operation of the AWAY system begins with the crewmember assuming a position at the egress opening, either by initiating the seat mover system (if installed) or by manually releasing the seat restraint harness and moving with the aid of an overhead knotted rope.

If additional crewmembers are onboard, the pilot would continue flying the airplane in an effort to ensure stabilized flight was maintained while the other crewmembers egressed. Once positioned at the egress opening the crewmember removes one of the hooks at the end of the rocket lanyard from the stowed position on the launcher stanchion. This action also exposes the proper (i.e, associated) rocket initiator "T" handle. The rocket lanyard hook is connected to the helmet ring aided, if necessary, by a mirror installed on the launcher stanchion structure. The crewmember initiates escape by pulling the "T" handle. The rocket orients itself into the relative wind at a trajectory that is upward, outward, and forward relative to the airplane body axes. The extraction takes about 0.25 seconds. Parachute deployment is initiated by an automatic aneroid-timer which pulls the ripcord opening the parachute pack. A pilot chute provides stabilization and extracts the main chute which inflates fully 4.8 seconds after the "T" handle is pulled. In high-altitude escape situations, automatic parachute deployment is delayed until a barometric device indicates less than 12,800 feet pressure altitude, at which point the parachute will deploy automatically. Manual ripcord pull is available at the discretion of the crewmember.

Acceptance Testing

This system was tested in two phases, a static and a dynamic test. Previous extensive tests conducted by Stanley Aviation on the crew escape system used on the Airbus included physiological measurements on an instrumented dummy for extraction and parachute deployment and comments of a live subject who did a number of line stretch tests and made an actual 0/0 extraction. Since all components of the Lear Fan crew escape system except the launching stanchion are identical to those used on the Airbus, the Lear Fan system tests were limited to demonstrating clearance from the aircraft and verifying system function.

The static test employed the crew escape system mounted on a truck bed with a representative fuselage egress opening mockup. A dummy larger in size and weight than 95% of the United States male population, attired in normal flight clothing, torso harness, helmet and parachute assemblies, was positioned in the doorway in a crouching attitude simulating an ejection without the Pilot Seat Repositioning System.

The system was remotely activated and the dummy was extracted and descended under a fully blossomed parachute. Due to a loose helmet fit and to misrigging of the automatic parachute ripcord lanyard, the helmet was lifted momentarily from the dummy's head after exit from the mockup. The cause of the malfunction was easily found, the dummy "survived" the test, and the test was considered successful. Zero-zero capability was not a requirement for the system, and the manufacturer does not claim it, but there is little doubt in view of this test that the system would readily extract and recover an average size individual under static conditions.

The dynamic test was conducted at the Hurricane Mesa sled track (which was at that time owned and operated by Stanley Aviation and has since been acquired by Stencel). Lear Fan provided a mockup of the fuselage center section with the door cutaway which was mounted on a jet sled using a J-57 engine with afterburner (refer to Figure 3).

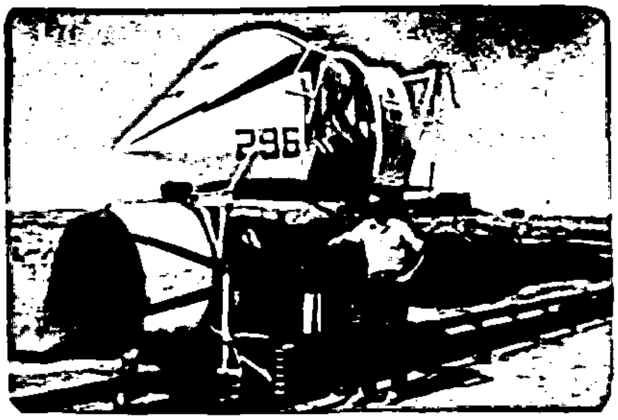

Figure 3 - Dynamic Test Set-Up

A calibration run was made to determine that the required speed could be attained and that all launch and data acquisition equipment functioned correctly. The test was then conducted on October 3, 1980. A sled speed of 352 knots was attained at launch, corresponding to 309 knots equivalent airspeed. The dummy, which was the same one used in the zero/zero test, was seated in a crew seat simulating the use of the Pilot Seat Repositioning System. The dummy was restrained in place with a 5 point harness featuring the single point gas pressure actuated buckle release mechanism.

The extraction was shown by the sled mounted cameras to be clean and free of any contact with the mockup. The only anomally occurred after the rocket separated from the dummy, when the dummy rotated several times during parachute deployment. The film coverage showed this to be the result of the loose-fitting helmet falling off the dummy's head when the torso harness extraction lines went slack. The helmet then hung by these lines in the airstream and supplied the rotational force. Neither the helmet nor the head of the dummy showed any sign of duress. A fully blossomed parachute was achieved despite the rotation. The test was therefore considered successful, particularly in view of the previously mentioned Airbus data which had shown no problems in this area. The system was accordingly delivered to Lear Fan by Stencel with two rockets, and has been installed and operational from the first flight of the airplane.

It was Lear Fan's responsibility to provide the egress opening for the AWAY system. A NASTRAN structural analysis was done which showed that the fuselage would possess sufficient structural integrity for any foreseeable egress situation with the egress opening structure cutaway. Explosive Technology, of Fairfield, California, developed and supplied the system, which consists of an explosive initiation handle mounted on the instrument panel to the left of the pilot's control wheel, flexible detonating cord, linear shaped charge placed around the egress opening structure and the associated connectors and hardware. The handle is a two action (squeeze-pull) "T" handle which mechanically activates an explosive initiator. Energy is transmitted through the detonating cord which activates the linear shaped charge affixed to the aircraft structure. The charge is so shaped and sized (by explosive strength) that the structure will be completely severed from the fuselage by a high energy gas cutting jet. Gravity, the external flow field and cabin pressurization forces will then separate the structure from the airplane.

Tests of this system were conducted by the supplier. Lear Fan supplied to Explosive Technology samples of representative graphite/epoxy structural components utilized within the fuselage structure around the cabin door frame. These structural coupons were fitted with linear shaped charges and tested for clean severance. Several sizes (grains per foot) of charge affixed to parts of the coupons were tested until a clean completed severance was obtained. The successful charge size and configuration for each test sample was recorded and subsequently fitted to the actual aircraft structure. The actual structural thicknesses of the airframe structure to be cut were ultrasonically measured to insure that properly sized charges, as determined by the coupon tests, were installed.

Closing Remarks

This paper would not be complete without some discussion of the costs and justifications of the entire egress system. We all wish that cost were no consideration in protecting the lives of aircrew members, but we also know that in the real world that will never be the case. The costs of the entire system were in the neighborhood of $300,000. It is the author's opinion that, for what it bought, this was quite inexpensive. That, of course, is arguable. However, for other companies who may wish to consider such a system, the following purely economic and technical advantages of the system can be cited:

1) Except for attachments, no airframe modifications are required for any part of the system. Many alternative egress schemes, require structural modifications, reinforcements, and/or re-routing of systems and control cables. Although the costs are likely to be buried in an experimental program budget, engineering and building such modifications is not cheap. Such modifications may make rerouted systems or relocated equipment unrepresentative for test purposes and difficult to maintain, and can make conversion of a prototype to a standard airplane prohibitively difficult and expensive.

2) A door cutter can be installed in one day's work. The AWAY system and pilot seat mover are easily moved from one airplane to another. The system can therefore be used in any of the prototypes. Most alternative schemes are difficult or impossible to move, requiring a separate system for each test airframe even when there is no simultaneous use requirement.

3) It is often necessary for verification purposes to do stall characteristics or flutter tests on a production airplane. Such tests are frequently done with no escape provisions whatever because of the difficulty and expense of

installing and removing them. In the case of the Lear Fan system, it is easily possible to have an entire system operational for such tests. The only hardware lost is the door cutting charge, which is destroyed during removal and which costs under $1,000.

In summary, the Lear Fan egress system provides capabilities for crew escape which are far superior to those found in most civil prototypes. Its advantages are reliability, safety, ease of installation and removal, and portability which makes it more than a one airplane system. It's initial cost is felt to be justified in comparison with the cost of more conventional egress provisions by these characteristics.

QUANTIFYING AFTI/F-16 GUST ALLEVIATION CHARACTERISTICS USING FREQUENCY RESPONSE ANALYSIS

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Abstract

The AFTI/F-16 multi-mode flight control laws are task-tailored to include gust alleviation of normal acceleration or pitch rate response to turbulence. A new data analysis method was developed to compare the four different control laws in terms of the cockpit vibration environment, ride quality, pitch response and control surface activity in rough air. Results for one flight condition clearly show the effects of the different control law architectures. The maneuver enhancement (decoupled control law) architecture was found to be the most effective in terms of gust alleviation obtained from a given level of control surface activity.

Background - Aircraft Gust Alleviation

Winds and turbulence have helped and plagued manned flight since the beginning.[1] Over the years much study has been made into the origins, description, and effects of atmospheric turbulence.[2] Military aircraft in the sixties were tested in turbulence to ascertain dynamic structural loads response to gusts.[3,4] This led to the application of power spectral gust design procedures to bomber aircraft.[5]

Turbulence is a random phenomenon. Military Specifications specify the power spectrum of turbulence, and also the expected intensity as a function of altitude based on experimentally determined exceedance probabilities.[6] Aircraft designers have sought to reduce the adverse effects of turbulence either through aerodynamic (decreased aspect ratio and wing area) or active control technology.[7] As ride smoothing systems were developed for low altitude, high speed aircraft, a ride quality criterion was incorporated into the appropriate Military Specification.[8] Called the Discomfort Index, it accounts for the relative discomfort caused by vibration of any given frequency. The curves of Figure 1 show the relative weighting assigned to both vertical and lateral vibration. Low frequency vibration (below one hertz) induces motion sickness. Vibration at higher frequency causes various body parts to resonate uncomfortably. Later, during the design of the B-1 aircraft, active control technology was applied to improve the pilot's ability to perform low-level terrain-following tasks in rough air.[9] A crew task performance (impairment) index was developed to account for the relative tracking error induced by vibration of any given frequency.[10,11] Figure 2 shows the weighting functions used by this index. The relative weighting given to high frequency vibration is more for this index than for the Discomfort Index mentioned earlier. Reference 12 compared these two ride quality criteria for the B-1 aircraft and showed that the crew task performance index correlates better with crew comments from flight test.

A most challenging design task for active control technology is to improve the ride quality of a low-wing-loading, highly maneuverable fighter aircraft. The YF-16 control configured vehicle demonstrated that its maneuver enhancement control mode could use active flaps and horizontal tails to reduce the normal acceleration and pitch rate time history response to a given patch of turbulence.[13] The much more complex AFTI/F-16 flight control design[14] demanded a more quantitative flight test method for demonstrating task-tailored gust alleviation. This new capability at the Air Force Flight Test Center is the subject of this paper.

The AFTI/F-16

The Advanced Fighter Technology Integration (AFTI)/F-16 is an advanced development program sponsored jointly by the Air Force, NASA and the Navy. The aim is to develop, integrate and demonstrate aircraft technologies to improve fighter mission effectiveness. Some of these technologies are listed in Figure 3. The primary objective of the first phase of development and flight testing is to validate the practicality and benefits of a triplex digital multi-mode flight control system (DFCS) for task-tailored weapon system operation. During the second phase enhanced fire control computers and a low-drag sensor-tracker will be added to develop and demonstrate an Automated Maneuvering Attack System capability.

Task-Tailored Control Modes

The AFTI/F-16 longitudinal control laws use active flaps and horizontal tails in a task-tailored control law architecture designed for improved performance in specific mission phases.

Standard Normal Mode (SNRM) is designed for cruise and formation flying tasks. It is a normal acceleration command system that alleviates normal acceleration response to gusts.

Standard Air-to-Surface Bombing Mode (SASB) increases and quickens the flight path control and has more normal acceleration gust alleviation to provide a stable bombing platform.

The Standard Air-to-Air Gunnery and Strafe Mode (SAAG or SASG) commands pitch rate and reduces pitch attitude response to gusts for stable weapon line pointing. This mode does not drive the flaps for gust alleviation.

Each of the three standard task-tailored modes has a corresponding decoupled control mode. These are based on the same maneuver enhancement feedback structure and thus have identical gust response. Pilot stick inputs, however, command either rapid, precise flight path or pitch rate response depending on the mission phase. Other controllers are also used to command task-tailored decoupled motions such as pitch pointing or direct lift.

Turbulence Flight Testing and Data Analysis

Each of the task-tailored control laws was flown in low altitude turbulence at several airspeeds to verify freedom from aeroservoelastic instabilities.

Each turbulence test run lasted from one to two minutes. A post flight data analysis method was developed to calculate vertical gust velocity (see Appendix), to quantify the task-tailored gust alleviation characteristics of each mode and to compare them with each other. All data presented in this paper are from tests conducted at 500 knots airspeed at 5,000 feet altitude.

Frequency Response Analysis

The heart of the new data analysis method is a versatile fast fourier transform computer program called Frequency Response Analysis (FRA). This was originally developed at AFFTC to identify the handling qualities of aircraft.[15] FRA takes input and response time histories and computes the magnitude ratio and phase angle relationships (transfer function) between them as a function of frequency (Figure 4). FRA also computes the power spectral density (PSD) of each time history along with confidence bounds and coherence functions. The two-input, one-output version of FRA was used to derive the transfer function of the normal acceleration response to vertical gust velocity while subtracting out the effects of pilot inputs on that response. This effectively normalizes the gust response data for each mode so that results can be compared from different flights on different days having different turbulence levels and pilot inputs.

The next step is to standardize these data to the same turbulence model. All data were standardized with a unity-intensity Von Karman spectrum representing the turbulence encountered at 500 feet altitude and at the true airspeed corresponding to the dynamic pressure of the nominal test conditions. The result is a normalized response power spectral density (PSD) function (Figure 5). The area under this PSD function is the mean square of the response to turbulence of unity intensity. One can then easily determine the root mean square (rms) of response expected from any given intensity of turbulence through multiplication. Ride quality acceleration weighting functions for vertical vibration (Figures 1 and 2) are applied to the normal acceleration PSD as illustrated in Figure 6. Ride quality criteria are in this way directly incorporated into the data analysis method.

In order to quantify the gust alleviation present for the strafing task, the pitch rate response PSD was obtained in the fashion described above for normal acceleration. Then it was converted by simple algebra (divide by frequency squared) to quantify the pitch attitude response to turbulence.

Control surface position and rate PSDs were also obtained in a similar fashion. These quantify the cost, in terms of control surface activity, paid to obtain the resultant gust alleviation.

Results

The normalized cockpit vibration environment (PSD) of reach different control mode is presented in Figure 7. At low frequencies the level of gust alleviation corresponds directly with the magnitude of the control law feedback gain to the flaps: SASB has the most, SNRM has about 50% less, the decoupled modes have less still, and SAAG has no normal acceleration feedback to the flaps. At higher frequencies where pitch accelerations induce a high percentage of cockpit vibration, the normal acceleration alleviation is reversed: SAAG, which tightly controls pitch rate (and hence pitch acceleration) is best. SASB is the worst in this region due to a resonance peak at a frequency of about 24 radians per second. This peak is caused by extra control law dynamics introduced into the elevator command path to improve stability for the higher flap gains used in SASB. Pilots flying SASB in turbulence have felt a pronounced "shaking" of the aircraft and observed a more active flight path marker display. Because of this effect, pilots preferred using the decoupled bombing mode during low altitude ingress to a pop-up bombing attack.

Figure 8 shows how the four control modes compare when the Discomfort Index and Crew Task Performance Index weighting functions (Figures 1 and 2) are applied to the vibration environments of Figure 7. The Crew Task Performance (Impairment) Index quantifies the relative tracking performance degradation expected in a vibration environment. The decoupled modes fall within the "acceptable for normal tasks" region in a moderate turbulence environment. The other modes lie within a region of "satisfactory performance with considerable concentration". For tasks demanding less pilot tracking activity (e.g., cruise), the Ride Quality Discomfort Index may be used. No AFTI/F-16 mode quite reaches the level specified for .5 to 1.5 hours tolerance. SASB is best here because this index weights the low frequency end of the spectrum (the motion sickness region) relatively more heavily than the previous index.

Figure 9 shows the normalized pitch attitude response of these four modes. SAAG and the decoupled modes exhibit extraordinary pitch attitude gust alleviation.

Figures 10 and 11 show the trailing edge flap and horizontal tail activity, respectively, for each mode. It is evident that SASB works the control surfaces much harder than any other mode. The decoupled modes seem to give the best all-around gust alleviation for the least control surface activity.

Conclusions

A new capability has been introduced and utilized at the Air Force Flight Test Center to quantify and compare the gust alleviation characteristics of the AFTI/F-16 multi-mode task-tailored flight control system. The new method applies ride quality, task performance, and pitch attitude criteria applicable to each of the AFTI/F-16 design tasks. Results show that the decoupled (maneuver enhancement) control law alleviates gust upsets as well as or better than any standard mode. The standard bombing mode has an objectionable high frequency normal acceleration resonance in turbulence. The standard gunnery mode exhibits excellent pitch attitude gust alleviation.

Acknowledgement

The author is grateful for the support and encouragement of the AFTI/F-16 Joint Test Force, the tutelage and insight into gust velocity measurement provided by Jack Ehrenberger of NASA-Dryden, and the computer programming support and advice of Robin Holmes and others at the AFFTC.

Appendix

The computation of the time history of gust velocity is the first step in gust response analysis. Figure 12 illustrates the equation for vertical gust velocity. The quality of each measurement will determine the validity of all other results.

Angle-of-attack (AOA) is the critical sensor. The AFTI/F-16 has three types of AOA sensors available: one flight test nose boom, two mechanical cones on either side of the forward fuselage, and a hemispherical probe on the right forward fuselage. Each has its own frequency response characteristics which were determined using FRA on flight data. The flight test nose boom AOA sensor has significant moment of inertia from its heavy stainless steel construction and exhibited light damping in turbulence above 30-40 radians per second frequency. The hemispherical probe uses differential pressure to sense AOA; four foot line lengths caused its frequency response to droop at middle frequencies. The best AOA sensor turned out to be the average of the two mechanical cones (averaging cuts the effective resolution and breakout in half and eliminates any errors induced by roll rate) whose frequency response was good to about 69 radians per second.

Pitch attitude and vertical (inertial) velocity were obtained via numerical integration of pitch rate and normal acclerometer sensors, respectively. To reduce integrator bias, the mean of each signal was subtracted before integration.

True airspeed was computed post flight from total temperature and Mach number. Mach number was derived from altitude and calibrated airspeed corrected for position error.

The quality of sensor resolution, phasing, noise level and bandwidth were verified through doublets and other maneuvers in smooth air. The resulting plots and crossplots were also used to verify AOA sensor calibrations.

Preliminary gust transfer functions showed good agreement with a computer simulation model of the AFTI/F-16.

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Crew Task Performance Index
Weighting Functions (From ASD-TR-70-18)

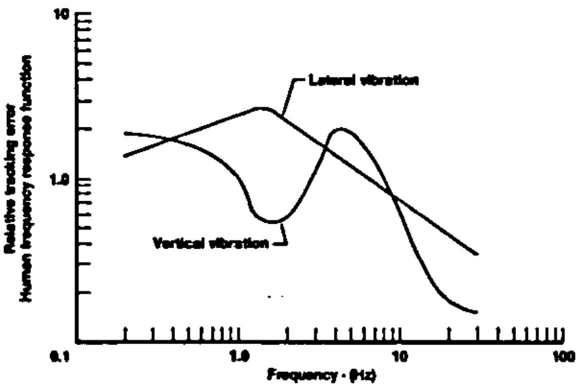

Figure 1.

Discomfort Index Weighting Functions
(From MIL-F-9490)

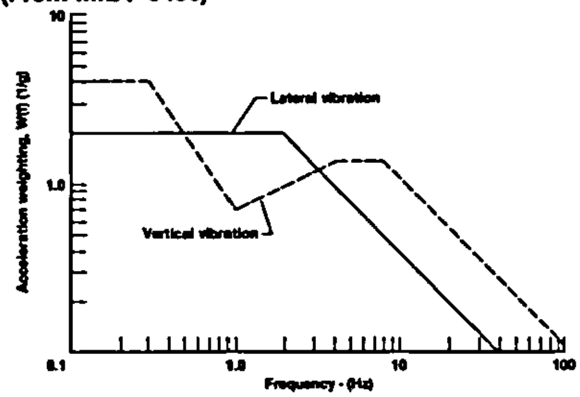

Figure 2.

AFTI/F-16 Technologies for Improved Fighter Combat Effectiveness

Digital Flight Control System
- Triplex redundancy
- Task-tailored multimode control
- Decoupled flight path control

Advanced Control Modes
- Direct force control
- Fuselage aiming
- Gust alleviation

Integrated Flight/Fire Control
- Avionics subsystem integration through digital data bus
- Radar & electro-optical sensor/trackers

Pilot Vehicle Interface
- Multipurpose controls/displays
- Wide field-of-view heads-up display
- Helmet mounted sight
- Voice command

Figure 3.

Frequency Response Analysis (FRA)

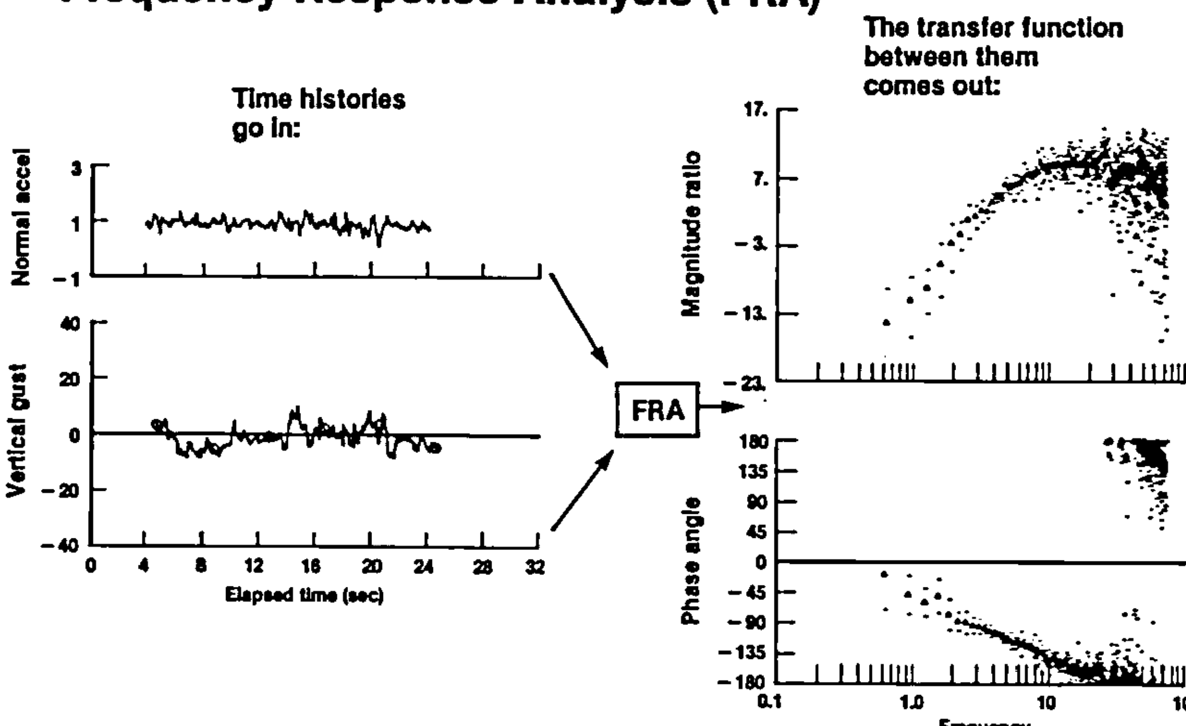

Figure 4.

Turbulence FRA Post-Flight Data Analysis Method Standardized Flight Test Results

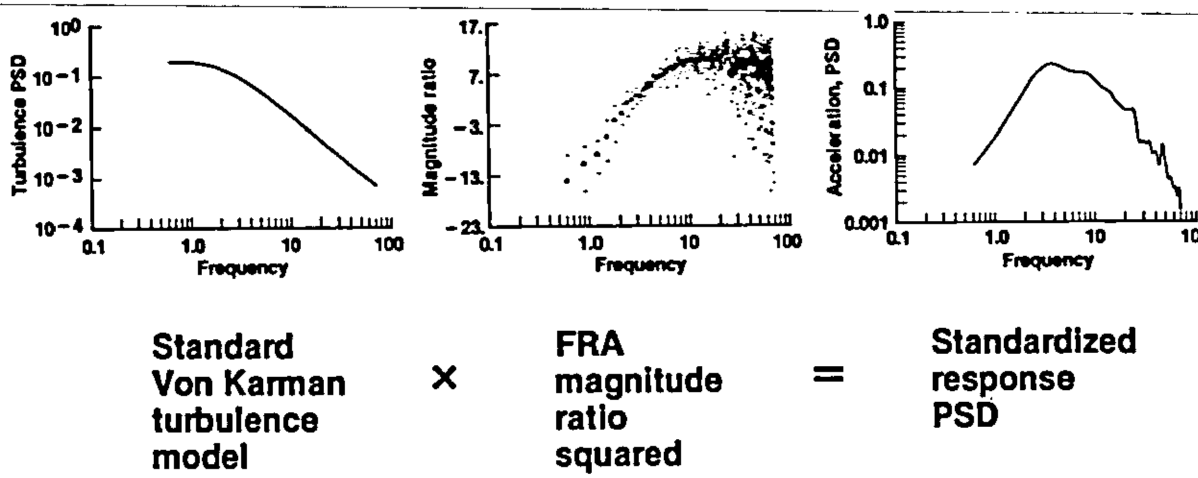

Figure 5.

Turbulence FRA Applies Ride Quality Weighting Functions to the Acceleration PSD

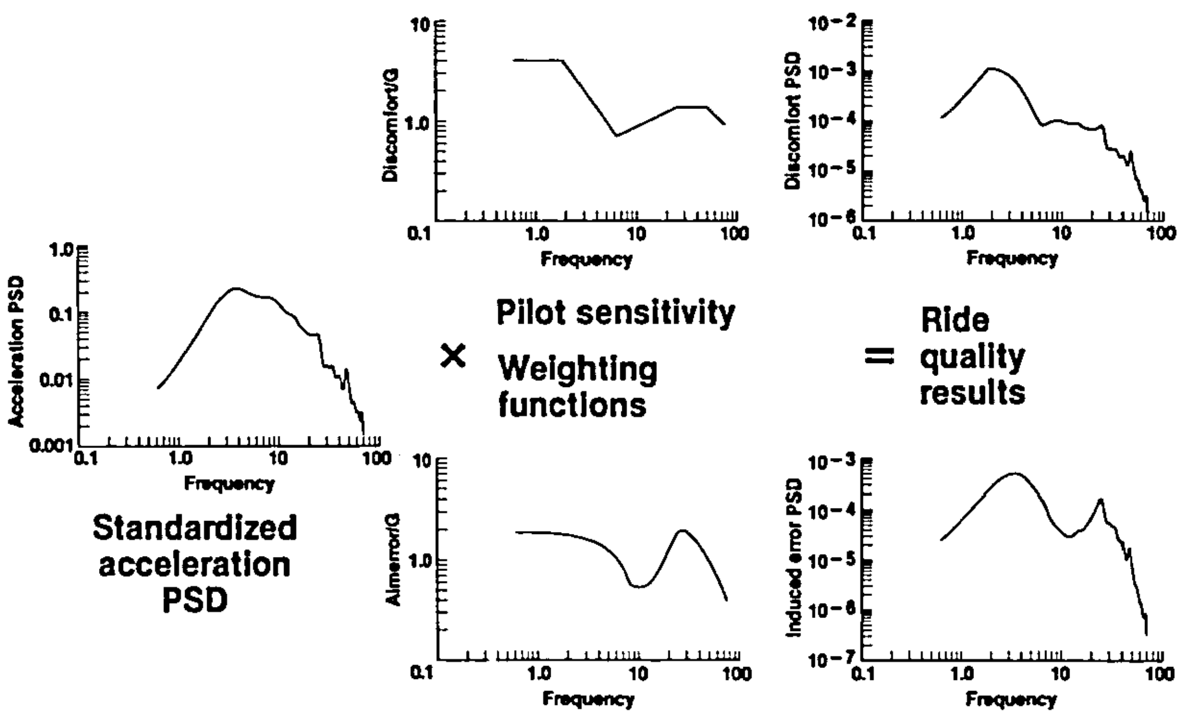

Figure 6.

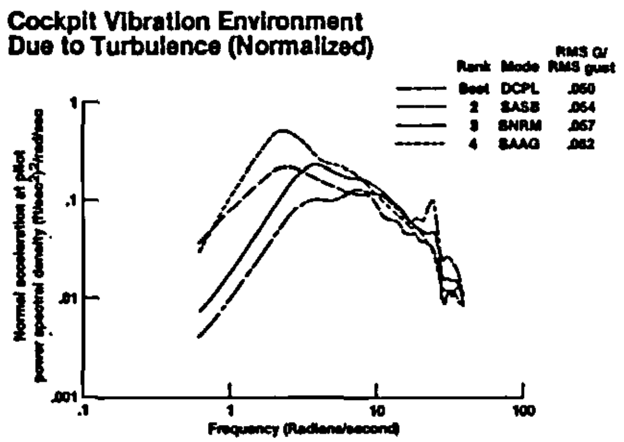

Figure 7.

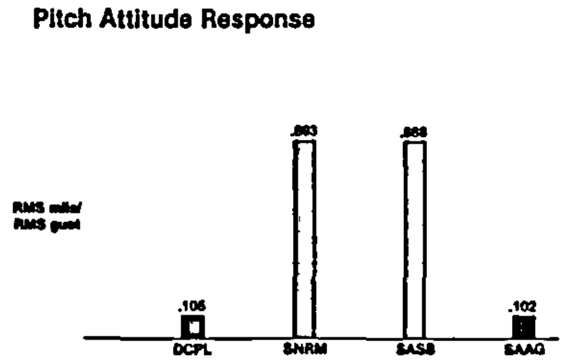

Figure 9.

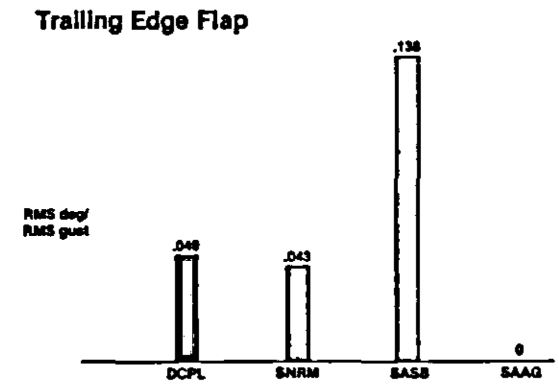

Figure 10.

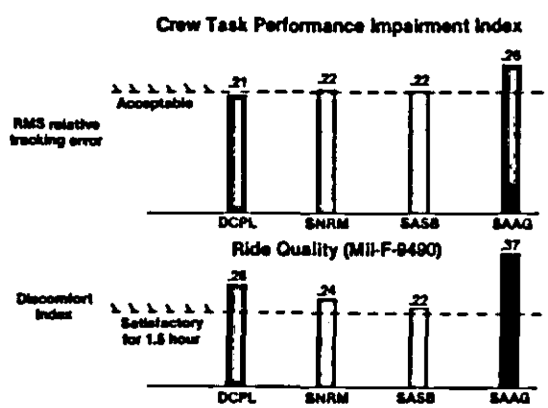

Figure 8.

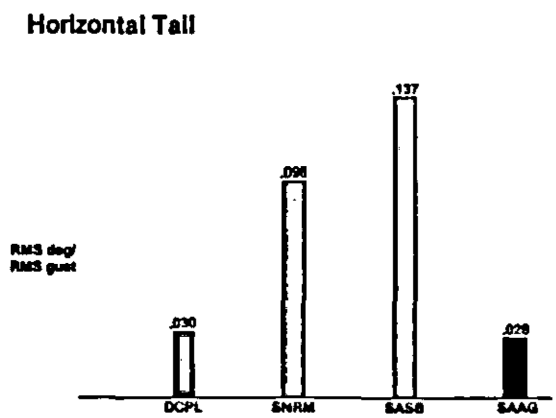

Figure 11.

Vertical Gust Velocity

$$w_g(t) = V_{true}(t) \cdot (\Theta(t) - \alpha(t)) - \int_0^t a_n(t)dt - \ell_x q(t)$$

$$\Theta = \int_0^t q(t)dt$$

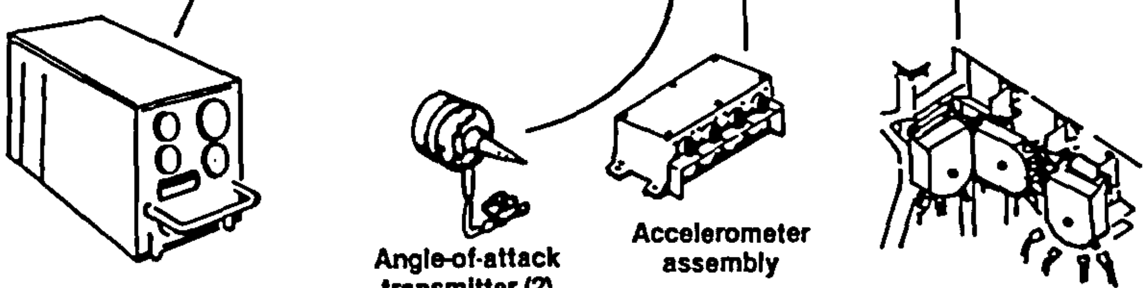

Figure 12.

NATURAL ICING FLIGHT TESTS OF THE BEECH MODEL F90-1 PROTOTYPE

by
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Abstract

Certification of the Beech Model F90-1 King Air required flight tests in natural icing to substantiate the anti-icing capability of the new nacelle design and PT6A-135A engine. This paper describes the pretest planning, modifications to the test aircraft, test procedures, method of data analysis, and comparison of results to certification requirements. A series of fifteen test flights were made from January 29 to March 15, 1982 with a total flight time of 51 hours and 25 minutes. A total of 9 hours and 50 minutes were flown with the ice detector indicating ice buildup. Ground testing included operation on a slush covered runway and during conditions of blowing snow. Icing conditions within the continuous maximum envelope of FAR 25 Appendix C were experienced during eight encounters on four flights. Conditions within the intermittent maximum icing envelope were encountered on one flight. The test aircraft was damaged on two occasions: once by lightning and once by hail while attempting to find and penetrate intermittent maximum icing conditions. All of the ice protection and anti-ice equipment installed on the aircraft performed satisfactorily during all of the test flights. The particle spectrometer icing test equipment performed satisfactorily and represented a great advance over equipment used previously.

Introduction

This paper presents the results of flight tests conducted in natural icing conditions with the prototype Beech Model F90-1 King Air S/N LA-91.[1] These flight tests were required to certify the new pitot cowling and PT6A-135A engine installation for flight into icing conditions. Preflight meetings with the Central Region FAA resulted in the following recommendations:

- An icing encounter in both the continuous maximum and intermittent maximum envelopes of FAR 25 Appendix C
- An icing encounter in either envelope lasting at least 45 minutes[2]
- Ground operation in blowing snow
- Multiple icing encounters during one flight could be considered as one continuous encounter if no accumulated ice was shed from nonprotected areas between ice exposures

*Flight Test Engineer

Configuration

The test aircraft, a standard Beech Model F90 King Air, was modified to the F90-1 configuration by installing pitot cowlings designed by Pratt & Whitney Aircraft of Canada Ltd. These cowlings were fabricated and installed by Frakes Aviation of Cleburne, Texas. Changes from the standard F90 cowling included a reduced area inlet, inlet lip anti-iced by exhaust gases, and a bypass ratio of 21.5% in the icing mode vs. 45% for the standard F90.

Pylons were designed and built at Beech to mount the particle measuring equipment, ice detector, and liquid water content sensor below the wings. A plexiglass observation window was installed in the right engine cowling to allow inflight viewing of the engine inlet screen. Ice depth gages were mounted on each wing tip outboard of the wing deice boots. The test aircraft was equipped with standard airframe equipment for flight into icing conditions.

Instrumentation

These tests were the first Beech icing tests conducted using instrumentation purchased from Particle Measuring Systems Inc. of Boulder, Colorado. The instrumentation package consisted of a DAS-64 Data Acquisition System, FSSP-100 and OAP-200X particle probes, Rosemount 871FA ice detector, Johnson-Williams liquid water content (LWC) probe, Rosemount 102AU1AF deiced OAT probe, and a PERTEC tape drive. The particle measuring probes were mounted on pylons below both wings and just inboard of the wing tips. The right pylon (Figure 1) contained the OAP-200X and Johnson-Williams probes. The left pylon (Figure 2) mounted the FSSP-100 probe and the Rosemount ice detector. The Rosemount deiced total air temperature probe was mounted on an access panel under the left wing. Data lines and heater power cables were routed from the pylons into the wings and to the equipment rack located on the left side of the cabin. A short description of each of the main components of the system and its method of operation are given below:

FSSP-100

The Forward Scattering Cloud Droplet Spectrometer Probe sizes and counts particles from 0.5 to 47 μM in diameter. Particles are sized by measuring the amount of light scattered from a focused laser beam by particles passing through the sample volume. The particles are sized into one of 15 "bins" according to the size range used. The number of particles in each bin are totaled over each data frame.

OAP-200X

The Optical Array Particle Size Spectrometer Probe sizes and counts particles from 20 to 200 µM in diameter. Particles are sized by a linear array of photodiodes sensing the shadows of particles passing through the unit's sample area. Like the FSSP, the OAP-200X sizes particles into 15 "bins."

DAS-64

The DAS-64 Data Acquisition System accepts particle data from the OAP-200X and FSSP-100 probes and data from the airspeed-altitude and engine parameter transducers installed. The DAS-64 has a CRT that displays a hystrogram of particle sizes as they are detected. The flight test engineer may select either particle probe for display. The DAS-64 also has a time code generator.

The DAS-64 unit was integrated into a rack containing signal conditioning equipment, system power controls, probe heater controls, and a PERTEC model T7640-9 tape drive. The control head for the Johnson-Williams probe was mounted on top of the rack. An HP-41CV calculator was programmed with the FAR 25 Appendix C charts and used to calculate icing exposure time required.

Procedures

Preflight

The particle probes were checked for proper operation and calibration before each flight. If necessary, alignment and/or cleaning of the probes were done. The Johnson-Williams probe and the ice detector were functionally checked and the readings of the other transducers were noted. Meanwhile, the flight crew telephoned our contract meteorological consultant for a briefing on that day's icing conditions. When a route and altitude were known, an IFR flight plan was filed to that location. Flights were planned to originate and terminate at Beech Field, if possible, so that a GAR departure slot was not required. Close preflight coordination with Air Traffic Control was essential to the success of the test program, since the controllers strike had just occurred and system capacity was reduced.

Inflight

Flight data was recorded at data rate 2 (2 frames/sec) for continuous icing conditions and at data rate 4 (10 frames/sec) for intermittent maximum icing. The Johnson-Williams LWC meter was zeroed in clear air before entering icing conditions. When clouds were entered, a climb or descent was made until liquid water content reached a maximum. During an icing encounter the flight test engineer monitored the LWC from the Johnson-Williams probe and the drop sizes from the DAS-64 and used the HP-41CV to determine the exposure time required.

Data Reduction

After each flight the data tapes were taken to the Flight Test Data Processing Center for data reduction with the DEC PDP 11/70 computer system. The first step was to convert the flight tape into a disk file of instrument corrected analog data and calculated MVD and LWC from the particle probes. Three time history plots were then made for each flight, as shown in Figures 3-5. The plots of FSSP and OAP parameters were then studied to find periods where the following occurred:

- High concentrations of liquid water
- Temperature below freezing
- Ice detector light on

Time periods meeting these criteria were then checked to see if they constituted a valid encounter in either of the icing envelopes of FAR 25 Appendix C. A computer program was developed to calculate distance averaged values of LWC, MVD, TOAT, KTAS, and altitude over selected time periods. The program also had the ability to reject data below a specified minimum LWC. The distance averaged MVD and TOAT were then used to interpolate in the graphs of FAR 25 Appendix C and to determine the expected LWC. The ratio of distance averaged LWC to expected LWC could then be used to enter the cloud horizontal extent chart from FAR 25 Appendix C to find the distance required for a valid encounter. An example of output is shown in Figure 6.

Results

Ground Tests

Engine operation in moderate to heavy snow and fog was checked on February 2, 1982 at Beech Field. Visibility was 1/4 to 1/2 mile in snow and blowing snow with a temperature of 20°F. The test aircraft was taxied on the snow-covered runway using beta and reverse thrust to create more blowing snow than that which was falling. Fifteen minutes of taxi was followed by fifteen minutes of holding at ground idle power. Three rejected takeoffs were then conducted by accelerating to 80 KIAS and then applying reverse thrust to stop. Inspection after the test runs showed some runback ice a few inches aft of the heated inlet lips both inside and outside of the cowling. There was only a small amount of this ice and it had no effect on engine operation.

On February 12, 1982, taxi tests were done at Beech Field in 2 to 4 inches of slush and snow. Most of the slush displaced by the nose tire appeared to travel over the nacelles and well behind the engine inlets. Both engines operated normally during these tests.

Icing Flights

Table 1 summarizes the continuous maximum and intermittent maximum encounters that met or exceeded the requirements of FAR 25 Appendix C. Compliance with continuous maximum requirements was shown by using the mean volume diameter (MVD) and LWC from the FSSP-100 without taking advantage of any LWC from the larger drops present. Figure 7 shows the continuous maximum encounters on the FAR 25 Appendix C envelope. To show compliance with the intermittent maximum requirements, LWC's from both of the particle probes were summed but the MVD from the FSSP-100 was used. This method was conservative, since the required LWC decreases with increasing MVD. Figure 8 shows the intermittent maximum encounter on the FAR 25 Appendix C envelope. Notable icing encounters included one of 59.3 minutes duration on flight 102 with an accumulation of 6 inches of rime ice on unprotected surfaces. A lightning strike occurred on flight 106 while circling in cumulus clouds near Pocatella, Idaho. The lightning caused damage to the OAP-200X, DAS-64, and to both of the fluxgate compasses. While penetrating cumuliform clouds on flight 111, large hail was encountered that resulted in damage to the wing and stabilizer leading edges, nose cone and tail bullet, prop spinners, and the upper surface of the nose skin. This flight completed the test program.

Conclusions

1. The ice protection equipment including the heated inlet lips and electrically actuated ice vanes satisfactorily performed their intended function.

2. There were no powerplant difficulties, ice buildup on the inlet screen, or uncommanded power changes while operating in either continuous maximum or intermittent maximum icing conditions.

3. There were no flight control or systems difficulties experienced with accreted ice.

4. The aircraft performance was satisfactory with accreted ice on the aircraft, despite the additional penalty of ice on the particle measuring equipment pylons.

5. The Particle Measuring Equipment Inc. icing test equipment is satisfactory to perform icing tests and represents a great increase in accuracy, precision, and convenience over previous equipment used.

6. Penetrating thunderstorms to seek intermittent maximum icing conditions is a very hazardous procedure. Efforts should be made to determine another means of compliance with this requirement, or to determine if the intermittent maximum envelope of FAR 25 Appendix C is realistic. If flights are to be made in these conditions, the test aircraft should be equipped with as much weather avoidance equipment as possible.

References

1. Beech Engineering Report 90E1413F, "Natural Icing Flight Tests of Model F90-1 Prototype With Pitot Cowls" by Brian Mee dated March 23, 1982.

2. Advisory Circular 20-73, "Aircraft Ice Protection," Dept. of Transportation FAA dated April 21, 1971.

Table 1. Icing Certification Points

Flight	Type	TOAT °F	FSSP LWC gm/m^3	OAP LWC gm/m^3	Total LWC gm/m^3	MED Microns	LWC FAR 25 gm/m^3	LWC Factor	CHE Req. NM	CHE Flown NM	Time Flown Min.
94	CON	23.9	.3248	—	.3248	25.0	.4118	.7886	33.5	40.6	12.3
94	CON	25.4	.3411	—	.3411	21.8	.5104	.6683	49.0	50.7	15.8
94	CON	26.7	.5364	—	.5364	22.5	.5082	1.055	14.2	32.4	9.8
100	CON	26.6	.3403	—	.3403	15.6	.7179	.4740	94.0	113.5	42.7
102	CON	20.1	.3540	—	.3540	22.3	.4364	.8113	31.0	147.7	59.3
102	CON	21.5	.5573	—	.5573	23.0	.4356	1.279	6.3	11.9	4.3
105	CON	1.9	.2010	—	.2010	21.8	.2500	.8039	32.0	41.3	11.0
111	ITT	18.5	.1092	.9138	1.023	29.5	1.132	.9037	4.1	4.3	1.3

NOTE: LWC from larger drops not used to show compliance for continuous icing.

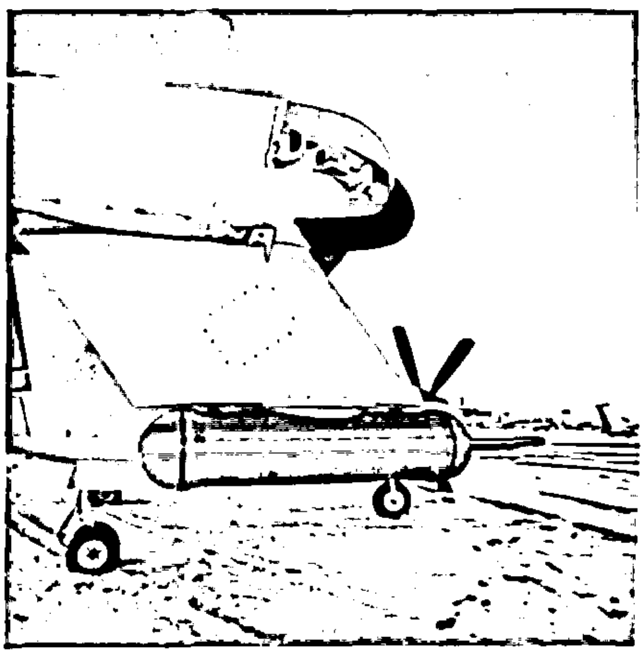

Figure 1. Right wing pylon with OAP-200X.

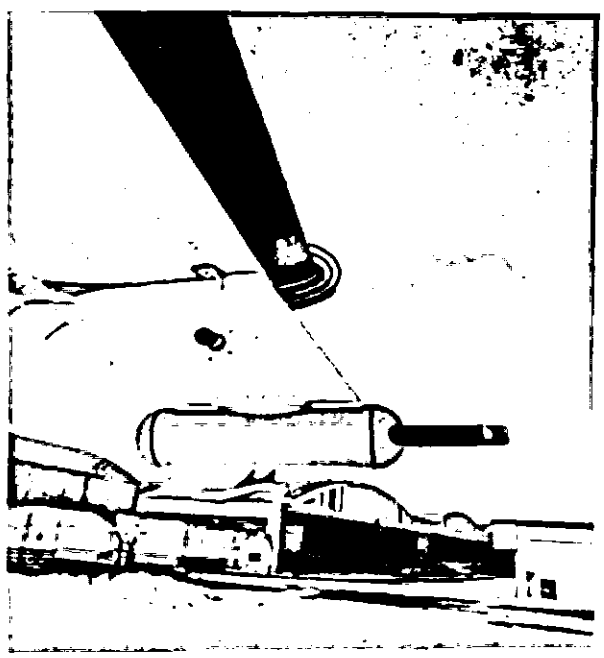

Figure 2. Left wing pylon with FSSP-100 and ice detector - also showing ice depth gage.

11-JUL-83

CEECH TIME SERIES PLOTTER VERS. 3.1

82/ 2/ 2 YEAR/MONTH/DAY AT START OF DATA COLLECTION

83/ 6/18 YEAR/MONTH/DAY OF PROCESSING DAS-64 DATA TAPE

LA91 FLT 95

FSSP PROBE "A" INPUT SELECTED FOR ANALYSIS
OAP PROBE "C" INPUT SELECTED FOR ANALYSIS

FAST DATA AVERAGED OVER 10 OBSERVATIONS

SLOW DATA AVERAGED OVER 1 OBSERVATIONS

AUTO RANGING PROCESSING OPTION NOT SELECTED

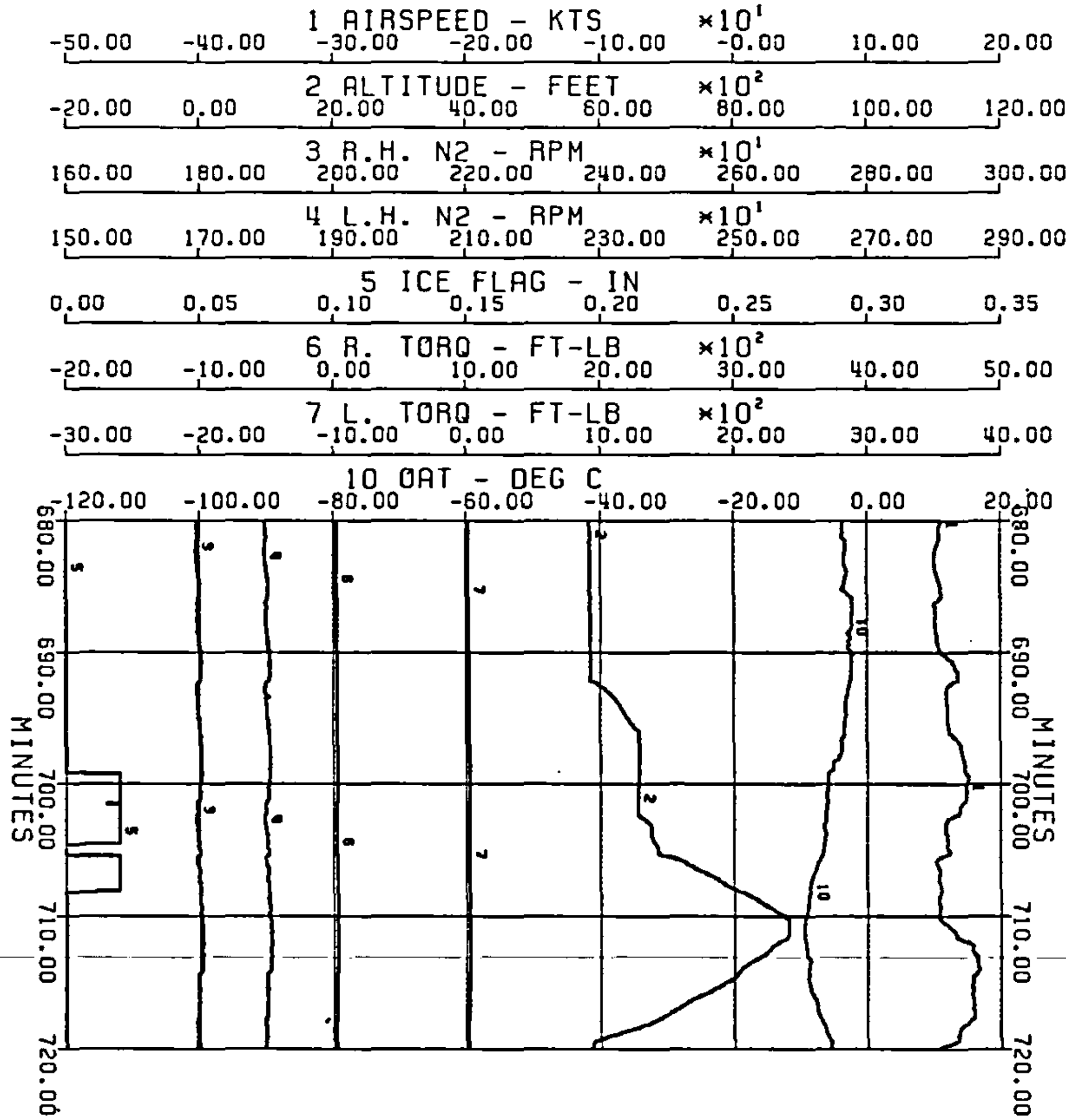

Figure 3. Time History Plot 1 - Engine and Flight Condition Data

27-JUN-83

BEECH TIME SERIES PLOTTER VERS. 3.1

82/ 2/ 2 YEAR/MONTH/DAY AT START OF DATA COLLECTION

83/ 6/18 YEAR/MONTH/DAY OF PROCESSING DAS-64 DATA TAPE

LA91 FLT 95

FSSP PROBE "A" INPUT SELECTED FOR ANALYSIS
OAP PROBE "C" INPUT SELECTED FOR ANALYSIS

FAST DATA AVERAGED OVER   10   OBSERVATIONS

SLOW DATA AVERAGED OVER   1   OBSERVATIONS

AUTO RANGING PROCESSING OPTION NOT SELECTED

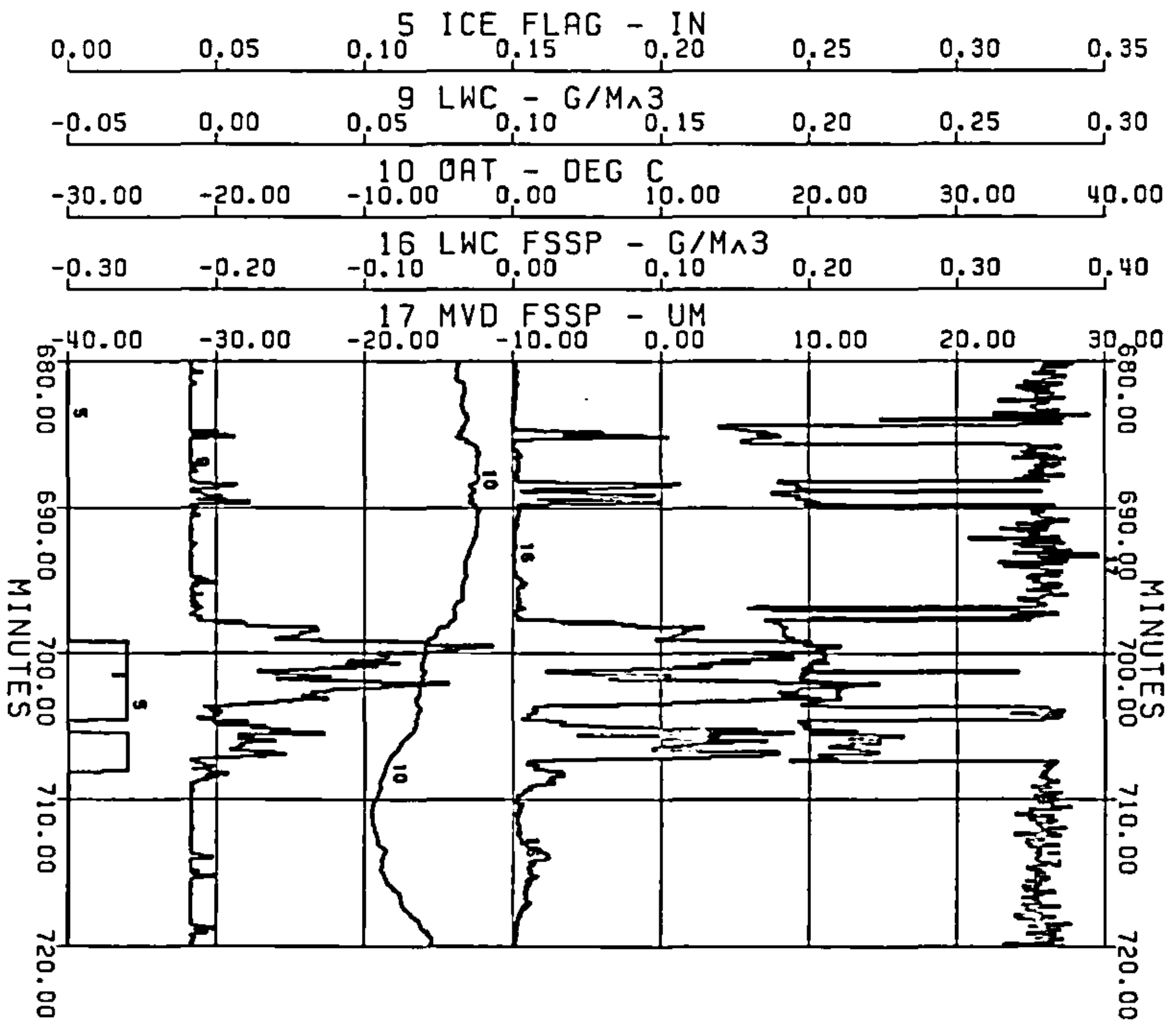

Figure 4. Time History Plot 2 - FSSP-100 Data

27-JUN-83

BEECH TIME SERIES PLOTTER VERS. 3.1

82/ 2/ 2 YEAR/MONTH/DAY AT START OF DATA COLLECTION

83/ 6/18 YEAR/MONTH/DAY OF PROCESSING DAS-64 DATA TAPE

LA91 FLT 95

FSSP PROBE "A" INPUT SELECTED FOR ANALYSIS
OAP PROBE "C" INPUT SELECTED FOR ANALYSIS

FAST DATA AVERAGED OVER 10 OBSERVATIONS

SLOW DATA AVERAGED OVER 1 OBSERVATIONS

AUTO RANGING PROCESSING OPTION NOT SELECTED

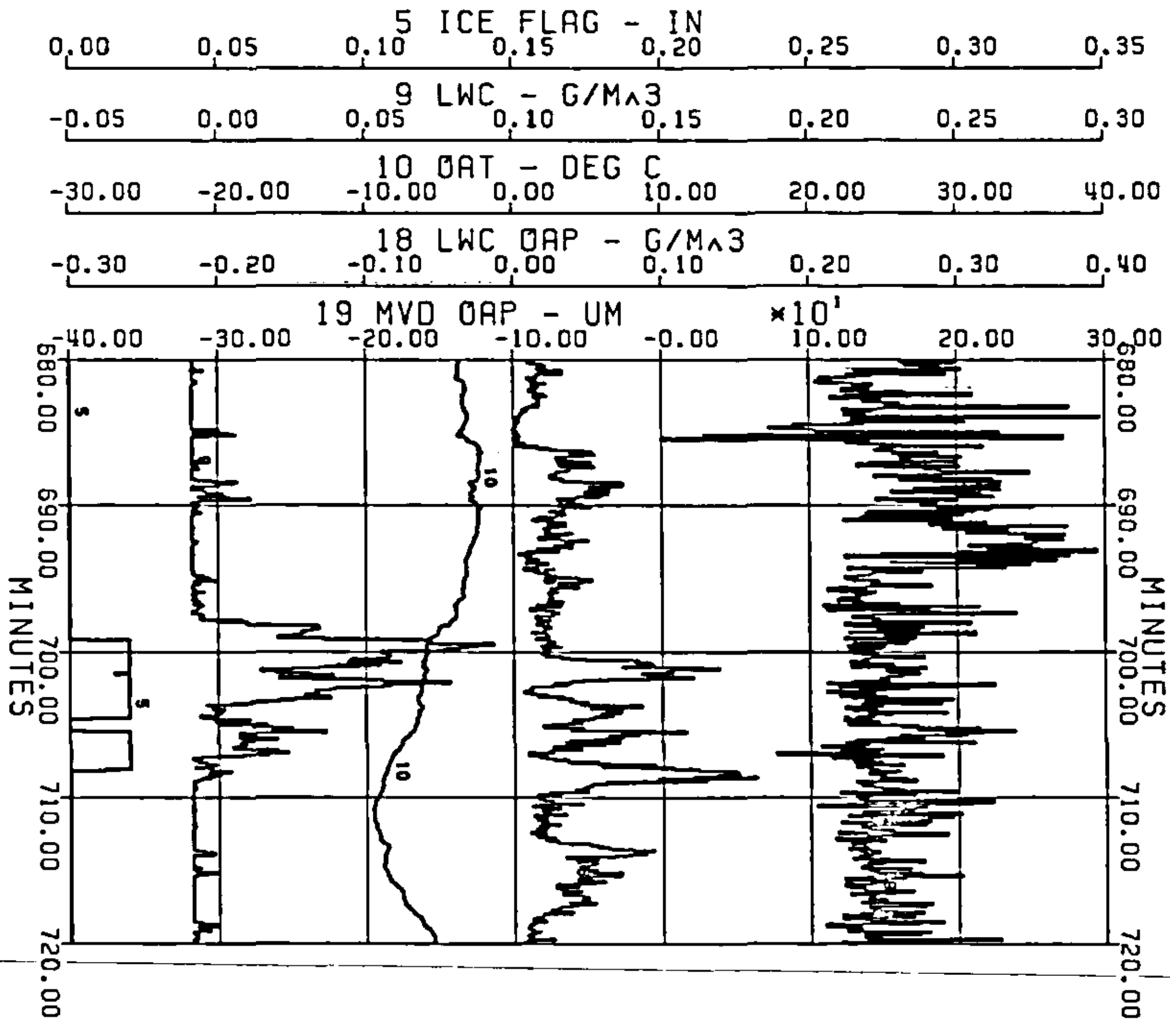

Figure 5. Time History Plot 3 - OAP-200X Data

```
AIRCRAFT SERIAL NO. LA91
DATA FILE FLT95.ICE
TIME INTERVAL FROM  699 TO  709 MINUTES
REQUESTED MINIMUM LWC      0.04 G/M**3
CONTINUOUS REQUIREMENTS


------------------------------------------------------------------

                         FSSP PROBE
                         ----------

DISTANCE WEIGHTED AVERAGE AIRSPEED...........  179.86 KTAS
DISTANCE WEIGHTED AVERAGE ALTITUDE...........  6835.3 FEET
DISTANCE WEIGHTED AVERAGE TEMPERATURE........20.27522 DEG F

MINIMUM LWC IN DATA.......................... 0.00679 G/M**3
MAXIMUM LWC IN DATA.......................... 0.24735 G/M**3

MINIMUM MVD IN DATA..........................   8.685 MICRONS
MAXIMUM MVD IN DATA..........................  27.253 MICRONS

TOTAL TIME................................... 0.16528 HRS
PERCENT TIME ABOVE REQUESTED MIN LWC.........  68.065 %

DISTANCE FLOWN IN TIME INTERVAL..............    29.4 NMI
DISTANCE FLOWN ABOVE REQUESTED MIN LWC.......    20.2 NMI

DISTANCE WEIGHTED AVERAGE MVD................  11.392 MICRONS
DISTANCE WEIGHTED AVERAGE LWC................ 0.14779 G/M**3
LWC FROM FAR 25 APP C........................-1.00000 G/M**3
LWC RATIO....................................-1.00000


------------------------------------------------------------------

                         OAP PROBE
                         ---------

DISTANCE WEIGHTED AVERAGE AIRSPEED...........  176.86 KTAS
DISTANCE WEIGHTED AVERAGE ALTITUDE...........  7132.8 FEET
DISTANCE WEIGHTED AVERAGE TEMPERATURE........19.41410 DEG F

MINIMUM LWC IN DATA.......................... 0.00780 G/M**3
MAXIMUM LWC IN DATA.......................... 0.16312

MINIMUM MVD IN DATA..........................  77.511 MICRONS
MAXIMUM MVD IN DATA.......................... 238.658 MICRONS

TOTAL TIME................................... 0.16528 HRS
PERCENT TIME ABOVE REQUESTED MIN LWC.........  54.621 %

DISTANCE FLOWN IN TIME INTERVAL..............    29.4 NMI
DISTANCE FLOWN ABOVE REQUESTED MIN LWC.......    15.9 NMI

DISTANCE WEIGHTED AVERAGE MVD................ 155.575 MICRONS
DISTANCE WEIGHTED AVERAGE LWC................ 0.08453 G/M**3
LWC FROM FAR 25 APP C........................-1.00000 G/M**3
LWC RATIO....................................-1.00000
------------------------------------------------------------------
```

Figure 6. Output of Distance Averaging Program

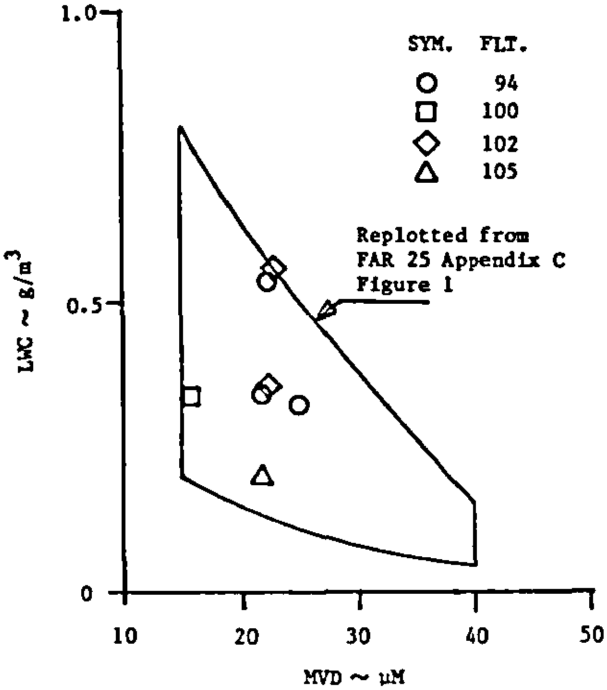

Figure 7. Continuous Maximum Icing Conditions

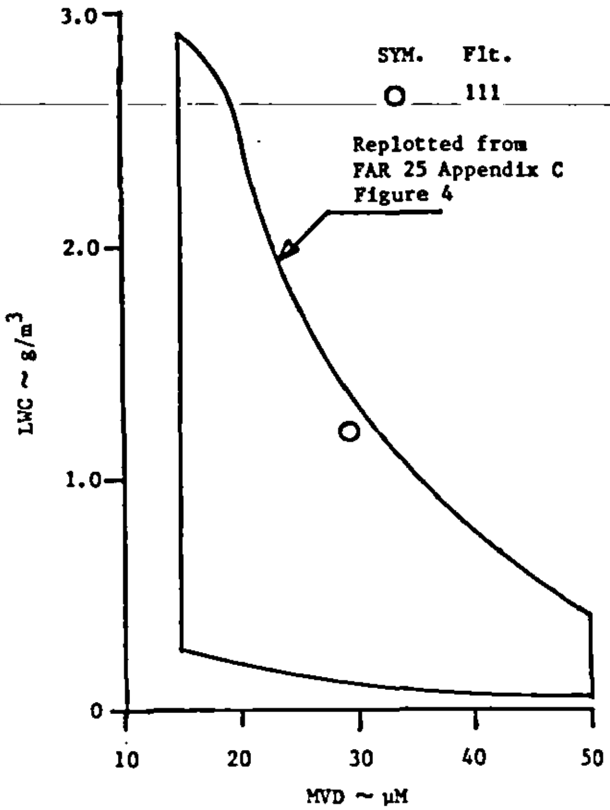

Figure 8. Intermittent Maximum Icing Conditions

CONVENTIONAL TAKEOFF AND LANDING (CTOL) AIRPLANE SKI JUMP EVALUATION

by

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and
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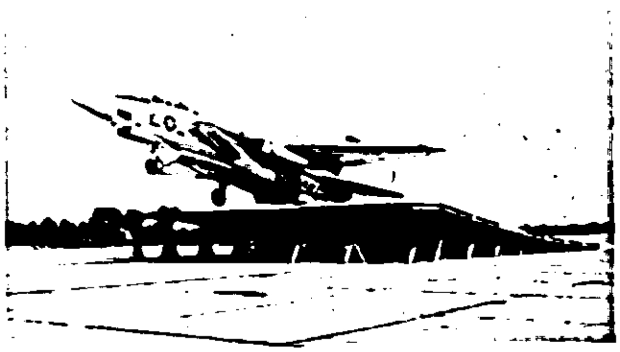

Abstract

The U.S. Navy is evaluating ski jump launches as an alternative to shipboard catapult launch for conventional takeoff and landing (CTOL) airplanes. The Naval Air Test Center (NAVAIRTESTCEN) conducted a ski jump launch test program using a T-2C and an F-14A airplane operating from a variable exit angle ski jump to: (1) evaluate the feasibility of the concept; (2) define the operating limitations; (3) document performance gains; and (4) verify aerodynamic and structural ski jump simulations. A ground and flight test build-up program was conducted prior to actual ski jump operations. This phase consisted of ground acceleration runs, definition of aborted takeoff/committed to takeoff criteria, and high angle of attack (AOA) and dynamic single engine flight characteristics. A total of 112 ski jump takeoffs with the T-2C and 28 with the F-14A was obtained. Tests were conducted from both a 6 and 9 deg exit angle ramp. Significant performance gains were obtained. Reduction in takeoff ground roll in excess of 50% was obtained with the T-2C. Maximum capability with the F-14A was not achieved due to single engine considerations. With longitudinal trim set properly, stick free ski jump takeoff is possible. A stick free ski jump launch is an easier maneuver than a normal field takeoff. Any operational CTOL ski jump airplane should have a Head-Up Display (HUD), nosewheel steering, stability augmentation in all axes, and an accurate, repeatable flight control trim system. Investigation should continue to fully define the application of the ski jump takeoff to both shipboard and shorebased use.

Nomenclature

V	-	Groundspeed (kt)
W	-	Airplane Gross Weight (lb)
g	-	Gravity
F_G	-	Airplane Installed Gross Thrust
a	-	Acceleration (fps)
S	-	Ground Roll

Background

The U.S. Navy is evaluating ski jump launches as an alternative to shipboard catapult launch for conventional airplanes. NAVAIRTESTCEN was tasked to conduct a ski jump launch test program using a T-2C and an F-14A airplane operating from a variable exit angle ski jump to:

- Evaluate the feasibility of the concept.
- Define the operating limitations.
- Document performance gains.
- Verify and update aerodynamic and structural ski jump simulations.

Test Equipment

Ski Jump Ramp

A variable exit angle ski jump was constructed at NAVAIRTESTCEN. The ski jump structure is 60 ft wide and approximately 120 ft long, depending on ramp angle. It is comprised of modular steel construction of which the first 42 ft is a fixed angle ramp with the remainder constructed of 10 ft X 30 ft steel modules secured to steel pedestals. The height of these pedestals is variable to give the desired ramp curvature. The ramp exit angle is defined by the angle of the last inclined module. The last module was level to allow the landing gear to unload prior to ramp edge roll off. Figure 1 presents the general ramp arrangement and specific heights for the two ramp angles. A quarter aft view is presented as figure 2. Leading into the ramp is runway 60 ft wide and 2,000 ft long constructed from AM-2 matting. Additional AM-2 matting was placed on the right side of the runway near the ramp permitting taxi around the ramp and onto an existing concrete runway. For use with the F-14A airplane, a modified holdback/release system was developed permitting stabilized power prior to the ski jump acceleration run. This system could be positioned at any position along the runway to provide the desired ramp speed. Centerline marking consisted of two tram lines 2.5 ft either side of centerline.

Test Airplanes

The T-2C airplane is a two-place airplane with straight, tapered, mid-mounted wings powered by two J85-GE-4 turbojet engines. This airplane is currently a main line U.S. Navy jet trainer. It has hydraulically powered irreversible ailerons, a hydraulically boosted reversible elevator, and a reversible mechanical rudder control. Figure 3 presents a three view drawing.

The F-14A is a supersonic, two-place, twin engine, air superiority fighter powered by two TF30-P-414 dual axial flow turbofan engines with an afterburner for thrust augmentation. All control surfaces are positioned by irreversible hydraulic actuators. Stability augmentation is provided in all axes. A three-view drawing is presented in figure 4.

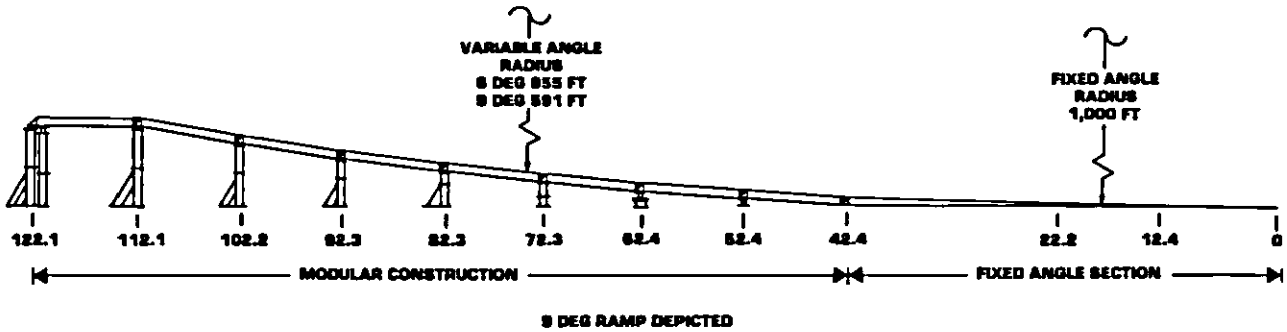

Figure 1 - General Arrangement

Distance Along Ramp (ft)	Height (ft)	
	6 deg Exit Angle	9 deg Exit Angle
0	0	0
12.4	0.19	0.19
22.2	0.41	0.41
42.4	1.16	1.16
52.4	1.68	1.71
62.4	2.30	2.44
72.3	3.03	3.33
82.3	3.88	4.40
92.3	4.81	5.62
102.2	5.85	7.02
112.1	5.85	8.58
122.1	-	8.58

Figure 2 - Quarter Aft View
9 Deg Ramp Shown

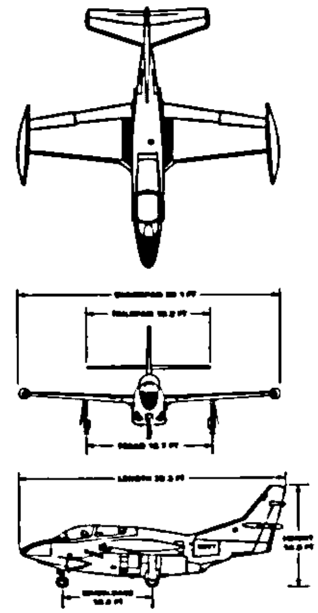

Figure 3 - T-2C Airplane

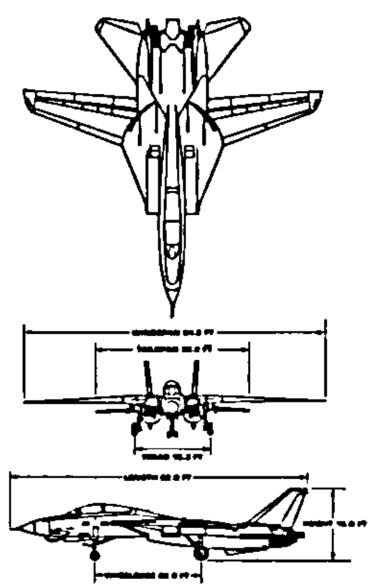

Figure 4 - F-14A Airplane

Both test airplanes were production configured airplanes. However, airborne telemetry/magnetic tape instrumentation systems were installed. In addition, the landing gear of both airplanes were instrumented to obtain shock strut deflections and structural loads during ski jump ramp transition. LASER retro-reflectors were installed permitting real time LASER tracking data during the ground acceleration run and takeoff trajectory.

Test Configurations

All build-up ground and flight tests and ski jump launch operations were conducted in the normal takeoff configurations of the two test airplanes. Table 1 details the test configurations. Two airplane gross weights were chosen to vary the thrust/weight ratio.

Table 1 - Configuration Summary

	Flaps	Gross Weight (lb)	Thrust/Weight
T-2C	Half and	10,000	0.50
	Full (33 deg)	11,800	0.42
F-14A	Full (35 deg)	48,000	0.42
		55,000	0.36 (0.53 A/B)

Build-up Test Operations

In preparation for ski jump operations, extensive build-up ground and flight tests were performed. Considerations for a ski jump takeoff from a viewpoint of the takeoff ground roll included:

- Acceleration performance (ground speed versus distance).
- Abort capability and procedures.
- Emergency limits once past the abort capable point.

Acceleration Performance

Following thrust stand calibration, normal takeoff ground roll tests were performed to equate ground roll to airplane flap configuration, thrust, and gross weight. Simulated single engine takeoff ground roll data were obtained for use in definition of abort capable/committed to takeoff criteria. Takeoff ground roll was related to a takeoff parameter defined as

$$\frac{V^2}{2g}\ \frac{W}{F_G}$$

which is a variation of

$V^2 = 2aS$

and

$F_G = W/g\,a$

The results are shown in figure 5.

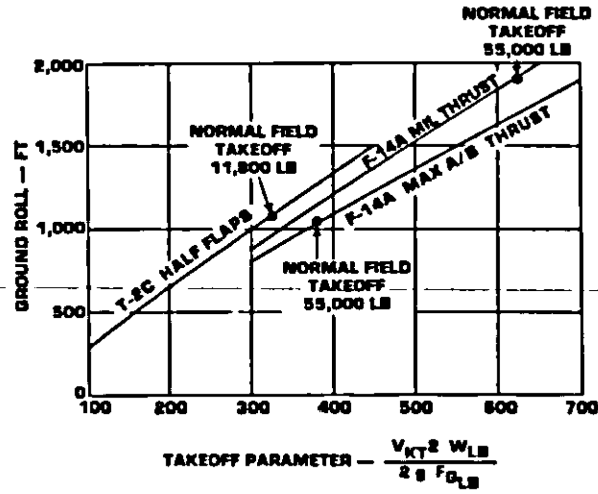

Figure 5 - Takeoff Ground Roll

Abort Capability and Procedures

The abort capability and pilot procedures were defined during simulated aborted takeoffs with the additional requirement of the pilot taxiing around the ski jump (ramp was simulated in position). For each takeoff, the airplane was positioned 30 ft left of runway centerline. The 200 ft wide field runways at NAVAIRTESTCEN were used. During the takeoff ground roll at the desired groundspeed, the pilot would retard one engine to idle (simulated failure). After 1 sec, to simulate reaction time, the pilot retarded the other throttle to idle and made aggressive lateral/directional inputs to move to the right of the runway centerline. The distance traveled down the runway from simulated engine failure to when the left main load gear crossed the runway centerline, denoting clearing to the right side of the ski jump ramp, was obtained from LASER tracking data. Tests were also conducted on the ski jump runway AM-2 matting prior to construction of the ramp. The results are shown in figure 6.

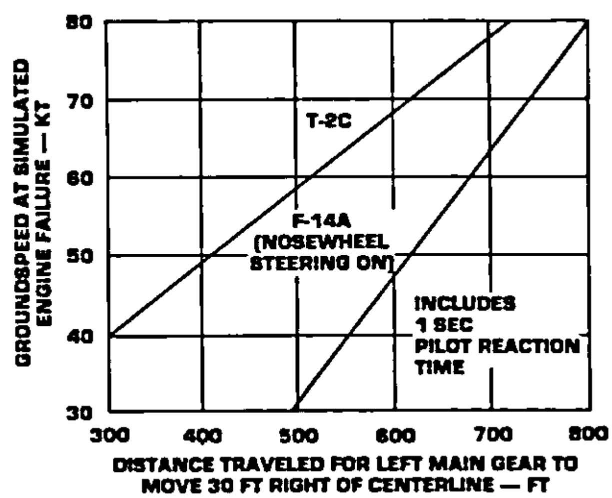

Figure 6 - Abort Capability with Ski Jump Taxi-Around

Single Engine - Committed to Takeoff

Once past the abort capable point during a takeoff ground roll, the airplane is committed to a ramp takeoff. A single engine failure during this time is the most critical from a standpoint of keeping the airplane within the 60 ft width of the ski jump runway and ramp. The T-2C had negligible runway centerline lateral deviation following a simulated engine failure. For the F-14A, the magnitude of asymmetric thrust causes significant lateral deviation. As with the aborted takeoff tests, takeoff runs on 200 ft wide runways were conducted. At the target airspeed, one throttle was retarded to idle. The pilot task was, after a 1 second reaction time, to stop the lateral drift of the airplane. An additional task when using maximum afterburner was to retard the good engine back to MIL thrust. As figure 7 shows at groundspeeds up to 100 kt, the F-14A could be maintained within the 60 ft width of the runway and ramp (20 ft deviation +8 ft gear semispan). However, for these runs, the airplane had traveled approximately 1,000 ft down the runway before the maximum deviation occurred. Note that if a single engine failure had occurred at this worse case point, due to the closeness of the airplane to the ramp, the airplane would have exited the front of the ramp prior to the maximum lateral deviation occurring.

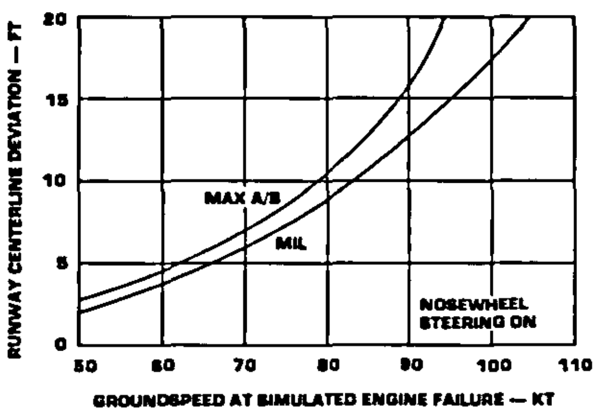

Figure 7 - F-14A Maximum Runway Lateral Deviation Following Simulated Engine Failure

Flight Operations

As a final build-up to actual ski jump takeoffs, flight operations were conducted involving the areas of:

- High AOA characteristics. The maximum AOA was dictated by the airplane type Navy flight manual limits.
- Static/dynamic single engine characteristics. Dynamic single engine characteristics were evaluated both in a 1 g flight regime and less than 1 g as is encountered during the initial part of a ski jump flyaway.
- In ground effect tests at low altitude over a runway to familiarize the pilot with the airplane in this environment and to provide visual cues of airplane attitude and AOA during a ski jump launch.

Computer Simulation

Extensive computer simulation effort was expended prior to conduct of the first ski jump launch. Simulations included both an aerodynamic model and a landing gear/load factor structural model. The simulation was coordinated mainly by the Naval Air Development Center, Warminster, Pennsylvania. Man in the loop simulation for the T-2C was conducted at the Naval Training Center, Orlando, Florida, and the F-14A used the motion based Flight Simulator for Advanced Aircraft (FSAA) located at NASA Ames, Moffett Field, California. The simulations not only predicted the performance gains and structural loading but enabled the test team to develop a test build-down procedure during actual ski jump operations. Also, airplane single engine failure response characteristics, ramp departure minimum single engine airspeeds, and optimum pilot technique were defined. For the F-14A, aircrew coordination during a ski jump takeoff ground roll and flyaway was refined.

Ski Jump Procedures

Flyaway Considerations

The following criteria established the minimum takeoff airspeed during a ski jump launch:

- Reaching limit AOA.
- A zero rate of climb.
- Pilot takeover during rotation and flyaway due to undesirable rotation characteristics. It was desired to trim the airplane to permit stick free flyaway.

Minimum Single Engine Ramp Exit Airspeed

The minimum ramp exit airspeed for a single engine condition was based on the results of computer simulations mentioned earlier. The basis of this airspeed was dynamic single engine controllability or airplane touchdown within established design sink speeds. For the T-2C, dynamic single engine control was possible at all airspeeds investigated; however, sink speed at touchdown beyond the ramp was the limiting factor. For the F-14A, static/dynamic single engine control was the limiting factor. Because of limited flight tests in this flight regime, the initial limit airspeeds based on simulation were increased by 5 kt to provide a safety margin. These minimum ramp exit airspeeds are summarized in table 2.

Table 2 - Minimum Ramp Exit Airspeeds for a Single Engine Condition

	Flaps	Airspeed (KEAS)
T-2C	Half	70 (10,000 lb gross weight)
		80 (11,000 lb gross weight)
	Full	70 (All weights)
F-14A	Full	100 (Mil Thrust)
		115 (Max A/B Thrust)

Build-Down Procedures

The initial ski jump takeoff airspeed for each flap and gross weight combination was targeted for the corresponding normal field takeoff airspeed. For successive ski jump takeoffs, the target airspeed was reduced, a maximum decrement of 3 kt, until one of the previous identified flyaway considerations was attained. The airplane was refueled after each event. An engine runup was performed prior to each event to determine actual test day installed gross thrust.

Individual Launch Parameters

An actual ski jump launch scenario is depicted in figure 8. For the target end airspeed and the test day conditions, the target ramp end airspeed is calculated. With the results of engine runup, the required start position is determined. The normal ground acceleration line depicts airplane groundspeed during the takeoff ground roll. The ground abort capability (from figure 6 plus the 110 ft length of the ramp) is shown. The intersection of the ground abort and ground acceleration lines (point A) defines the limit abort point. At anytime prior to point A, the takeoff ground roll can be safely aborted with the airplane taxiing clear of the ramp. Past this point, the airplane is committed for takeoff.

The second emergency consideration is the ramp exit airspeed for single engine. For this example, T-2C at a gross weight of 10,000 lb with full flaps, the desired minimum ramp exit airspeed is 70 KEAS (table 2). Using test results obtained during the simulated single engine acceleration runs conducted as part of the build-up phase, a single engine acceleration curve is projected back from the desired minimum ramp exit speed until it intersects the normal ground acceleration line (figure 8, point B). At anytime after reaching this point, in the event of a single engine failure, the pilot can affect a safe flyaway or land on the runway beyond the ski jump ramp within the structural limits of the airplane. The time span (Δt) between the ground abort capability and single engine acceleration lines was calculated for each test event (in this case 1.3 sec). For the F-14A only, this time span was limited to a maximum of 3.0 sec.

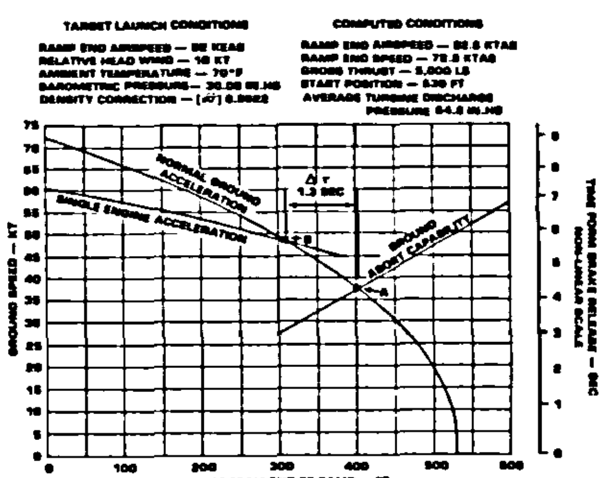

Figure 8 – Typical Ski Jump Launch
T-2C Airplane Full Flaps
10,000 lb Gross Weight

The ski jump launch scenario was determined for each ski jump event. All launch computations, including a computer graphics display of figure 8, were performed by test engineering personnel located at the telemetry ground station. The pilot was given the ground abort groundspeed and distance from the ramp and the time span (Δt) between the ground abort position and the single engine acceleration position required to achieve the minimum single engine ramp speed. Real time LASER tracking during the ground run allowed engineering personnel to advise the pilot when he had reached the limit ground abort point. The pilot, knowing the time span to single engine capability, could make an assessment of single engine capability in the event of an engine failure.

Ski Jump Test Results

General

A total of 140 ski jump launches (112 with the T-2C and 28 with the F-14A) operating from both the 6 and 9 deg ramp was obtained. Significant reductions in takeoff ground roll were obtained up to a 52% reduction in takeoff ground roll for the T-2C. A one-third reduction in ground roll was obtained with the F-14A. The full potential capability of the F-14A was not attained due to the limited minimum airspeed for single engine considerations. With the proper longitudinal trim set prior to the ski jump takeoff, a "hands off" capability during rotation and flyaway following ski jump ramp exit was possible. Minimum ramp exit airspeed for the T-2C operating from the 6 deg ramp was dictated by a performance limit (zero rate of climb during flyaway) but was dictated by undesirable negative pitch characteristics when operating from the 9 deg ramp. The F-14A was neither performance nor flying qualities limited within the range of airspeeds tested. Neither airplane was limited by structural loads during ramp transition for the gross weights and ramp speeds obtained. However, high normal acceleration was experienced by the aircrew at the higher ramp speeds associated with the F-14A.

Performance Gains

As the ski jump launch exit airspeed was decreased, the minimum rate of climb during the ski jump flyaway slowly decreased. The minimum rate of climb as a function of ramp exit airspeed for the T-2C for both the 6 and 9 deg ramp is shown in figure 9. Simulator results are shown for comparison.

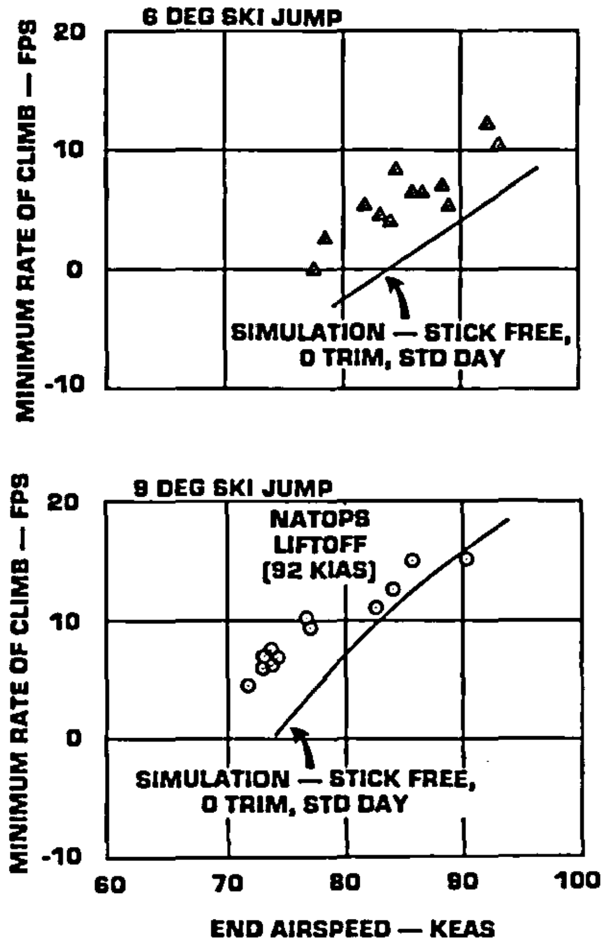

Figure 9 - T-2C Minimum Rate of Climb
During Ski Jump Takeoffs
Half Flaps - 10,000 lb Airplane Gross Weight

Comparable results for the F-14A are shown in figure 10. Only MIL thrust was used during ski jump operations. No MAX A/B ski jump launches were performed. For both test airplanes, for all flap positions and gross weights tested, similar trends were noted. T-2C actual test results showed improved minimum rate of climb performance over that predicted from computer simulation. Likewise, the F-14A on the 6 deg ramp showed improved performance. For all T-2C 6 deg ramp test flap/gross weight combinations, the minimum takeoff airspeed was dictated by reaching a minimum rate of climb of approximately zero during the ski jump flyaway. Minimum exit airspeeds during 9 deg ramp tests were dictated by flying qualities considerations (see below). For the F-14A, the minimum ramp exit airspeed was dictated by single engine control considerations. No other critical parameter, such as zero rate of climb, high AOA, or undesirable flying qualities, was approached within the range of airspeeds tested. Minimum takeoff airspeeds achieved are summarized in table 3. For three of the four T-2C test sets, slower airspeeds were obtained on the 9 deg ramp. The slower airspeeds were most significant (approximately 5 kt slower) for the 10,000 lb gross weight (higher thrust/weight ratio).

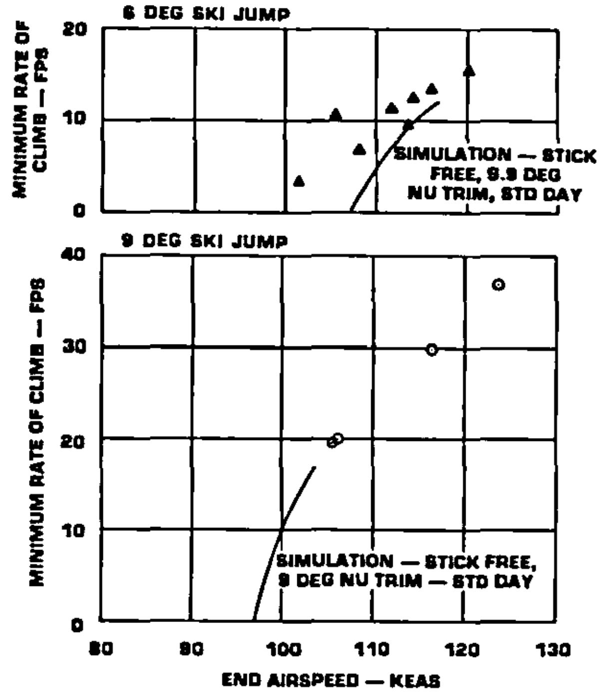

Figure 10 - F-14A Minimum Rate of Climb
During Ski Jump Takeoff
Full Flaps - 55,000 lb Airplane Gross Weight
Mil Thrust

Table 3 - Ski Jump Launch Minimum Takeoff Airspeed

Airplane	Flaps	Gross Weight (lb)	Minimum Takeoff Airspeed (KEAS) 6 deg Ramp	9 deg Ramp	Normal Field Takeoff Airspeed (KCAS)
T-2C	Half	10,000	78	72	89
		11,800	86	85	99
	Full	10,000	70	64	89
		11,800	76	78	93
F-14A (MIL Thrust)	Full	48,000	101	103	125
		55,000	101	106	127

Minimum airspeed criteria:

1) T-2C - Proximity to zero rate of climb on 6 deg ramp and undesireable rotation characteristics on 9 deg ramp.

2) F-14A - Single engine airspeed considerations.

With the reduction in ski jump takeoff airspeed was a corresponding reduction in takeoff ground roll. T-2C ski jump launch reduction in takeoff distance for both takeoff ground roll and to clear a 50 ft obstacle for half flaps at a gross weight of 10,000 lb is presented in figure 11. Corresponding F-14A results are shown in figure 12. The reduction in distance is related to the airplane's flight manual performance data for the test day conditions. The maximum reduction in takeoff ground roll relates to the minimum takeoff airspeed, whether dictated by zero rate of climb, undesirable rotation characteristics, or single engine airspeeds. For any ski jump where minimum ground roll is required and the takeoff trajectory is not critical, the lowest takeoff airspeed is necessary. The data relating distance over a 50 ft obstacle is shown to provide a basis of comparison of ski jump performance during flyaway. It is obvious that, at least for the T-2C, ski jump takeoff airspeeds somewhat higher than the minimum takeoff airspeed provide the optimum takeoff trajectories to clear a 50 ft obstacle. The T-2C relative increase in takeoff distance is directly related to the flattening of the takeoff trajectory at an altitude less than 50 ft and remaining there until the airplane accelerates and increases rate of climb. With the F-14A, no clear trends are noted other than the improved 50 ft obstacle clearance of the 9 deg ramp compared to the 6 deg ramp. Reduction in takeoff distances are summarized in table 4. For the T-2C, the 9 deg ramp provided the maximum performance gains. The 9 deg ramp also provided the best performance benefits for the F-14A airplane.

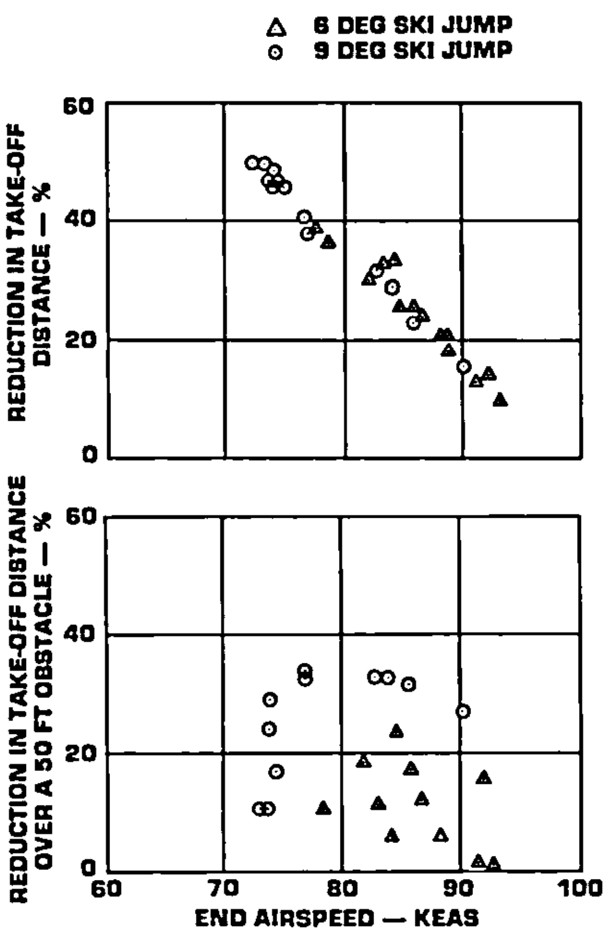

Figure 11 - T-2C Reduction in Takeoff Distance
During Ski Jump Takeoff
Half Flaps - 10,000 lb Airplane Gross Weight

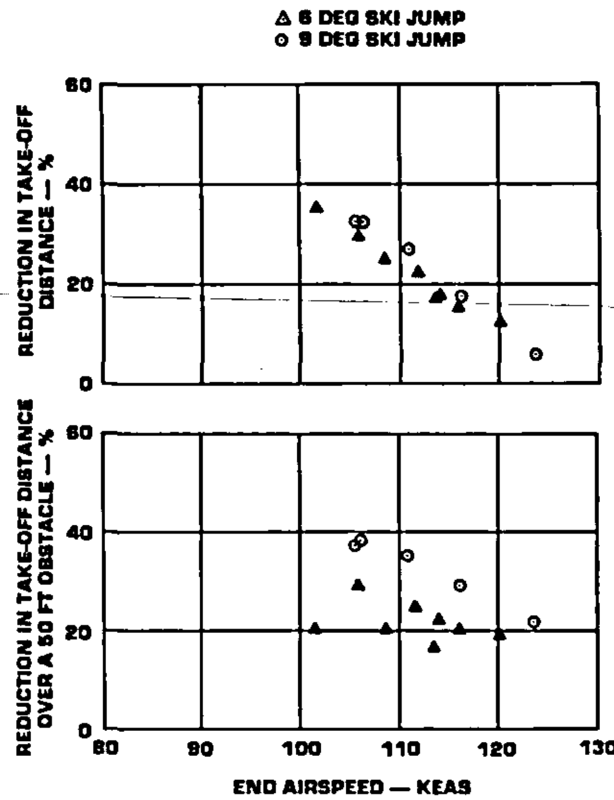

Figure 12 - F-14A Reduction in Takeoff Distances
During Ski Jump Takeoff
Full Flaps - 55,000 lb Airplane Gross Weight
Mil Thrust

Table 4 - Comparison of Reduction in
Takeoff Distance

Airplane	Flaps	Gross Weight (lb)	% Reduction in Takeoff Ground Roll		% Reduction in Distance Over a 50 ft Obstacle			
					Maximum		Corresponding Takeoff Airspeeds (KEAS)	
			6 deg Ramp	9 deg Ramp	6 deg Ramp	9 deg Ramp	6 deg Ramp	9 deg Ramp
T-2C	Half	10,000	38	50	23	34	85	77
		11,000	30	34	28	35	90	89
	Full	10,000	46	52	34	40	84	67
		11,000	46	52	26	39	89	81
F-14A	Full	48,000	36	36	35	39	102	103
		55,000	35	33	29	38	106	106

Ground Handling and Flying Qualities

T-2C Airplane

The ski jump takeoff began with clearance for takeoff. The engines were stabilized at 93-95% RPM followed by simultaneous release of wheel brakes and advancement of the throttles to MIL power. Ground acceleration was moderate and easily monitored. The lateral axis (raising a wing) was not significantly affected by the crosswinds (8-10 kt maximum) experienced. Because of the lack of nosewheel steering (NWS), small differential brake applications were used judiciously to maintain directional control until reaching rudder effectiveness speed (40-45 KIAS). Directional control, either with differential braking or rudder, was easily maintained within ±2.5 ft of centerline. Although no problems were encountered in the ground tracking task without an NWS system, all operational CTOL ski jump airplanes should be equipped with NWS for safe and effective use of the ski jump.

After the first sequence of 4-6 ski jump takeoffs, the pilot was able to judge the abort capability point within ±50 ft (1/2 to 3/4 sec); however, the radio call by the engineers was retained as a safety factor. Once beyond the abort point and committed to takeoff, the pilot monitored airspeed and engines while concentrating on the directional tracking task. Entry onto the ramp was characterized by a build-up in normal acceleration to 1.6 to 2.5 g which was quite comfortable in onset rate, duration (≈1 sec), and was by no means debilitating to the pilot. No adverse flying qualities were induced due to landing gear dynamics. Because the rudder is a reversible control system, the decision was made to not use the yaw damper system due to the high probability of the system inducing yaw excursions or kicks to the rudder and rudder pedals. Consequently, any rudder input on the ground for crosswind correction immediately translated itself into a lightly damped Dutch roll mode (≈ 4 sec period) when airborne which was reinforced by the pilot centering the rudder or attempting to counter the Dutch roll. With symmetrical rudder on takeoff, the airplane gently crabbed (yawed) into the relative wind and a small opposite rudder input was all that was necessary to counter the induced roll due to yaw. The Dutch roll mode posed no problems during these evaluations; however, to accommodate adverse conditions, all operational CTOL ski jump airplanes should have stability augmentation in all axes and that feedback of these systems to the pilots flight controls be nonexistent.

The initial pitch attitude off the ski jump was that induced by the inclination of the ramp itself (6 or 9 deg). Adjustments to longitudinal trim provided for small pitch rates (2 - 3 deg/sec) which generally provided satisfactory pitch attitude and AOA combinations. Several pilot takeovers occurred during the initial jumps because of the difficulty in accurately trimming the elevator statically on the ground and the poor repeatability and accuracy of the trim system. All operational CTOL ski jump airplanes require accurate, repeatable, and easy to read trim systems.

The airplane flew an arc with normal acceleration beginning at 0.5 ± 0.1 g and increasing to 1 g over a 1 to 2.5 sec time frame, which usually coincided with the minimum rate of climb for that takeoff. Increase in AOA during the arc resulted from the change in relative wind caused by decreasing flight path angle and increasing pitch attitude. The maximum AOA occurred close to the minimum rate of climb point when flight path angle was lowest and pitch attitude the most positive. The longitudinal trim setting during all ski jumps was for a trim AOA much greater than the slightly negative AOA at ramp exit. Thus, the short period response about the trim AOA was excited and evident to the pilot by an initial positive pitch rate. During the same time frame, the airplane continued to accelerate to wingborne flight. The short period amplitude appeared to damp out and the airplane stabilized about the trim AOA and continued to climb. Ideally, an optimum performance ski jump takeoff would achieve maximum AOA utilizing a longitudinal trim setting which would generate an acceptable pitch rate to achieve the pitch attitude which equated to the airplane's maximum AOA when the flight path angle was zero. The ability to trim the aircraft to achieve this ideal combination of AOA, pitch attitude, and pitch rates was closely approximated off the 6 deg ramp but not achievable off the 9 deg ramp. This phenomena appeared to be related to the difference between the inclination of the ramp and the maximum AOA.

Consequently, the 6 deg ramp permitted the airplane to rotate through an additional 6 deg of pitch and the flight path angle to decrease 6 deg to zero (zero rate of climb) which resulted in maximum AOA. The 9 deg ramp was restricted in that the short period characteristics about the trim AOA did not permit the flight path angle to decrease 9 deg to zero while only increasing the pitch attitude 3 deg. With the 9 deg ramp, maximum AOA was achieved while retaining substantial minimum rates of climb. As ramp exit airspeed was reduced further, less longitudinal trim was required to stay within AOA limits. Decreasing the longitudinal trim resulted in undesirable negative pitch rates after the maximum AOA was reached resulting in an AOA well below an optimum desired during flyaway. Several attempts were made to capture AOA since the AOA indicator was the only "head-up" instrument. The indicator was damped such that aggressive maneuvering only induced a longitudinal pilot-induced oscillation (PIO) with slightly degraded performance. Capturing a target pitch attitude which was compatible with the AOA limit required a small doublet maneuver to stop the pitch rate. Once nose attitude was established, it was easily maintained within ±1 deg of the desired position with very slight control movements. This procedure required the use of outside visual cues (unsatisfactory if operational and IFR) and/or reference to the primary attitude indicator inside the cockpit. This procedure would not be satisfactory for operational use of the ski jump. Consequently, it is recommended that a precise and timely display of all critical flight information be available on a HUD for effective and optimum use of the ski jump, especially if the pilot must enter the loop to improve performance. An enhancing characteristic for any CTOL ski jump airplane would be the ability to achieve optimum performance stick free. A stick free ski jump takeoff is a maneuver which is easier than a normal, pilot in the loop, field takeoff.

F-14A Airplane

The F-14A ski jump began when the operator at the edge of the runway released the holdback bar. Irregularities in the base of the AM-2 matting produced a side to side oscillation about the main gear which, once experienced, did not concern the aircrew and did not affect performance. Maintaining directional control within ±2.5 ft of centerline was easier than the T-2C because of NWS. Recognizing small deviations was just as easy but small NWS inputs greatly reduced pilot workload throughout the entire ground acceleration. The F-14A occasionally experienced a short period (≈ 1 sec) longitudinal oscillation about the main gear due to nose gear dynamics. The nose gear sometimes came off the runway at the higher airspeeds just prior to going onto the ramp. Directional control was not as precise because crosswinds and lateral motions set up by irregularities in the AM-2 matting moved the nose gear directionally when it was not in contact with the runway. The use of NWS allowed the pilot to keep the airplane barely within the tram lines by making small NWS inputs when the nose gear was in contact with the runway.

As in the case of the T-2C, the aircrew was able to recognize the abort capability point within ±50 ft (1/2 sec) but radio call by the engineers was still made as a safety factor. Once beyond the abort point and committed to takeoff, the pilot monitored engines and concentrated on the directional tracking task while the radar-intercept-officer monitored groundspeed on the inertial navigation system (INS) read out. Entry onto the ramp was characterized by a build-up in normal acceleration to 3.1 to 3.7 g on the 6 deg ramp and 3.4 to 5.2 g on the 9 deg ramp as measured at the center of gravity. The duration was brief (≈ 1 sec on the ramp) but the onset rate was perceived as abrupt and unpredictable. The slower ramp endspeed ski jumps were devoid of large amplitude oscillations and produced normal acceleration only slightly higher than the highest T-2C accelerations. These normal accelerations were perceived as satisfactory. The high g points required the aircrew to position their bodies erect in the ejection set to prevent injury similar to whiplash. Upon leaving the ramp, no adverse flying qualities were induced due to landing gear dynamics. The F-14A utilized stability augmentation in all three axes. Once the airplane left the ramp, it would gently crab (yaw) into the relative wind and a small opposite rudder input was all that was required to counter the induced roll due to yaw. The absence of self-sustaining adverse flying qualities, like Dutch roll oscillations, alleviated pilot workload and substantiates the requirement that all operational CTOL ski jump airplanes have stability augmentation in all axes. The initial pitch attitude off the ski jump was that induced by the inclination of the ramp (6 or 9 deg). Longitudinal trim settings produced compatible pitch rates of 7-8 deg/sec which damped to zero or slightly positive in 2-3 sec. Difficulty in setting the longitudinal trim involved a small gauge whose location made it difficult for the pilot to accurately set because of parallax. Although a better trim system than the T-2C, it was difficult to use accurately without cross-checking engineering instrumentation readings. It is strongly recommended that all operational CTOL ski jump airplanes have separate, repeatable, and easy to read trim systems.

Like the T-2C, F-14A rate of climb varied as a function of ramp endspeed, thrust to weight ratio, ramp angle, and longitudinal trim setting. Because of airspeed constraints due to the F-14A single engine controllability considerations, these tests were not able to demonstrate any performance minimums which came close to zero rate of climb on either the 6 or 9 deg ski jump. Several ski jumps that investigated trim sensitivity at the beginning of the program resulted in one pilot takeover. One attempt was made to capture AOA but the indicator was too well damped, similar to the T-2C, such that aggressive maneuvering induced a PIO which was slightly easier to damp out by pilot inputs than the T-2C. Holding the maximum pitch attitude by slight application of aft stick when the nose began to fall through was easy - almost a cushioning effect. Once the pitch attitude was established, it was easily maintained within ±1 deg of the desired position with very slight control movements. This procedure utilized the F-14A HUD which is not precise enough for IFR but adequate for VFR conditions. The limited scope of the F-14A investigation prevented further investigation of pitch attitude capture techniques. In addition, the data to be compared were stick-free simulation. The effectiveness of the F-14A HUD substantiates the requirement that a precise and timely display of all critical flight information be available on a HUD for effective and optimum use of the ski jump.

Structural Loads

Significant structural loads are imposed on an airplane during transition of the ski jump ramp. The stringent structural design requirements of carrier based airplanes, such as the T-2C and F-14A, permitted operations from the ski jump. In fact, the design of the ramp tested was somewhat dictated by the anticipated structural loading of the test airplanes. The principle area of concern was landing gear loads. To alleviate wing bending moments, no wingtip tank fuel was carried in the T-2C and no internal wing fuel was carried in the F-14A. The desire to start ski jump launch operations as close to normal field operation airspeeds posed a dilemma in that the maximum loads would occur during initial ski jump takeoffs. The lighter gross weight ski jumps were obviously conducted first. An example of T-2C landing gear loading during ramp transition is shown in figure 13. It can be seen that the actual and predicted nose gear loads were in substantial agreement but with slightly more speed effect than anticipated. The main gear loads were substantially lower than predicted with less load increment for increasing ramp angle. It is felt that the decreased main gear loading was due to inability to accurately predict wing lift during the takeoff run. The use of full flaps tended to accentuate this trend as well as slightly increasing the nose gear load. The limiting landing gear load/stroke was on the nose gear. On the 6 deg ramp, the maximum nose gear compressions were achieved at an airplane gross weight of 11,800 lb and ramp speeds of 85 to 96 kt. While maximum compression was approached or achieved, it was determined that no detrimental bottoming occurred based on review of telemetry data. Maximum nose gear deflections were also achieved during several flights on the 9 deg ramp.

For the F-14A, actual measured loads validated the general magnitude of the simulation predicted loads but not the trends as a function of the launch variables. The inability to establish trend data is attributable to three factors. First, the very limited number of launches precluded repetition of launch conditions needed to establish data averages. Second, the very small allowable variation in launch speeds prevented determination of the speed effect on launch loads. Third, the unexpected "galloping" of the nose during the run-in to the ramp caused significant random variation primarily of the nose gear loads. Neither the nose nor main gears reached limit load or stroke during the test program. However, the nose gear would become the limiting factor for launches at increasing speeds. The most noticeable speed effect was seen on the 9 deg ramp with increasing launch speeds causing both increased nose and main gear loads. The effect of increasing the gross weight from 48,000 to 55,000 lb on the 9 deg ramp caused only slight increases in landing gear loads. The effect of speed on gear loads on the 6 deg ramp was masked by the random nature of the data. However, the increase of landing gear loads with increased gross weight was clear.

Most notable to the pilot during the ramp transition is the incremental normal acceleration. Peak incremental normal accelerations obtained during ramp transition are presented in figure 14. Accelerations experienced by the aircrew are higher. For the higher F-14A ramp speeds, the aircrew ensured proper seat posture prior to takeoff.

As stated earlier, the structural design requirements of U.S. Navy carrier based airplanes allowed operation from the ski jump ramp as it was designed. Airplanes operated by other services could be operated from ski jump ramps which have a lower curvature. However, to obtain any desired ramp exit angle, the ramp angle being the dominate factor in performance gains, the ramp length and height would be increased. Computer simulation could provide optimum ramp geometry for other airplanes. In addition, data need to be collected on wing structural requirements for carriage of internal fuel and externally mounted stores.

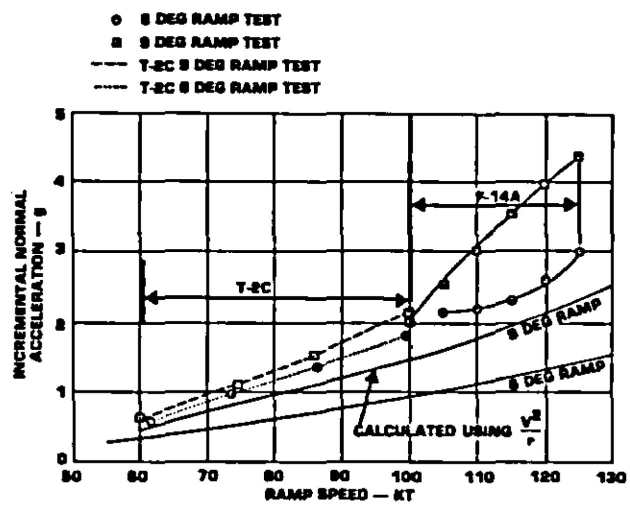

Figure 14 - Maximum Incremental Normal Acceleration During Ski Jump Ramp Transition Measured at CG

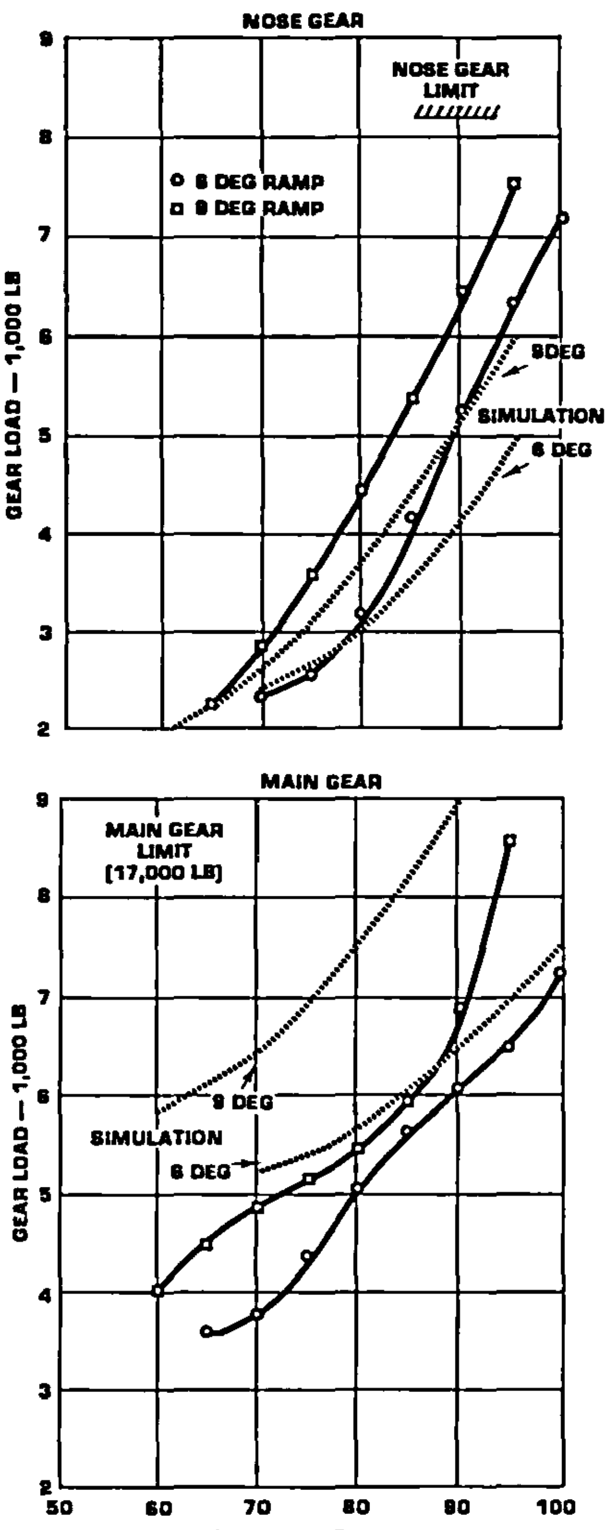

Figure 13 - T-2C Structural Loads 10,000 lb Gross Weight

Summary

Ski jump launch operations with current fixed wing airplanes are possible. The greater than 50% reduction in takeoff ground roll obtained with the T-2C airplane clearly demonstrates the performance gains that are achievable with a ski jump. The performance capabilities with the F-14A were never clearly demonstrated. From a ground handling and flying qualities standpoint, a ski jump takeoff is a maneuver that is easier than a normal field takeoff. For the majority of both T-2C and F-14A ski jump launches, longitudinal trim was set to provide a stick free flyaway capability. However, the difficulty in setting longitudinal trim dictates the requirement that any operational CTOL ski jump airplane have an accurate, readable, and repeatable trim system. Three axis stability augmentation is necessary. To permit the pilot to monitor and optimize flyaway performance and technique, a HUD is essential. Structural loads during ramp transit were well within the design limits of the two test airplanes. Noncarrier based airplanes, generally involving less stringent landing gear strength requirements, could be operable from a ski jump ramp with a increased radius of curvature ramp design.

Computer simulation of the ski jump is an invaluable tool. Simulation predicted performance gains were slightly conservative. Additional efforts need to be expended in the area of landing gear structural modeling. Simulation provided the pilot with the ability to qualitatively access ski jump launch characteristics. Motion based simulations provided the best cues. Emergency conditions, such as an engine failure, could be induced, allowing refinement of single engine techniques and procedures.

Future Plans

Ski jump testing with the T-2C and F-14A is complete. Follow-on test programs are planned which include the F/A-18A and S-3A airplanes. Two sessions of computer simulation have been completed with the F/A-18A. Flight tests are scheduled to start in July 1983. The S-3A is still in the advance planning phase with no firm schedule dates at this time.

DEVELOPMENT AND QUALIFICATION TESTING OF S-76 HELICOPTER TAKEOFF AND LANDING PROCEDURES FOR REDUCED FIELD LENGTH

by

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Abstract

During the basic certification of the Sikorsky S-76A helicopter, standardized piloting procedures were developed for normal dual engine takeoff and to account for the occurrence of a single engine failure anywhere along the prescribed takeoff flight path. Total rejected and continued takeoff distances with one engine inoperative were verified during initial certification flight tests and subsequently published in the Rotorcraft Flight Manual (RFM). In order to enhance the operational capabilities of the S-76A from confined areas, a flight test program was undertaken to develop and qualify a vertical takeoff procedure for reducing field length requirements.

A discussion of the design and conduct of that test program which resulted in CAA approval of S-76 Vertical Operation from Ground Level Heliports in full compliance with BCAR performance criteria for Group "A" Rotorcraft follows.

Nomenclature

AGL	-	Above Ground Level
BCAR	-	British Civil Airworthiness Requirements
CAA	-	Civil Aviation Authority
CDP	-	Critical Decision Point
FAA	-	Federal Aviation Administration
HDP	-	Helicopter Dynamic Performance
LDP	-	Landing Decision Point
Nr	-	Rotor RPM in %
OEI	-	One Engine Inoperative
RAPID	-	Real-Time Acquisition and Processing of Inflight Data
RFM	-	Rotorcraft Flight Manual
SAS	-	Stability Augmentation System
TM	-	Telemetry
V_2	-	Takeoff Safety Speed
WAT	-	Weight-Altitude-Temperature

Introduction

Sikorsky Aircraft initiated the S-76 program in 1974 to develop a modern, high performance, twin engine light helicopter for commercial applications. Initial FAA and CAA transport category certification approvals were granted in 1978 and 1979 respectively. The requirement to expand CAA certification approval to include reduced field length takeoff and landing procedures soon became apparent to improve the suitability of the S-76A for European commercial applications. A brief qualitative flight assessment was made on an S-76A helicopter during May 1981 to examine various procedures to obtain field lengths significantly less than those specified in Reference (1). Dual engine takeoffs were conducted using both vertical and 15 knot oblique climbouts, with simulation of single engine failure along the climb out flight path followed by recovery to continued flight. These tests indicated that a gross weight of approximately 9,000 pounds at sea level altitude and 20°C ambient temperature appeared feasible as a foundation for constructing a preliminary WAT curve. It was further learned that the previously established 65 knot takeoff safety speed (V_2) could be significantly reduced to 35 knots to provide reduced gross weight performance attributes favorable to shortening field lengths while complying with regulatory criteria. In addition it was determined that a dual engine vertical takeoff procedure, which is an obvious advantage when attempting to minimize field length requirements, should be pursued rather than the oblique takeoff/climb. These provisions were applied to a follow on development and qualification program which was initiated in July 1981 to demonstrate the potential of S-76A helicopter operating up to 9,000 pounds gross weight within the shortest possible field length bounds achievable. A basic S-76A configuration was instrumented to obtain desired quantitative data and a test program flown at the Sikorsky Development Flight Center in West Palm Beach, Florida.

Instrumentation

The test aircraft was equipped with a real time telemetry monitor package and a 14 track wide band magnetic on board tape recording system which recorded and displayed measurements crucial to maneuver validation. Key parameters which provided safety of flight information and assist to the pilot in maneuver development were monitored via telemetry by flight test personnel during all test flights. Cockpit measurements were obtained visually from pilot/copilot normal flight instruments. In addition, fuel quantity was monitored to track gross weight changes due to fuel burn-off. Sikorsky's RAPID facility provided 24 hour data turn-around for timely procedural change decisioning, and hard copy time history and tabulated data formats for subsequent detailed study.

A ground based Automax camera was used to record time sequenced events during each maneuver. The 35 mm film records were processed to obtain height and distance along the flight path. An externally mounted light on the aircraft provided the means of synchronizing data with on board recorded measurements. Figure (1) is an illustration of the data processing system and sequence.

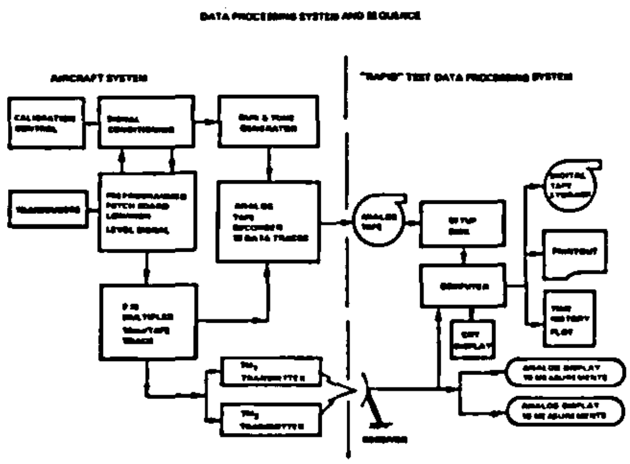


Figure (1)

Test Design

During the limited qualitative flight assessment of takeoff procedures for achieving reduced field lengths, semi-quantitative flight profile, flight speed, and engine power data were obtained from ground based observers and pilot knee board records for correlation with Sikorsky's Helicopter Dynamic Performance (HDP) flight path computer mathematical model. Takeoff fly-away distances were reasonably matched and the HDP was then utilized to project takeoff and landing profiles within dual/single engine operational capabilities and limitations of the helicopter. The proposed procedures for verification by flight testing incorporated a dual Critical Decision Point (CDP_1 and CDP_2) concept described as follows:

Commencing each maneuver with a dual engine vertical lift off from a five foot hover condition using the prescribed power, not to exceed takeoff power, and at a power on rotor speed of 107% Nr, a single engine power failure occurring during the vertical climb out would dictate one of the following actions:

1) Up to CDP_1 (precise height to be determined from flight tests), landback vertically.

2) At CDP_1, the pilot has the option to landback vertically or rotate the aircraft forward to develop forward motion and descend to make a roll on landing.

3) Between CDP_1 and CDP_2 (precise heights to be derived from flight tests), the pilot must rotate the rotorcraft forward to develop forward motion and descent to make a roll on landing.

4) At CDP_2, the pilot rotates the aircraft forward to develop forward motion and has the option to either make a roll on landing or accelerate to V_2 and climb out for continued flight.

5) Above CDP_2 the pilot applies controls to achieve nose down attitude and accelerates to V_2 and continued flight.

Each maneuver would be subjected to a normal pilot recognition time (pilot time delay after engine cut) of at least one second for possible future FAA certification approval. Upon failure of one engine it is noted that compensating power maybe applied from the remaining engine up to maximum contingency (2½ minute) power level to execute recovery to either a rejected takeoff or continued flight. An engine topping check is therefore required for the on-line engine on the day of scheduled testing as a preflight qualification.

To develop the test WAT curve, a potential test weight-altitude-temperature relationship was derived from analysis of the qualitative test flights and the Helicopter Dynamics Performance Program. This condition was then applied to the Sikorsky Energy Method S-76 Predicted Performance Program, and forward flight climb performance was computed at the projected V_2 speed of 35 knots using maximum contingency power at the 96% Nr optimum OEI climb rotor RPM. The resultant 6.5% net climb gradient, taken as a fixed level of performance, was then used to compute variations of weight as a function of ambient conditions. The provisional WAT curve (Figure 2) was then prepared for use during follow on tests to qualify maximum operational gross weights and to schedule aircraft loadings for initial takeoff. Torque available charts, based on engine manufacturers specification horsepower levels, were also prepared and are presented in Figure (3a and 3b). These include a topping chart at maximum contingency rating at 96% Nr, used during maneuver recovery, and rated takeoff power at 107% Nr to select applied torque levels for dual engine takeoffs.

Test objectives during the ensuing flight program were to establish specific CDP_1 and CDP_2 heights and to develop and qualify a repeatable piloting procedure for achieving safe, repeatable maneuvers within the shortest possible field lengths, while operating at limiting WAT conditions and in compliance with the British Civil Airworthiness Requirements of Reference (2).

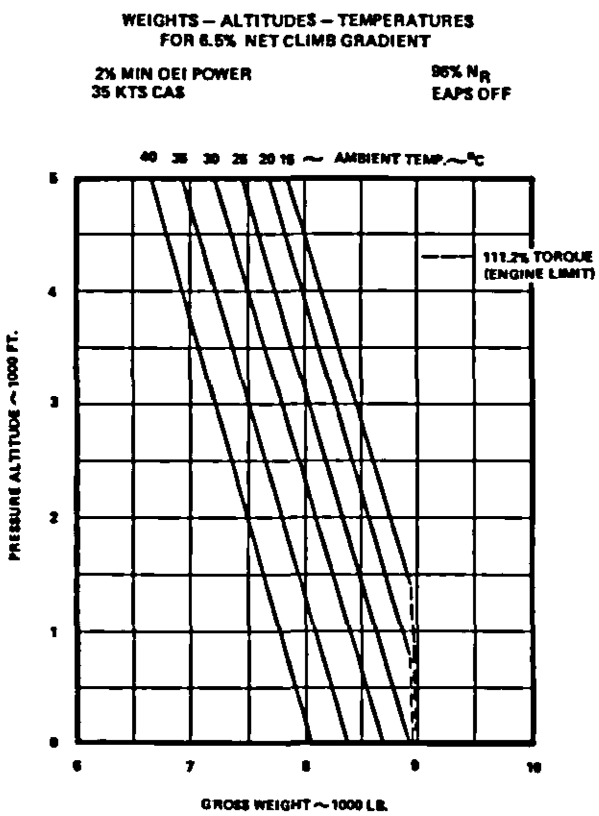


Figure (2)

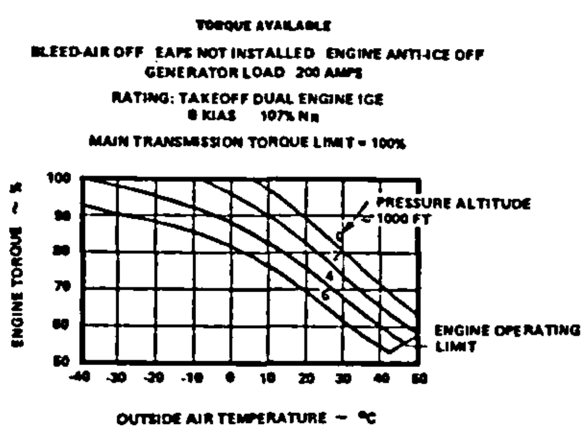


Figure (3a)

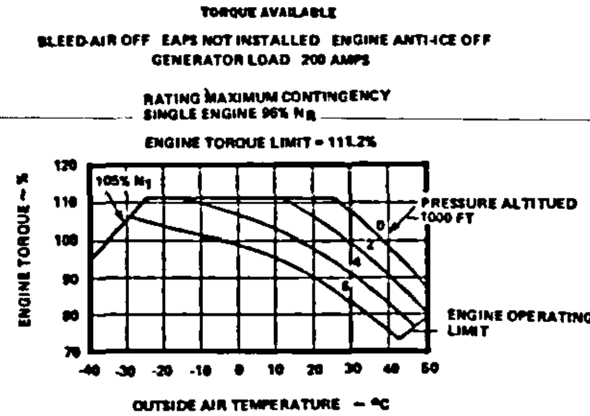


Figure (3b)

Test Procedures

In preparation for scheduled flight, the helicopter was ballasted and weighed to the conditions of Figure (2) and preflight instrument and TM calibrations performed. Assigned test personnel were positioned at the telemetry station, remote Automax camera site and in the mobile ground vehicle. Radio communication was then established between the aircraft, control tower and pertinent test personnel. After engine start and prior to initial takeoff, the pilots conducted a power assurance check on each engine. The on-line engine, which was not to be cut, was trimmed and the speed selector locked into position, so as not to exceed the engine manufacturers contingency power rating of Figure (3b). On-board instrumentation and Automax camera records were activated simultaneously for each maneuver by using a countdown radio sequence. Crucial parameters such as engine torque, landing gear loads, pitch attitude, rotor RPM, wheel clearance and aircraft control positions were continuously monitored and updated by telemetry. Pilots used the cockpit radar altimeter as the primary height source and reference for single engine cuts along the flight path. Target gross weights were maintained to within ±100 pounds tolerance by monitoring cockpit fuel gages and periodically adding ballast to compensate for fuel burnoff. Activities were scheduled and limited to wind conditions of less than three knots.

In order to establish the upper limit of CDP_1, after dual engine vertical takeoff, single engine malfunction was simulated at successively increasing heights along the vertical flight path in build up fashion sequence from 5 to 100 feet above ground level, then making a vertical landback. The lower limit of CDP_1 was established by simulating an engine failure during vertical climb out at 100 feet (tentative CDP_2 height) with successive lower engine cuts along the dual engine flight path and rotating the helicopter forward to develop forward motion and descent to roll on landing. Optimized CDP_1 was the lowest point at which either type of landback maneuver (vertical or pushover) could be made with sufficient controllability/control margins, acceptable rotor droop characteristics, and non-critical landing gear loads. The height of CDP_2 was established along the vertical climb out path primarily by initiating a pushover of the helicopter after engine cut to develop forward acceleration to takeoff safety speed and to continued forward climb out. Optimized CDP_2 height was based on pilot's judgement of flight safety and test maneuver repeatability, as well as complying with regulatory criteria for minimum ground clearance and the ability to landback within a reasonably short distance of the takeoff point.

Piloting technique development was a continuing process during the flight program in an effort to maximize helicopter performance by optimizing inputs such as torque per engine during vertical climb, rate/amount of helicopter pitchover after engine cut, definitive collective time/rate application after engine cut and during descent to flare, and proper control inputs after engine cut at CDP_2 for satisfactory transition to V_2 and desired climb gradient along the takeoff flight path.

After development and qualification of the vertical procedures previously described, investigative flights were made at significantly reduced gross weight, at most aft center of gravity, and with SAS off to demonstrate satisfactory performance at these conditions. Headwind and crosswind accountability studies were also conducted with the perfected maneuver technique. In separate qualitative trials, takeoffs and landbacks were performed satisfactorially on and over a grass surface area without adverse effects. During a night flight the radar altimeter was evaluated as the primary visual height reference under limited visibility supplied by external aircraft lights. Adjunct to procedural development to achieve reduced field lengths, required change in single engine landing approach path was made by flight experimentation of entry speeds and descent rates to accommodate landing distances within the rejected takeoff area. A modified landing approach and landing technique was subsequently developed and qualified.

Test Results and Discussion

General

A total of 24.3 flight hours were flown to develop and collect qualification data for vertical takeoffs and landings during the reduced field length development program. An additional 2.7 flight hours were flown by a CAA flight test team to evaluate the perfected technique.

Recorded measurements obtained from the onboard data acquisition system and Automax camera were processed through Sikorsky's computer facility and the data presented in time history data formats. Typical data plots shown in Figure (4) which were developed for each maneuver consisted of the following measurements: rotor speed, engine torques, airspeed, control positions, pitch attitude and aircraft heights and distances. In addition, landing gear loads were monitored for all touchdown maneuvers for comparison with design limit load criteria. Highlights and significant measurements obtained from the time histories pertinent to each of the vertical rejected takeoffs, pushover rejected takeoffs, vertical takeoff go

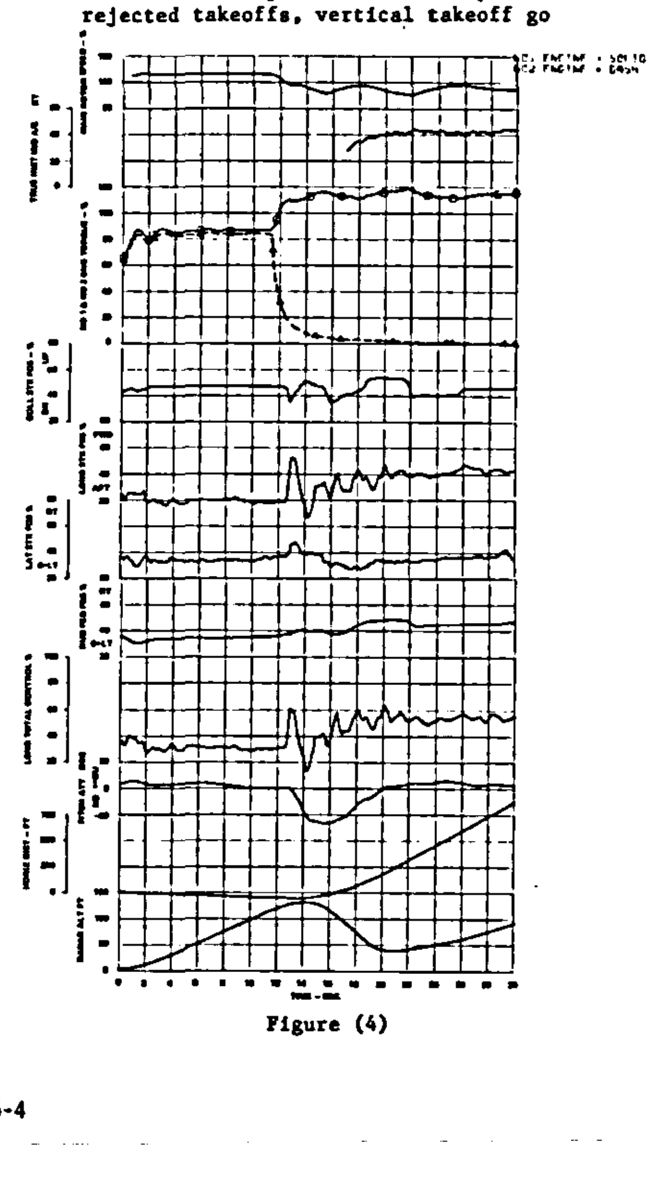


Figure (4)

arounds, single engine landings and balked landing maneuvers were compiled in tables for review purposes. Results of these summaries are discussed in subsequent paragraphs.

Vertical Takeoff Technique Development

Initial test efforts were directed toward experimentation with piloting procedures while establishing effective CDP heights during single engine vertical landbacks, rejected takeoffs to roll on landing and pushover to acceleration for continued flight. Single engine vertical landbacks were evaluated at heights up to 105 feet, but in-house discussions regarding associated piloting skill requirements prompted a decision to lower CDP$_1$ height. Pushover rejected takeoffs to roll on landings were, therefore, investigated commencing at 100 feet and at consecutively lower altitudes until pilots assessment made either type of maneuver acceptable at a height consistent with 40 foot radar altitude for CDP$_1$. Multiple maneuvers were flown to confirm repeatability. Takeoffs to single engine continued flight were then evaluated to establish a CDP$_2$ height which would give a consistent 10 foot ground clearance with engine cuts being made at radar altitudes from the 85 to 95 foot range. The 10 foot clearance, consistent with proposed CAA helicopter code of practice, was originally proposed as a manageable deviation from the regulatory criteria of 35 feet. A review of the takeoff procedure was prompted, however, when maneuvering inconsistencies and lack of repeatable 10 foot ground clearance occurred. It was subsequently reasoned that a 35 foot clearance would be obtained by increasing the CDP$_2$ height to approximately 120 feet. Technique development, however, also required acceptable pushover rejected takeoffs to roll on landing be made from the prescribed CDP$_2$ height. After repeated test maneuvers the selected 120 foot CDP$_2$ height was confirmed.

As testing progressed it was found that better uniformity of piloting technique and repeatability of maneuvers could be obtained when operating at a fixed vertical climb rate during the takeoff sequence through CDP$_2$. Detailed study indicated that a climb performance level of 600 feet per minute was appropriate. Figure (5) was then prepared to account for maneuver sensitivity to gross weight/dual engine vertical climb out torque available and is intended to be used in conjunction with the WAT curve. Additional reviews and studies resulted in final adjustments, Figure (6), to the WAT curve such that both climb performance minima of 6½ percent net climb gradient at 35 knots using maximum contingency power or 600 feet per minute dual engine vertical climb using takeoff power would be met when the helicopter was loaded accordingly.

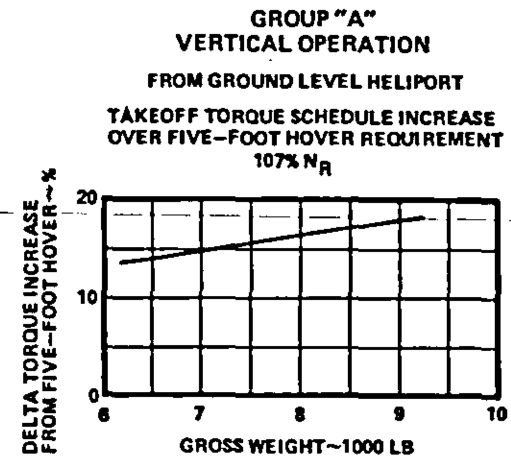


Figure (5)

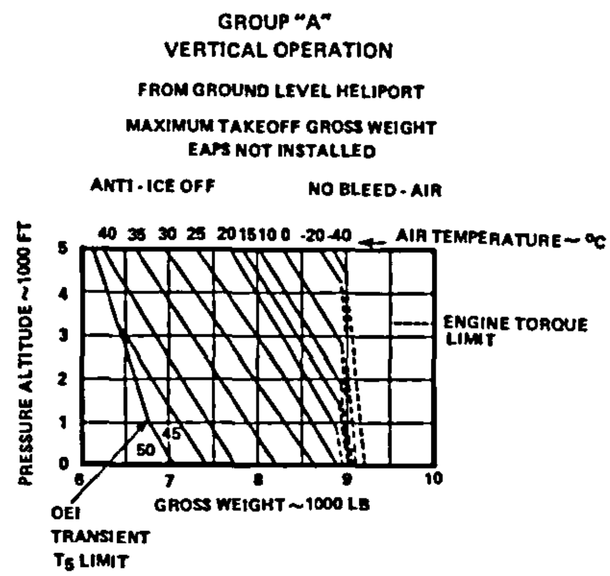
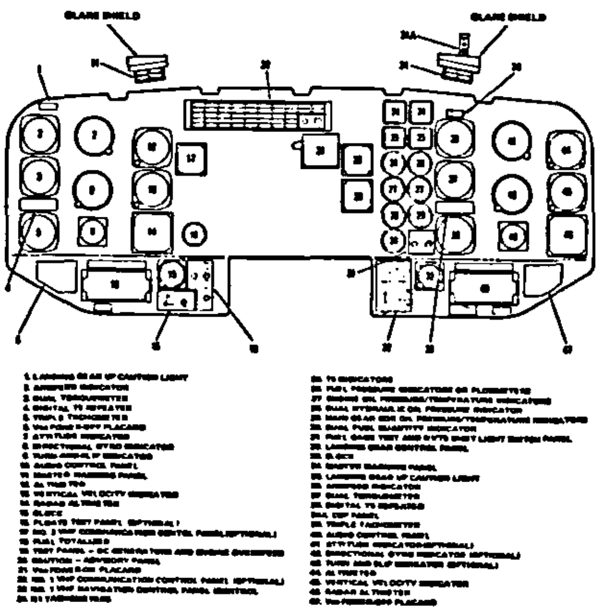


Figure (6)

It was found during the rejected takeoff maneuvers that the rotor RPM would occasionally droop below the established 82% minimum single engine power-on transient limit at the point of touchdown. Subsequently, lowering the minimum (down to 75%) single engine power-on transient RPM for vertical operations was substantiated by review and analysis of rotor system dynamic component structural data recorded during these maneuvers.

A set of small amber lights was developed during the course of testing and inserted into the pilots instrument panel, Figure (7), for evaluation so that the final developed technique could be flown by a single pilot. These lights were actuated through radar altimeter reference. The bottom light illuminated at 40 feet and the upper at

120 feet. At altitudes above 120 feet both lights are on and may be switched off by the pilot. The lights were found to be an acceptable pilot reference.



INSTRUMENT PANEL – TYPICAL VFR CONFIGURATION

Figure (7)

Landing Technique Development

Single engine approach and landing procedures were adjusted during the course of the development program. These tests were conducted to provide shorter landing distances consistent with reduced takeoff field lengths. A compatible LDP was established at 100 foot AGL height with landing approach speed of 40 knots at a descent rate of 500 feet per minute. Balked landings, from this point were flown with no apparent problem. An average height loss of 40 feet occurred between LDP and recovery which was well above minimum acceptable ground clearance criteria.

Field Length Determination

Quantitative assessment of field length requirements was made by graphical analysis of applicable height, distance and performance related items extracted from time histories of maneuvers flown. Data points representing the maneuvers flown during development and qualification testing were plotted as a function of engine cut height as the fundamental parameter. Figure (8) presents pushover rejected takeoff point to point distances to roll on landing from engine cut height to touchdown, ground roll distance and total distance as the summation of air and ground distances. When abnormally long air/ground distances are discounted, the predominance of data originating at similar engine cut heights indicated that a 400 foot distance would be representative for any height from CDP_1 to CDP_2. After adding appropriate allowance for the helicopter length dimension, a total prepared rejected takeoff area of 450 feet was established which would apply to all combinations of allowed takeoff conditions. Summary data for single engine continued flight maneuvers, presented in Figure (9) indicates that minimum ground clearance criteria of 35 feet is attained after engine cut heights at CDP_2 levels at/or greater than 120 feet with distance to minimum height occurring at 250 to 350 feet. The criteria for establishing takeoff space required (distance to 50 foot recovery height or to 3% net climb gradient, whichever is greater), are included in Figure (9) and indicate that the 50 foot recovery height is the deciding factor in determining a takeoff space requirement of 700 feet point to point and 750 feet total. This distance would apply to all combinations of allowed takeoff conditions. Summary data for single engine landings are plotted in Figure (10) and show air distance, follow-on ground roll distance and average descent rates versus measured height at engine cut during single engine approach to landing. Predominant data indicate that air distances and ground roll distances are 450 feet and 175 feet respectively for the established LDP. The total horizontal distance of 625 feet is based on a point to point distance and must be increased to account for aircraft length. Operationally, the specific landing space of 675 feet from 100 foot height necessitates consideration of obstacle clearances along the landing approach path to make the maneuver compatible with the field boundary dimensions established for rejected takeoff space required. The final Group "A" vertical operation takeoff and landing profiles are illustrated in Figures (11) and (12).

Wind Accountability

Sensitivity of these procedures to wind was assessed by flying takeoff and landing maneuvers in up to 20 knots of wind. Vertical rejected takeoffs from 40 feet (CDP_1) were flown in direct left and right crosswinds of 15 knots, with pushover rejected takeoffs, continued takeoffs and OEI landings flown in 15 to 20 knot headwinds which were 020 to 030 degrees relative to actual flight path. Flight characteristics were satisfactory for all maneuvers while headwind components greatly reduced field length requirements. These data were correlated with the Helicopter Dynamic Performance Flight Path Computer Model, which was then used to develop wind accountability derivatives for the takeoff space, rejected takeoff area and

SUMMARY DATA OF REJECTED TAKEOFF DISTANCE
(PUSHOVER TO ROLL ON LANDING – SINGLE ENGINE)

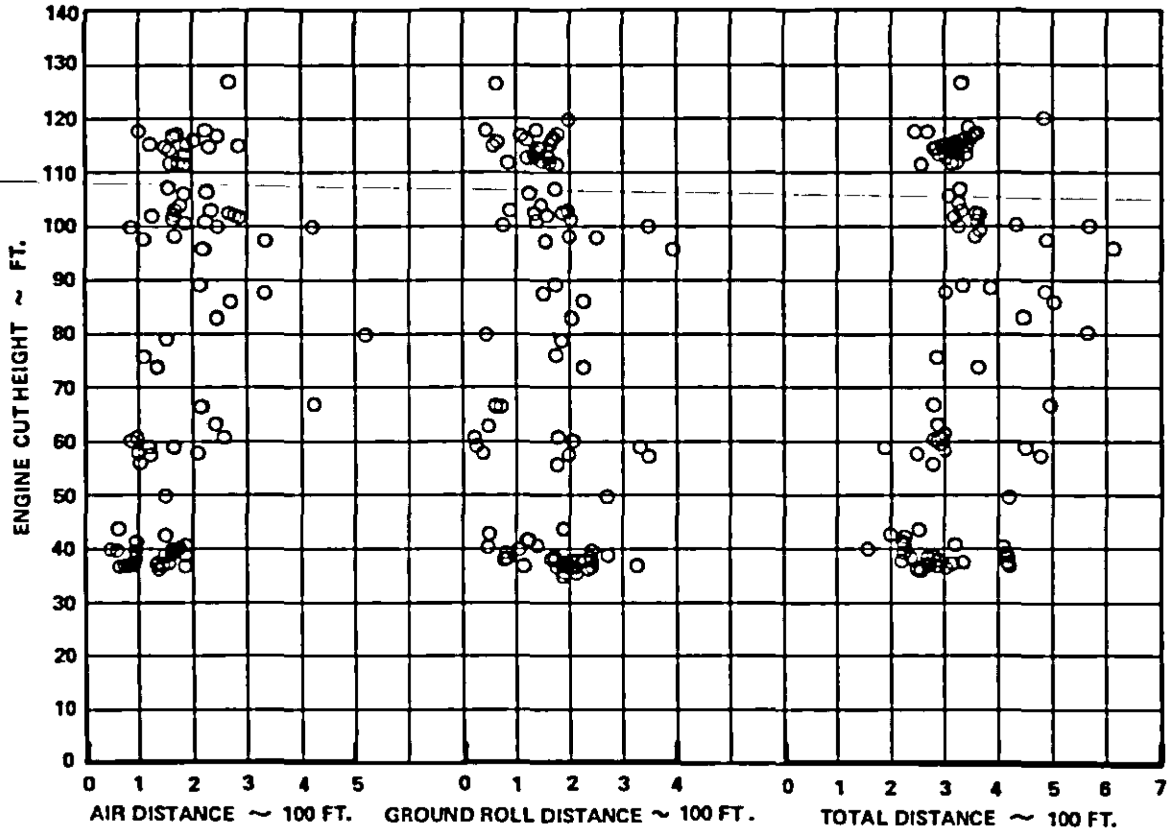


Figure (8)

SUMMARY DATA OF SINGLE ENGINE CONTINUED FLIGHT

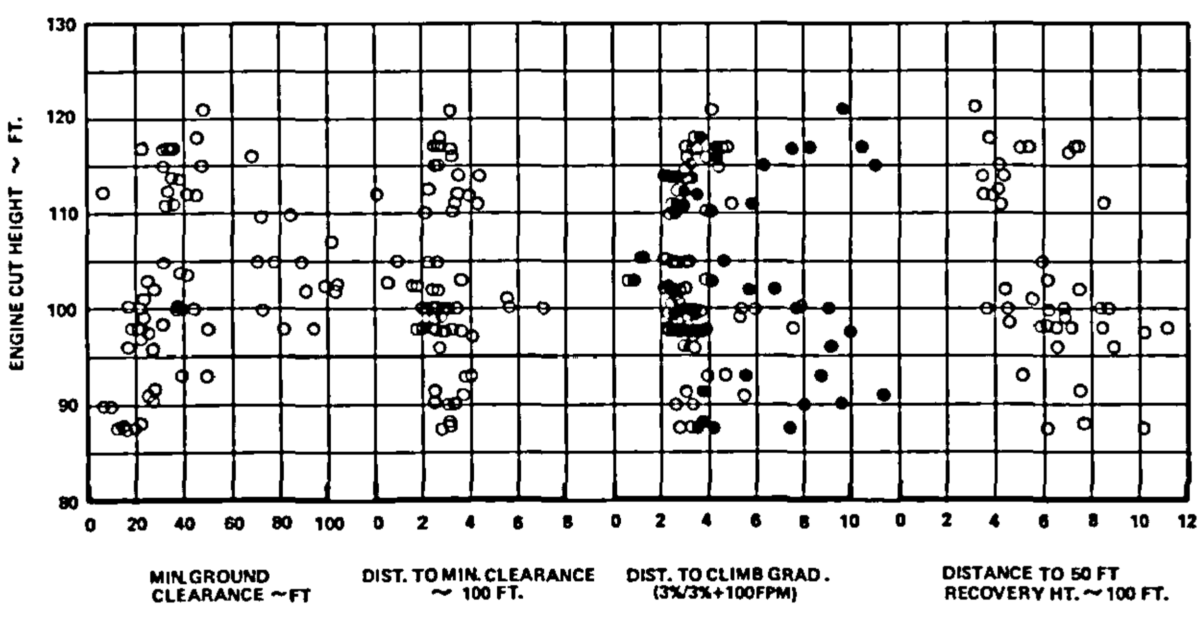


Figure (9)

SUMMARY DATA OF SINGLE ENGINE LANDINGS

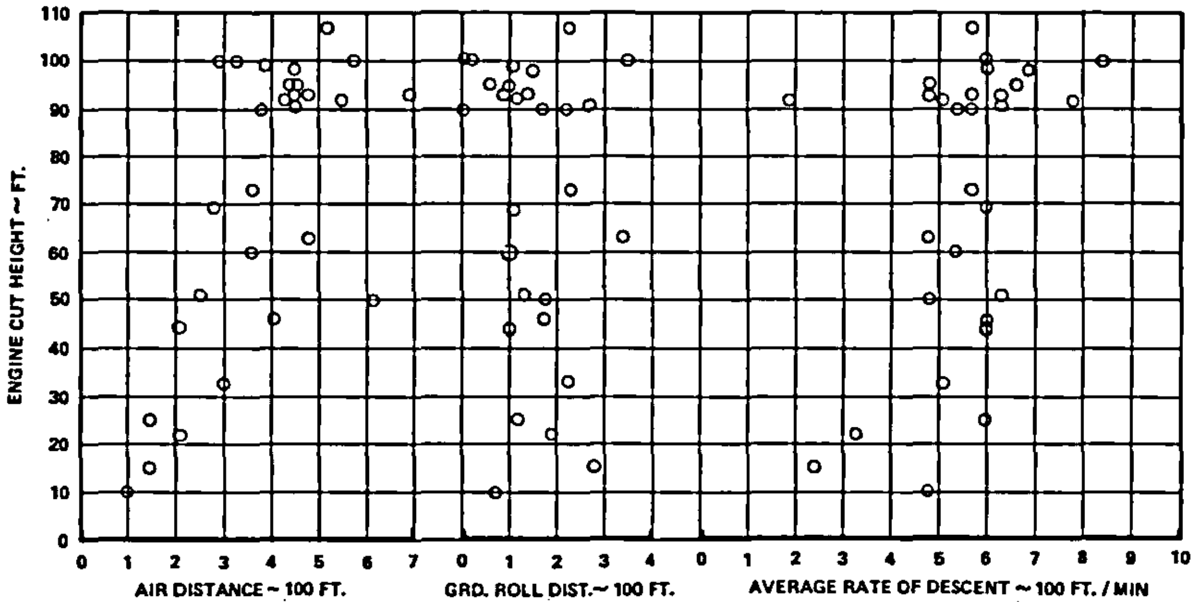


Figure (10)

GROUP "A"
VERTICAL OPERATION

TAKEOFF PROFILE

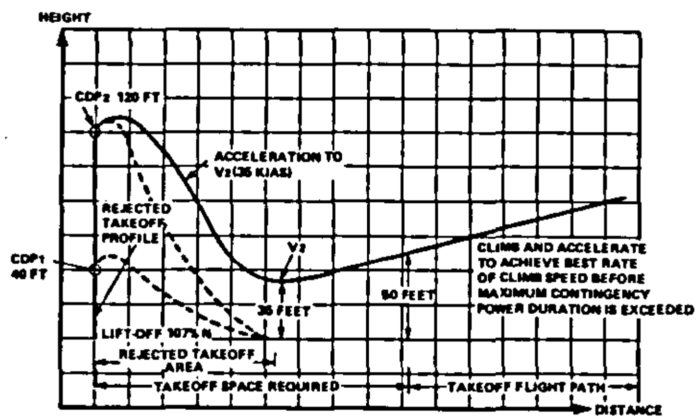


Figure (11)

GROUP "A"
VERTICAL OPERATION LANDING PROFILE

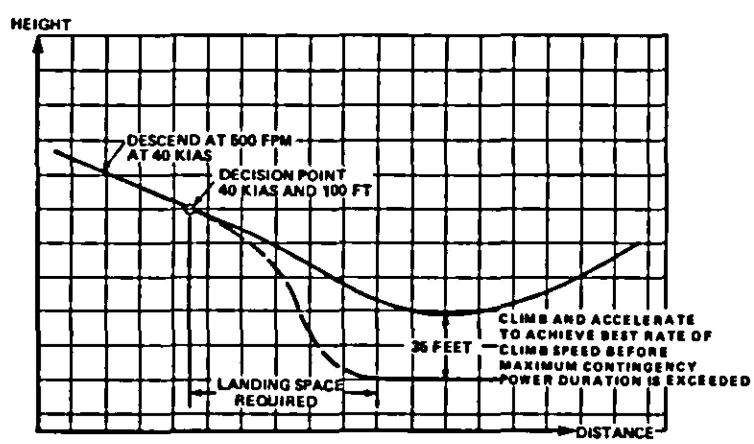


Figure (12)

landing space field length requirements. The established unfactored headwind accountability derivatives for each maneuver expressed in terms of field length reduction per knot of headwind (ft/knot) are 20, 12 and 25 respectively for the aforementioned maneuvers.

CAA Flight Evaluation

A CAA flight evaluation was conducted upon completion of the development program to verify Sikorsky test results. Takeoff go arounds, rejected takeoffs, single engine landings and balked landings were flown at aft center of gravity (worst case), limiting WAT curve gross weight and with SAS turned on and off. A night flight evaluation was also flown to evaluate night lighting requirements and pilot use of proposed radar altitude CDP lights for single pilot operation. Data which were collected confirmed the applicability of the scheduled field lengths which were proposed.

Conclusions

The Reference (3) certification approval greatly enhances the S-76's ability to operate from ground level confined areas at gross weights to 9200 pounds by substantially reducing the prepared runway surface area requirements. The procedure is presently utilized by several European operators.

Acknowledgements

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THE USE OF OIL FOR IN-FLIGHT FLOW VISUALIZATION

By

Robert Curry & Robert Meyer

National Aeronautics & Space Administration - Dryden Research Facility

PAPER NOT AVAILABLE AT TIME OF PRINTING

IN-FLIGHT FLOW VISUALIZATION, A FLUID APPROACH

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ABSTRACT

The tuft method is the in-flight flow visualization technique commonly used by the Douglas Aircraft Company. While the use of tufts generally yields satisfactory flow patterns, recent flow visualization tests performed on the MD-80 aircraft highlighted the following two tuft method limitations: Tuft patterns often are hard to see from a chase aircraft and tufts located in areas of unusual airflow yield obscure or inadequate airflow information. In order to overcome these limitations, a fluid flow visualization technique was developed which generated detailed fixed flow patterns that could later be examined on the ground. Two major obstacles in the development of the fluid flow visualization technique—design of the hardware necessary to dispense the fluid and choice of what fluid to use—were overcome. The fluid dispensing installation was comprised of externally routed tubing connected to a pressurized reservoir. Propylene glycol monomethyl ether, which is commercially available as DOWANOL PM, was chosen as the fluid with the desired volatility and viscosity characteristics at the flight temperature of −40°C. The definitive flow patterns obtained on the MD-80 aircraft upper vertical tail, aft fuselage, engine pylon, and nacelle not only established the success of the new method but also allowed correlation of patterns obtained on wind tunnel models and those obtained under actual flight conditions on the full-scale aircraft. The technique is currently being developed further with the investigation of other glycol ethers as potential fluid candidates. Additionally, multicolor fluids are being considered in order to obtain several flow patterns at different conditions during a single flight.

INTRODUCTION

Ongoing performance improvement programs on the MD-80 have included detailed studies of the airflow about the aft end of the aircraft. The areas of concern were the aft fuselage and the pylon/engine and horizontal stabilizer/vertical fin intersections.

An example of the surface airflow on the aft fuselage of an MD-80 wind tunnel model is presented in Figure 1 and shows a divergence of surface flow-lines aft of the engine pylon. Another wind tunnel model view is presented in Figure 2 which shows a region of nonuniform air flow on the aft upper tail cone. The surface airflow pattern on the pylon and engine nacelle area is presented in Figure 3 and

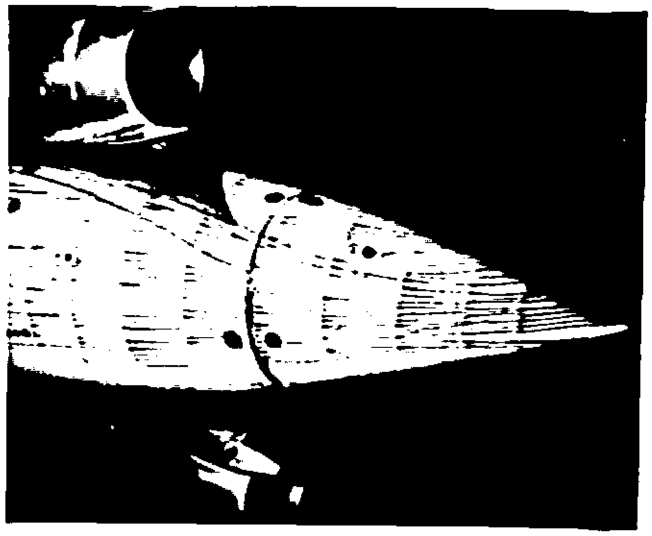

FIGURE 1. MD-80 WIND TUNNEL MODEL SURFACE AIRFLOW PATTERNS AFT OF PYLON TRAILING EDGE

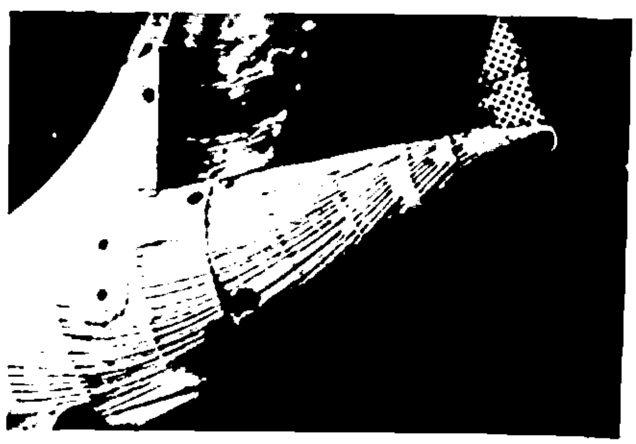

FIGURE 2. MD-80 WIND TUNNEL MODEL SURFACE AIRFLOW PATTERNS ON THE UPPER AFT PORTION OF THE TAIL CONE

FIGURE 3. MD-80 WIND TUNNEL MODEL SURFACE AIRFLOW PATTERNS ON THE NACELLE AND PYLON

shows a region of flow separation or reversal on the engine nacelle aft of the pylon trailing edge. The vertical fin/horizontal stabilizer intersection was of special interest because wind tunnel and flight test pressure survey data suggested the presence of high flow velocities in the wiper plate region immediately below the horizontal stabilizer.

In order to establish the extent to which these flow conditions existed at full-scale Reynolds number conditions, a tuft survey was conducted on an MD-80 aircraft. In addition to the aforementioned regions of unusual surface airflow, the areas around the the nose strake and wing root fillet were tufted in order to verify that no unusual flow patterns occurred in these regions.

The in-flight tuft flow patterns around the nose strake and wing fillet verified that smooth surface flow conditions existed in these areas. Figure 4 illustrates the tuft flow pattern around the wing, and shows that definitive flow patterns can be obtained with tufts when no unusual surface flow conditions exist. The tuft pattern obtained on the vertical fin/horizontal stabilizer gave no indication of the presence of anomalous flow. On the aft fuselage, the tuft patterns (Figure 5) yielded somewhat inconclusive airflow information. Furthermore, the tuft patterns on the side of the fuselage in the region aft of the pylon trailing edge indicated that anomalous flow conditions might exist; however, the boundaries of this region's unusual flow could not be ascertained.

In order to photograph the nacelle-to-pylon intersection in flight, the chase aircraft was equipped with a down-looking periscope interfaced with a 35-mm motion picture camera. The result of this effort is presented in Figure 6. As can be seen, the chase aircraft was unable to photograph the tufts on the pylon-to-nacelle junction with sufficient clarity to define the flow conditions. As a result,

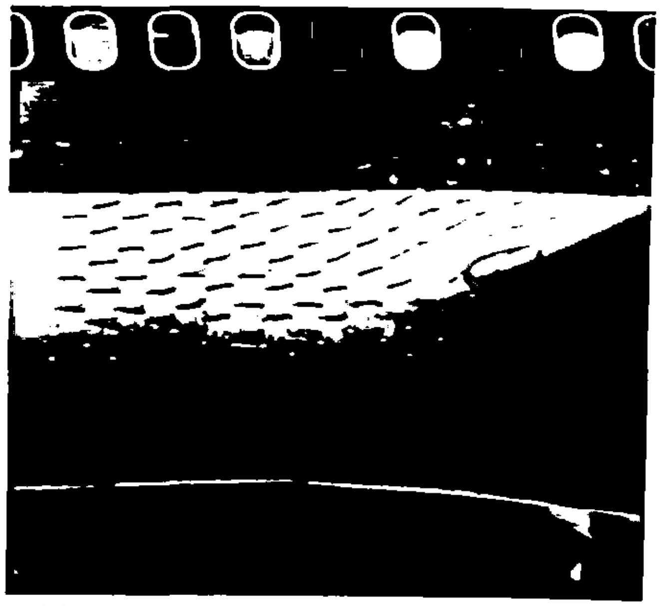

FIGURE 4. MD-80 IN-FLIGHT SURFACE AIRFLOW PATTERNS ON THE WING ROOT FILLET

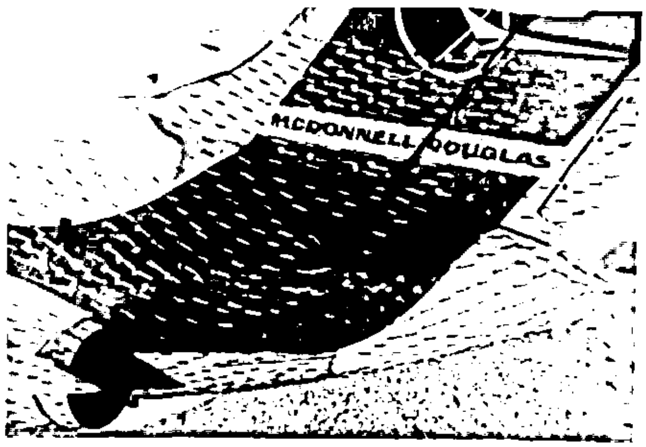

FIGURE 5. MD-80 IN-FLIGHT SURFACE AIRFLOW PATTERNS ON AFT FUSELAGE

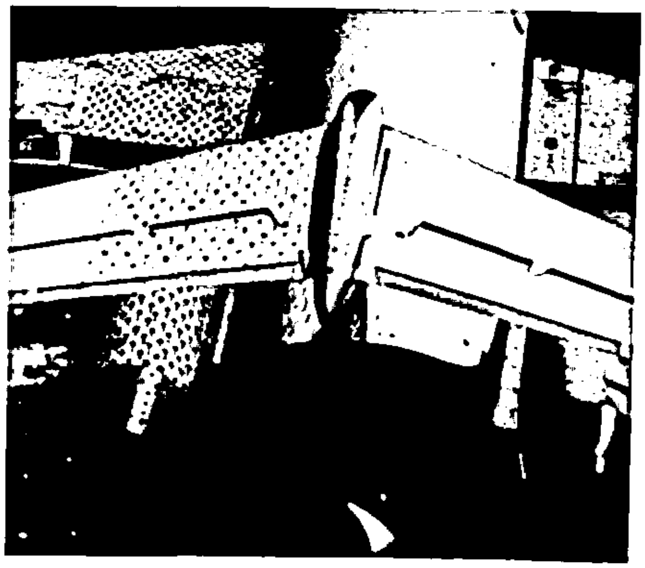

FIGURE 6. MD-80 IN-FLIGHT SURFACE AIRFLOW PATTERNS ON THE NACELLE AND PYLON

no conclusion could be made with respect to the nature of the surface airflow at the nacelle-to-pylon intersection.

The results of the tuft survey show that while the use of tufts generally yields satisfactory information on the air flow characteristics for regions of uncomplicated flow, the tuft method as a flow visibility technique has two limitations: The tuft patterns have to be observable from a chase aircraft during the specific flight conditions of interest and tufts located in areas of unusual airflow yield obscure or inadequate surface airflow information. In order to define the flow patterns on the nacelle-to-pylon intersection and boundaries of the unusual flow on the aft fuselage, a different flow visualization technique was required.

A FLUID APPROACH

BACKGROUND

In 1967, an in-flight flow visualization test was conducted on a DC-9 Series 30 aircraft. For this test, dyed water was ejected from tubes located around the lower half of the aircraft nose. Definitive and fixed surface airflow patterns were obtained on the fuselage during low-speed, low-altitude flight conditions.

It was proposed recently that a similar fluid-flow approach be developed for airflow visualization tests on the MD-80 aircraft. Two problems immediately became apparent. First, because the flow visualization tests would be conducted under high-altitude cruise conditions, a fluid had to be found which would have the necessary viscosity and volatility to flow and evaporate at temperatures of −40°C. Second, hardware had to be developed to dispense the fluid. While the 1967 test used 34 tubes closely grouped around the lower half of the fuselage nose to dispense fluid, the MD-80 requirement was for 64 tubes to dispense the fluid at several widely separated locations; and whereas, previously, cabin differential pressure was sufficient to expel the dyed water from the tubes, a larger pressure source would now be necessary in order to force the fluid through the 18 tubes to be routed to the top of the vertical fin.

It was initially proposed that the tubes be routed through hollow fasteners and onto the exterior surface of the aircraft; however, time and budget constraints as well as the large number of tubes required precluded that particular approach.

Instead, the tubes exited the aircraft cabin through specially fabricated window plugs and were externally routed along the aircraft surface to the prescribed locations.

In order to assure that the fluid would be dispensed evenly, the end portions of the tubes were aligned on the aircraft skin parallel to the direction of the airflow with the discharge openings facing downstream. The tubing used was 1/8-inch OD thin wall stainless steel. This tubing was selected because of its strength and also because its small external diameter would minimize any disturbance of the local surface airflow.

The heart of the fluid dispensing system was a 5-gallon reservoir pressurized by a regulated nitrogen bottle. The reservoir dispensed fluid to two manifolds. One manifold fed the 18 tubes routed to the upper vertical fin and the other manifold fed the remaining tubes routed to the nacelle, pylon, and aft fuselage. Fluid flow from the reservoir to each manifold was controlled by a valve arrangement which allowed fluid to flow under pressure to either of the manifolds. The reservoir/manifold installation is shown in Figure 7.

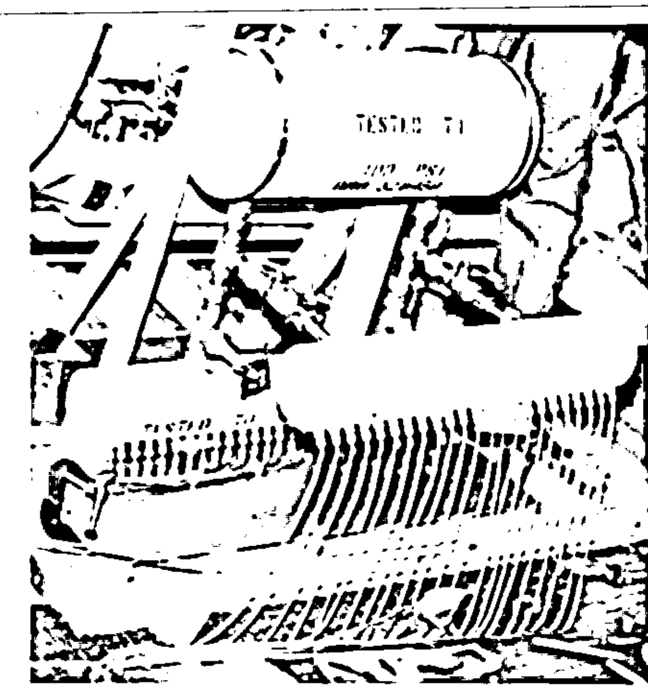

FIGURE 7. RESERVOIR AND MANIFOLD INSTALLATION

FLUID SELECTION

Several types of fluids were initially considered as potential fluid trace mediums, among them freon, alcohol, and glycol ethers. In addition to possessing the necessary viscosity and volatility properties, the fluids had to be nonflammable, noncorrosive and nontoxic. Freon Type 11 was selected because it met all the criteria. In-flight test results revealed (see Figure 8) that while the freon exhibited excellent viscosity characteristics, it was still too volatile at −40°C to be an effective trace medium since it evaporated approximately 5 feet downstream of the point of discharge.

A family of commercially available Dow Chemical glycol ethers were the next fluids considered as potential trace mediums. Viscosity tests of four glycol ethers were performed in the laboratory. The results are presented in Figure 9. As can be seen, DOWANOL EE, PM, and EM all exhibited acceptable and similar viscosity characteristics at a temperature of −40°C. At that temperature, the three glycol ethers had a viscosity similar to that of a light mineral oil. The physical properties of the four glycol ethers evaluated are presented in Table 1. Of these four fluids DOWANOL PM (propylene glycol monomethyl ether) was selected as a candidate fluid because of its acceptable viscosity at low temperatures and also because

FIGURE 8. MD-80 SURFACE AIRFLOW PATTERNS UTILIZING FREON TYPE 11

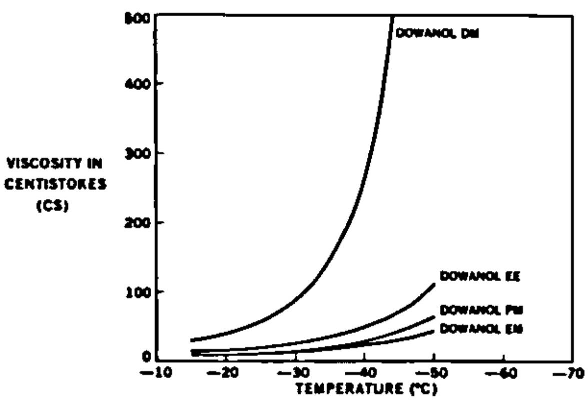

FIGURE 9. VISCOSITY VERSUS TEMPERATURE

of its relatively low toxicity. Although flammable, propylene glycol, if kept at a temperature below its flash point of 96°F, will not be sufficiently volatile to produce an ignitable mixture. Therefore, with proper precautionary handling, the fire hazard associated with this fluid can be virtually eliminated.

AIRCRAFT PREPARATION

A checkout flight was flown to evaluate the suitability of propylene glycol as a fluid trace medium. Before the fluid was dispensed, it was mixed with an organic soluble red dye. The fixed-flow patterns obtained confirmed that propylene glycol was a satisfactory fluid trace medium.

The MD-80 test aircraft was then painted white in the the areas of interest in order to provide contrast between the aircraft surface and the dyed fluid. Another preparatory step was the taping of all seams and gaps in the areas where the fluid was expected to flow. The taping was essential since the fluid tends to collect at any seam or gap, and this could prevent a flow pattern from developing downstream or could divert the fluid away from the area of interest. As a final preparation step, the aircraft was sprayed with a light coat of a solution of non-chlorinated Bon-Ami and water. When dried, a thin coating of Bon-Ami remained that served to act as a dye absorbent and enhanced definition of the flow pattern.

FLIGHT TEST

The test aircraft was flown to an altitude of 33,000 feet and its speed stabilized at 0.76 Mach number. Ambient temperature was −40°C. Fluid was first dispensed to the upper vertical fin region where approximately 20 PSID was required to eject the fluid. The fluid was then dispensed in the aft fuselage nacelle and pylon regions where approximately 8 PSID was required to eject the fluid. The sequence of the surface flow pattern development on the aft fuselage is presented in Figure 10.

As was observed from the chase aircraft, the flow patterns initially had a glossy red tone; however, as the propylene glycol evaporated the red color became flatter in tone. Approximately 2 minutes were required for this event to occur. After the propylene glycol evaporated, the pattern was fixed and now was available for detailed postflight inspection.

TABLE 1. PHYSICAL PROPERTIES

COMMERCIAL NAME	CHEMICAL NAME	CHEMICAL FORMULA	BOILING POINT AT 760 mm HG (°C)	VAPOR PRESSURE AT 25°C (mm Hg)	POUR POINT (°F)	SPECIFIC GRAVITY	FLASH POINT (°F)	TOXICITY THRESHOLD LIMIT VALUE
DOWANOL PM	PROPYLENE GLYCOL METHYL ETHER	CH_3-O-CH_2-CHOH-CH_3	23.7	10.9	−142	0.919	96	NONE ESTABLISHED
DOWANOL EM	ETHYLENE GLYCOL METHYL ETHER	CH_3-O-C_2H_4-OH	29.5	6.7	−124	0.963	105	25 ppm
DOWANOL EE	ETHYLENE GLYCOL ETHYL ETHER	C_2H_5-O-C_2H_4-OH	34.8	5.3	−148	0.928	100	100 ppm
DOWANOL DM	DIETHYLENE GLYCOL METHYL ETHER	CH_3-O-C_2H_4-O-C_2H_4-OH	81.5	0.18	−121	1.020	182	NONE ESTABLISHED

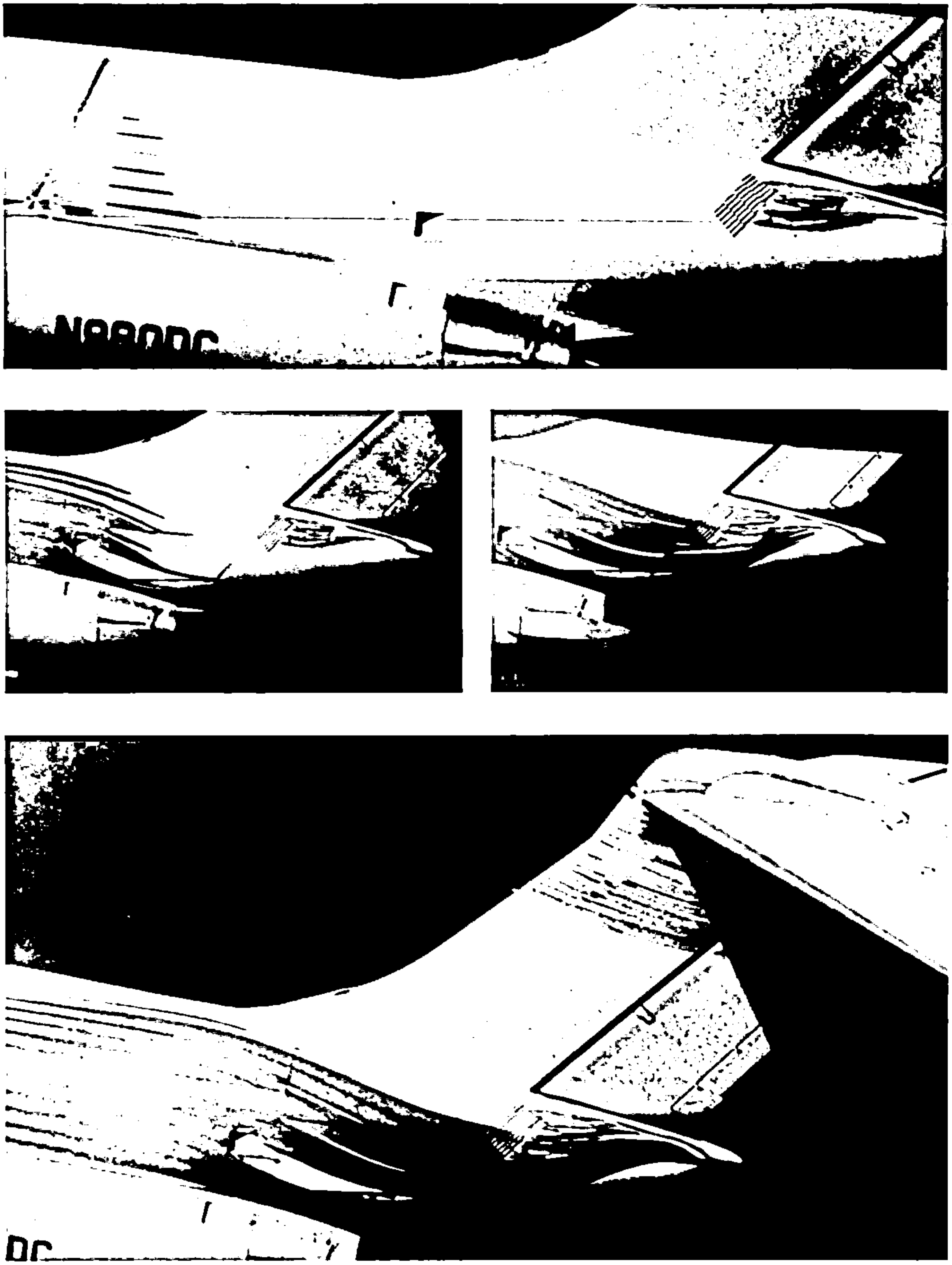

FIGURE 10. IN-FLIGHT SURFACE FLOW PATTERN DEVELOPMENT ON AFT FUSELAGE UTILIZING PROPYLENE GLYCOL

RESULTS

The fixed flow patterns obtained in flight made possible a detailed examination of the flow phenomenon in the areas of concern. The flow patterns on the upper vertical fin showed no evidence of unusual flow although spanwise flow was strong (see Figure 11). This was not a surprising result since spanwise flow typically occurs on any swept surface; in this case the flow was influenced further by the pressure distribution on the horizontal stabilizer during the test condition.

During flight, as the flow pattern was developing on the vertical fin, it became apparent that the airflow was being directed downward and away from the lower wiper plate lip. The wiper plate was removed during the postflight inspection and the red dye was found in the upper vertical fin structure. This suggested that ram air was getting past the horizontal stabilizer leading edge seal causing the wiper plate to bulge thus allowing air to leak overboard. This phenomenon may be a potential source of unwanted drag that is not normally duplicated in wind tunnel testing.

FIGURE 11. MD-80 SURFACE AIRFLOW PATTERN ON UPPER VERTICAL FIN

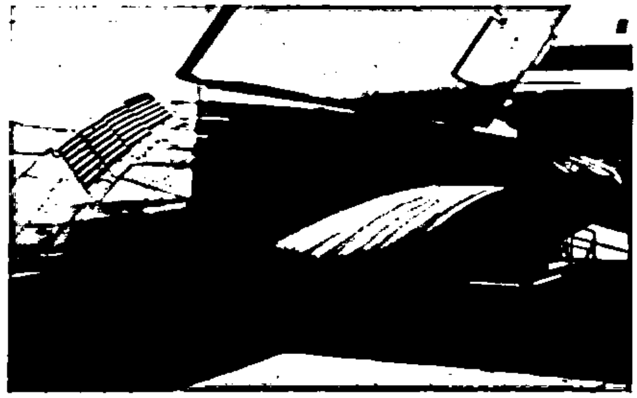

FIGURE 12. MD-80 SURFACE AIRFLOW PATTERN ON AFT FUSELAGE

The aft fuselage flow patterns (Figures 12 and 13) show a region of nonuniform airflow aft of and above the pylon trailing edge, a condition nearly identical to that observed in the wind tunnel tests. A second small region of nonuniform flow exists at the upper aft portion of the tail cone, also as observed in the wind tunnel tests. The white area on the aft fuselage above the pylon is not a separated area but was caused by the fluid being blown off the surface by air exiting the air-conditioning heat exchanger exhaust duct. The pylon-to-nacelle juncture flow pattern (Figure 13), unlike the wind tunnel results, suggests the, absence of regions of separated flow at flight Reynolds numbers. The tape which was used to seal the reverser linkage gaps was torn by relative movement of the nacelle and pylon. However, even these disturbances did not seem to adversely affect the local airflow. It was also noted during postflight ground observations that the nonflush rivets on the aft fuselage served to increase the detail of the flow patterns because the rivet heads collected fluid and then, releasing the fluid, served as a fluid source thus causing individual flow lines to develop downstream of each rivet head.

FIGURE 13. MD-80 SURFACE AIRFLOW PATTERNS ON AFT FUSELAGE, NACELLE, AND PYLON

CONCLUSIONS

FLOW CHARACTERISTICS

A comparison of wind tunnel and flight test flow visualization data yields the following conclusions with respect to surface flow characteristics:

- Wind tunnel and flight test airflow characteristics on the aft fuselage aft of the pylon trailing edge are quite similar.
- The flight test nonuniform flow characteristics on the upper aft tail cone are similar to but of somewhat less extent than that observed in the wind tunnel test.
- Flow in a region of the nacelle-to-pylon juncture, which was shown to be separated or reversed in wind tunnel tests, appears to be normal at full-scale flight conditions.

FLOW VISUALIZATION TECHNIQUE

The in-flight flow visualization technique was demonstrated to have the following advantages over the tuft flow visualization technique:

- The in-flight fluid flow technique allows surface flow patterns obtained in flight under full-scale Reynolds number conditions to be compared directly with those obtained on wind tunnel models.
- The in-flight fluid flow technique identifies the exact boundaries of nonuniform surface airflow.
- The in-flight fluid flow technique can be used to obtain flow patterns in regions which are unobservable from a chase aircraft.
- No chase aircraft is required because use of the in-flight fluid flow technique results in a permanent record of surface flow patterns at discrete flight conditions.

CONTINUED DEVELOPMENT AND TESTING

With the success of the initial fluid flow visualization program, plans are now being developed to obtain flow patterns on the MD-80 wing and nose. Additionally, aerodynamic improvements have already been designed to minimize the nonuniform flow patterns discussed in the previous section, with recent flight tests confirming a drag reduction. In-flight fluid flow visualization will be utilized to establish flow pattern changes due to the improvements. Also planned for the current year is the investigation of the flow properties of other glycol ethers under flight conditions. The one drawback of the fluid flow technique thus far is the restriction of obtaining only a single flow pattern at only one discrete condition per flight. Two different-color dyes will be used for different flight conditions, resulting in two or more flow patterns per flight. The dye colors will be complementary for purposes of obtaining maximum color contrast. This procedure should allow several data points to be gathered per flight thus resulting in a significant reduction in aircraft preparation and flight time and, hence, program cost.

Community Noise Testing - New Techniques and Equipment

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Abstract

The subject of this paper is the development and operation of the Boeing community noise testing system which was used for the Models 757 and 767 noise programs. On-site acoustic and airplane performance data must compare favorably with final data and must be produced in a cost-effective manner. If data were to be proven unacceptable after the completion of a noise test program, the expense and difficulty of re-testing would be enormous.

To accomplish this end, Boeing developed a system which makes extensive use of computer technology to precisely determine the test airplane position in real time, integrate that information with airplane performance data, and telemeter the data to a ground-based test coordination and control station.

The ground station performs an acceptance check on the telemetered data by automatically comparing them to pre-determined positioning and performance limits, then combines them with recorded acoustic data.

The entire process takes place while the test airplane flies a continual traffic pattern circuit, and the optimum time of six minutes between conditions is interrupted only by the soundings of the meteorological airplane which must telemeter an upper air meteorological profile of the test range to the ground station at regular intervals.

This paper will discuss the major components of the Boeing community noise testing system and the technical requirements which dictate their operation and coordination.

Introduction

All new and derivative large transport aircraft must comply with community noise standards established by the Federal Aviation Administration (FAA)[1] and the International Civil Aviation Organization (ICAO)[2]. The standards require that aircraft do not exceed prescribed noise limits specified for takeoff and landing operations.

Demonstration is the only acceptable method of showing Aircraft Type compliance with the standards. The airplane must be tested under prescribed conditions which include specific weather criteria plus airplane performance and position limits. The airplane is flown on a planned flight path over an array of microphones, with airplane position and performance controlled and measured (Figure 1). Once the atmospheric conditions and noise received on the ground are recorded, the airplane data are transmitted to the ground station where all parameters are normalized to reference conditions. Effective Perceived Noise Levels (EPNL) are then established at takeoff, sideline and approach reference locations. Boeing certification noise test programs have always been conducted by these general operating procedures. Prior to the Models 757 and 767 programs, on-site data acquisition, transmission and reduction methods were rather crude and labor intensive. The greatly increased use of computer technology has produced a revolutionary upgrade in efficiency of operations and data quality.

The current Boeing system consists of four independent computer systems. One is carried on the meteorological airplane and is used to record upper air weather data over the test range. The test airplane carries a Microwave Airplane Positioning System (MAPS) which produces positioning data in real time. The Airborne Data Analysis and Monitoring System II (ADAMS II) monitors MAPS data plus all relevant airplane performance data and interpolates these data to as many as four discrete range positions over the test range. Both airplanes telemeter data to the ground station which performs an acceptance check to determine compliance with planned test criteria. Acoustic Data Processor-4 (ADP-4) then normalizes the data for comparative evaluation. The entire data reduction process is repeated with a large, Seattle-based mainframe computer after testing is completed, so on-site data must be accurate as well as immediately available.

A typical time of six minutes between conditions allows condition setup and completion followed by onboard data review and telemetry to ADP-4. Performance and position data are reviewed on the ground by the test director and acceptability is determined within three to four minutes of condition completion, while the test airplane is on the downwind leg of a standard airport pattern. Condition acceptability is based only on airplane performance and position (altitude and centerline displacement) as well as airborne turbulence and ground-monitored wind speeds. All takeoffs and landings are simulated through the use of flight intercepts, and typically the test airplane is airborne for the duration of the day's testing.

The meteorological airplane must make the weather sounding within 30 minutes of any test condition, and the test airplane must leave the airport pattern and remain in a holding area for seven to ten minutes until the meteorological profile is completed. Testing then resumes and is uninterrupted until the next sounding.

The entire noise testing system will be discussed in two separate sections, the airborne systems (test airplane and meteorological airplane) and the ground station (ADP-4). Both airborne systems are linked to ADP-4 by telemetry, which will be dealt with as an ADP-4 subsystem. The third and final section briefly outlines the system operation and compares it with the system used prior to the 757/767 noise test programs.

Test Airplane

ADAMS II

Airborne Data Analysis and Monitoring System II (Figure 2) is a self-contained unit which includes a Rolm Model 1666 computer, disk drive, keyboard, CRT display, and a high-speed printer. It is capable of interacting with other peripherals including video graphics, X-Y plotters, remote digital displays, direct-write oscillographs, and digital-to-analog converters. ADAMS II obtains information for display from the High Speed Pulse Code Modulation (HSPCM) in-flight recording system. Through a variety of special application programs, the system is capable of computing and displaying in engineering units any parameters which are not directly measurable, such as airspeed, altitude, thrust, etc. The most notable program in this case is the Acoustics (AC) program, which displays a condition summary and telemeters that summary to ADP-4.

AC Program

The Acoustics program uses data from the MAPS to determine the airplane position with respect to a ground-based microphone field. It translates the MAPS antenna position information to any preselected position on the airplane using pitch, water line and butt line information, and compares the newly translated position with that of up to four operator-selected microphones as the test airplane flies over a microphone array. When a microphone crossing occurs, AC stores selected airplane data from the instant of crossover for future summarization. When crossings have occurred for all of the selected microphones, or at the operator's command, the crossover data are summarized. The summarized data is then prepared for transmission to the ground station. The operator has the option of displaying the summary before transmission and deleting any spurious data such as spikes or noise.

MAPS

The Microwave Airplane Positioning System (Figure 3) is a portable, state-of-the-art, real-time positioning system developed and built by Cubic Corporation under contract to Boeing. It replaces an airplane-mounted camera theodolite system, which was used for airplane positioning in past noise tests. The camera system was used for the 757 and 767 noise tests as a backup for MAPS and to gather data for further MAPS/camera comparison research.

MAPS consists of an airborne interrogator which utilizes an externally mounted antenna to communicate with 11 transponders positioned on the ground in a pre-surveyed array. Transponder information is sent to a computer which determines the position of the interrogator antenna within three feet. That information is recorded on the separate airborne data system, and is also used to drive the MAPS cockpit display for pilot guidance.

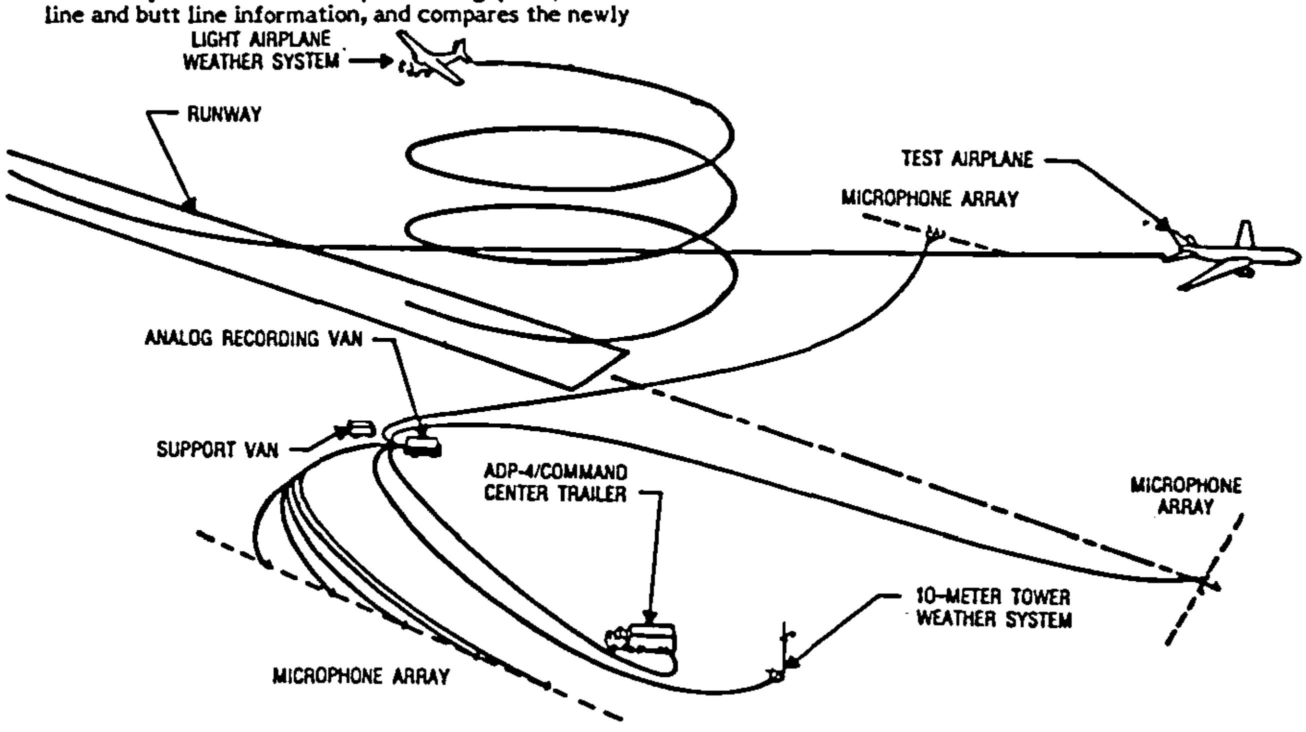

Figure 1 - Community Noise Test Overview

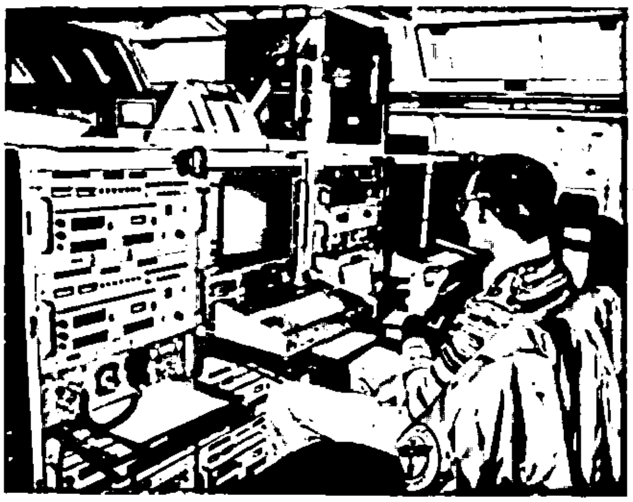

Figure 2 - ADAMS II

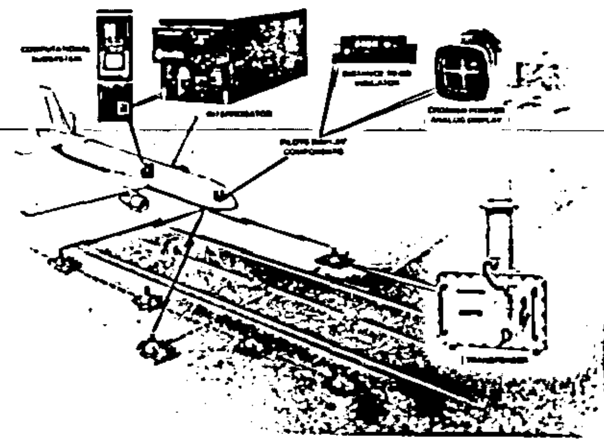

Figure 3 - MAPS

Basic Theory of Operation

Four harmonically related frequencies are used to determine the straight line distance from the transponder antenna to the interrogator antenna. The interrogator transmits these four frequencies (2927.661, 182.979, 11.436 and 0.715 KHz) to the transponder, which coherently retransmits them back to the interrogator. The slant range is determined within 25 centimeters. Doppler shift is used to determine slant range rate of change within 8.78 millimeters per second.

A Kalman filter algorithm processes range and range rate observations, estimates the current aircraft position and velocity, then predicts the condition forward in time for purposes of transponder interrogation selection and pilot guidance.

Computer Subsystem

The computer is a Rolm Model 1666 with 64K words of memory. Associated peripherals include a floppy disk drive unit, a keyboard controller with CRT display, and a high-speed line printer. All these units are of the same type used by ADAMS II, which considerably simplifies the logistics of maintaining spares. The computer subsystem performs all real-time functions including positioning, flight guidance, data transmittal to the airborne data system, and operator control interfacing.

Interrogator and Antenna

The interrogator unit is mounted in a conventional aircraft-type case and is normally installed near the antenna which is externally mounted on the belly forward of the wings.

Transponders

A transponder is housed in a light-weight, weatherproof case and weighs 22 pounds (not including antenna or batteries). It can be set on the ground or mounted on a standard surveyors tripod. The antenna can be mounted to the transponder case or remotely mounted and connected to the transponder by a coaxial cable. Each transponder requires 24 VDC power which is provided by two marine-type, deep-cycle batteries. There are currently 11 transponders in the Boeing inventory, but MAPS has a capacity for up to 19 as presently configured.

Pilot's Display

The pilot's display is comprised of an ILS-type cross-hair analog display and a digital "distance-to-go" display. The operator may define any vertical plane and describe a desired flight path on that plane with a simple linear or second degree equation, using range location as the single variable. The cross-hairs provide the pilot with vertical and lateral displacement information with respect to that defined line. The "distance-to-go" readout displays the distance in meters of the airplane from any predesignated waypoint on the desired flight path. Scale, sensitivity and waypoint are designated by the operator and may be modified at any time.

The original intent of MAPS was primarily to provide positioning data for noise tests. Its portability and versatility have already shown that MAPS is capable of being used more extensively in the future. As experience and familiarity with the system increase, so will its usefulness.

Upper Atmosphere Weather System

The upper atmosphere weather system is mounted in a meteorological airplane (Cessna 172) and consists of several transducers, a data acquisition and control unit, a desktop computer, and a telemetry transmitter (Figure 4). A Rosemount temperature probe, Statham pressure transducers, and General Eastern dewpoint hygrometers are mounted on the airplane strut. Data acquisition and control of functions are accomplished with an HP-3497A and an HP85 desktop computer. Raw data and calibrated data (in engineering units) are permanently stored on digital cassettes. While the HP85 is capable of performing some calculations and providing an on-line plot of weather parameters versus altitude, its primary function is to prepare the data for telemetry transmission. Data samples are transmitted to the ground station approximately every five seconds. This provides a detailed survey of the temperature and relative humidity variation with altitude which can be used immediately by the ground station.

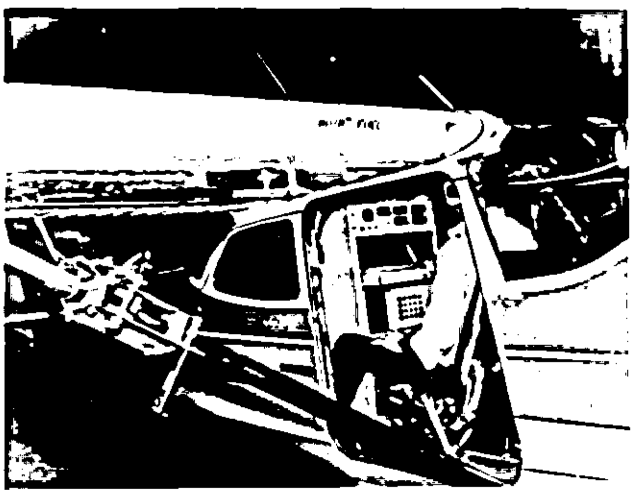

Figure 4 - Meteorological Airplane

Ground Station

ADP-4

Acoustic Data Processor 4 is the focal point for Boeing community noise testing. ADP-4 is a computer-based acoustic analysis system which records noise and weather data and receives telemetered upper atmosphere weather data from the meteorological airplane plus performance and positioning data from the test airplane.[3] All the above information is time-correlated and normalized to reference conditions then displayed and stored in permanent memory. ADP-4 is perhaps the most complex subsystem in the noise testing system and requires several operators.

Background

On-line acoustic data have been acquired in some form during all flyover testing conducted by Boeing in the past 30 years. Initially, hand-held sound-level meters were used for on-line visibility, then strip chart recordings of dBA or maximum-hold digital meters were used. Eventually a system was built around a Digital Equipment Corporation Model PDP-8 computer and a General Radio Model 1921 frequency analyzer to provide calculated noise numbers. In 1978, that system was replaced with an improved configuration using a PRIME Model 300 minicomputer and four analyzers. Incorporation of a minicomputer allowed noise number calculation and correction of the data for system response, test day conditions, and airplane performance. However, two major problems still existed with the PRIME 300-based system: (1) it was slow and (2) the corrections applied to the data were not as complete as those applied to the final data on a large mainframe computer. Weather parameters were being measured, but the data were not available to the computer except in limited summary form. Assumptions made from this summary introduced the potential for large errors. Target values were used for airplane performance and location information, and slight variations from the actual values added to the errors.

To overcome these limitations, ADP-4 was designed, with the major objectives being to provide the finest possible information with which to normalize the test data and to speed up the processing tasks. This objective differed from previous minicomputer-based certification system requirements in that more detailed and accurate information was desired along with greater speed.

System Attributes

To accomplish the desired goals, the system collects and analyzes acoustic data from up to four microphones simultaneously, either directly (on line) or from a tape recorder. The data are stored in a data base in engineering units by applying the necessary equipment corrections (such as microphone and analyzer frequency response and sensitivity). Analyzed acoustic data are presented as 1/3-octave band Sound Pressure Levels (SPL), Perceived Noise Levels (PNL), Tone Corrected Perceived Noise Levels (PNLT), and Effective Perceived Noise Levels (EPNL). Calculations had to be in accordance with FAR Part 36, Amendment 9. Both as-measured and normalized data are calculated and displayed.

Acoustic data are collected over the frequency range of 44 -11,200 Hz from key microphones for the duration of the aircraft flyover. Weather data, including temperature, relative humidity, wind speed, and wind direction at a height of ten meters above ground level, are monitored continuously and recorded for the time corresponding to the aircraft flyover. Temperature and relative humidity are sampled and recorded at altitudes from ten meters to the test aircraft flyover altitude on a regular basis. Airplane position, engine performance, and configuration information are sampled and recorded during each flyover.

To achieve a timely, accurate display of acoustic, weather, and aircraft and engine performance data, ADP-4 uses a radio telemetry system. The telemetry system eliminates the error-prone method of manual entry of performance and weather data and speeds data processing. In previous systems, manual inputs were often the source of delays in evaluating extrapolated acoustic data. By using the telemetry system, not only are manual input errors eliminated, but data acquisition and data base storage are accomplished automatically, providing an immediate opportunity to evaluate test progress.

These automatic features are used extensively. For example, upper air weather must be monitored prior to testing to determine when and how long an acceptable "weather window" (Figure 5) will be available. During testing, the weather is scrutinized to determine that each aircraft flyover is made under weather conditions that meet FAA requirements. Flight performance must be evaluated for each flyover to ensure that aircraft altitude, speed, position, and other requirements are met. Automatic acquisition, storage, and display of data through the use of telemetry enable a dramatic increase in speed and efficiency.

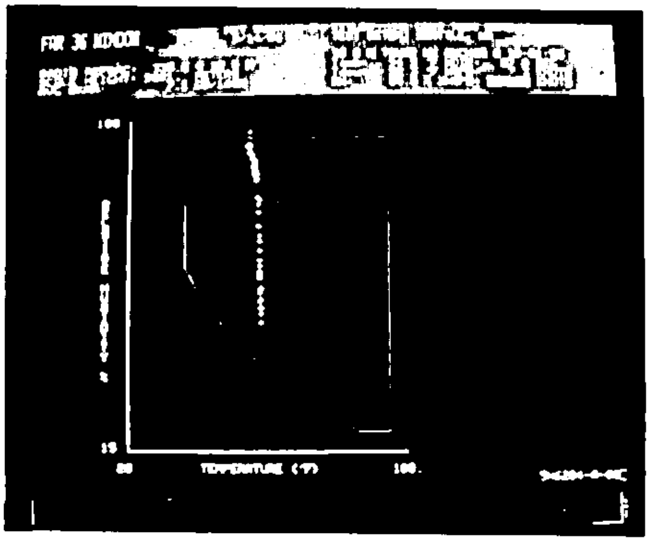

Figure 5 - Weather Window - Relative Humidity versus Temperature

Minimum operator intervention during testing is also important to ensure efficient data collection. Accordingly, the system was designed to accomplish data collection automatically as follows. Noise data are fed to a 16-channel multiplexer (MUX). (The user selects which four MUX lines are to be used.) The data are then read by microprocessor-controlled Bruel & Kjaer 2131 1/3-octave band analyzers. Data collection continues until the difference between a preselected band level and the maximum level exceeds some preset value. When this condition is met, the microprocessor sends an interrupt to the computer so that the data are transferred to the main memory, to be stored in digital format on magnetic disk. All of these operations are controlled through the on-line data acquisition program. The raw data, which consist of 1/3-octave band SPL's and associated test information (time of acquisition, analyzer number, MUX line, test location and description, and test number), are then analyzed using the acoustic analysis program.

Hardware Description

A PRIME 400 computer is used to accommodate the system described above. This virtual memory timesharing computer has 0.5 megabyte of error-correcting metal oxide semiconductor (MOS) memory and two kilobytes of cache memory to speed execution. An 80-megabyte disk is partitioned to provide a paging disk, program disk, and data storage disk. Additional disk cartridge drives provide raw data storage and an interchangeable medium with other systems for software data support. Backup storage and data transmittal are provided by an industry-standard nine-track digital magnetic tape drive. An asynchronous multi-line controller (AMLC) provides 16 RS232 communications lines for user terminals. A parallel interface drives the electrostatic printer-plotter via a copy controller that is also able to produce prints from a Tektronix graphics terminal. Two custom interfaces provide for direct memory access (DMA) to and from up to four Intel microcomputers each. As implemented, a total of six microcomputers is used. Each is erasable programmable read-only memory (EPROM) programmed to use the same program control protocol and data return buffer, and each has access to current test time code.

The entire system is housed in a 40-foot trailer that can be trucked conveniently. In addition to the minimum environmental considerations required for equipment functioning, certain operator environmental conditions are included for more efficient operation. A well-lighted, comfortable work area is provided for test programs, where the work atmosphere may vary from tedious to frenzied. The trailer is divided into two areas, an equipment area and a data evaluation area. The equipment area features a computer room type floor that allows convenient access to all cabling. An area behind the computer rack allows easy access for troubleshooting and maintenance. The data evaluation area is carpeted and has an office-type appearance. Paneling, fluorescent lights, storage cabinets, windows, and air-conditioning are provided in both areas. This arrangement has proved well-suited to keeping equipment functioning and personnel productive.

Test Operations

Simply stated, the object of community noise testing is to measure the noise produced by an airplane in a given configuration while it is flying along a specified path. Simple though it may seem, the amount of coordination required over a very brief period of time is surprising. Since the inception of noise testing to meet FAR Part 36 standards, this method of operating has remained basically the same, however, the 757 and 767 noise test programs were conducted more efficiently than previous programs because of the addition of MAPS, telemetry, and ADP-4.

A typical approach condition in an earlier program would proceed as follows:

The pilot flew level at a setup altitude maintaining the airspeed desired for the condition while all clocks on the ground and in the air were synchronized by a VHF time code generator. At a predetermined point, the

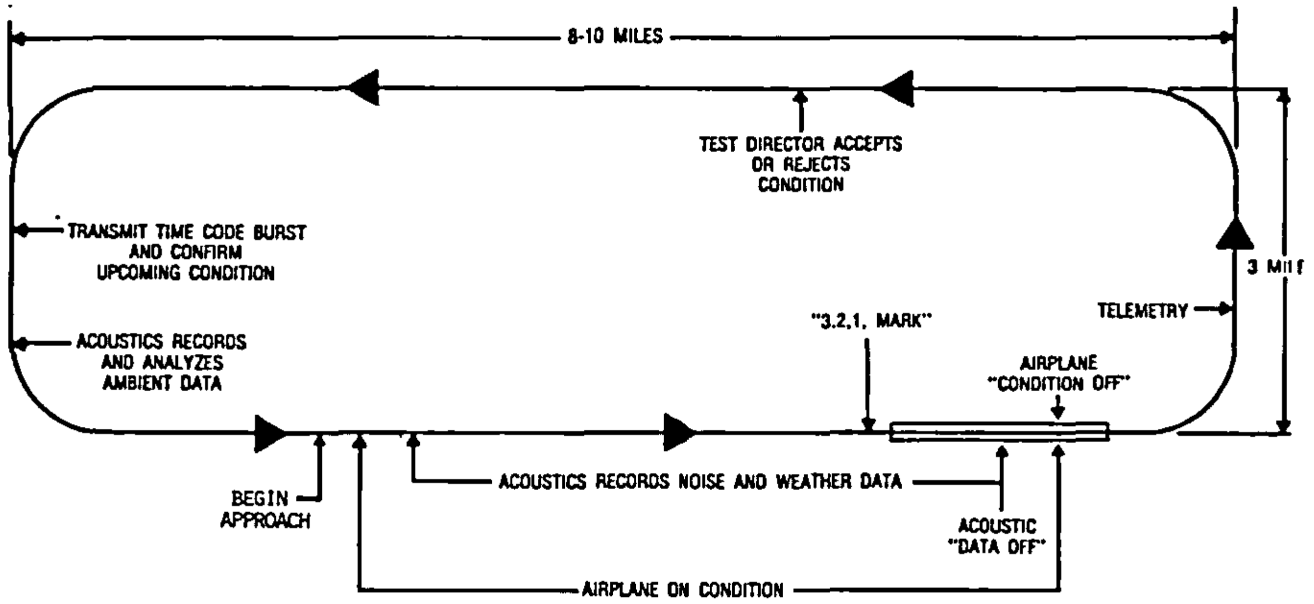

Figure 6 - Typical Noise Test Sequence

pilot would set the throttles at a specified N1 or EPR and begin approach. Since the airspeed, throttle setting and airplane configuration were fixed, the glide slope was a variable which could be estimated very closely, but the pilot had to begin descent when it "felt right" or when an observer at a fixed point on the ground called out. Either method entailed guesswork.

As the test airplane flew over the noise range primary location, an individual stationed there announced on VHF radio, "3...2...1...Mark" when the airplane was directly overhead, then estimated the number of wing-spans that the test airplane flew to the left or right of center. More guesswork. At the call "Mark" all test personnel noted the time and a hardcopy was made of the onboard data screen. The hardcopy was scanned for airplane performance acceptability (N1, airspeed and altitude), then relevant data were verbally radioed to the ground on VHF. The radioed data were manually entered into the mini-computer at the acoustic station, along with assumed airplane positioning data and a local weather summary. The acoustic data were displayed on a screen, and the only method of data storage was a hardcopy from that screen. Loss of the hardcopy required a complete re-analysis. Additionally, because of the work intensive methods of computer input, on-site data reduction generally lagged behind data acquisition.

Acoustic data could be fully normalized only with a large mainframe computer in Seattle, and accurate positioning data required at least 24 hours for the theodolite film to be returned to Boeing Field, developed, and analyzed. If a particular condition produced questionable acoustic data, a thorough analysis of the condition would sometimes take two to three days.

The new system eliminated most, if not all, of the problems inherent with the previous method of noise testing. The differences are outlined below.

Time coordination remains unchanged and the condition setup is the same, with the exception that the pilot watches the MAPS analog cross-hairs and begins approach when the vertical bar reaches center, which will occur at a known value of the "distance-to-go" digital display. The pilot then "flies to the cross-hairs," flying up and to the right when the cross-hairs are up and to the right, etc., varying airspeed within acceptance limits. For general coordination and as a backup, the "3...2...1...Mark" call is still performed, but the similarity ends there. At the "Mark" the pilot's digital display should pass through zero and the ADAMS II AC program is in the middle of determining overhead times for up to four range locations (including the "Mark" location) and interpolating up to 40 parameters to those times. After the last range point is passed, the AC program displays a condition summary of all 40 parameters at all four overhead locations. The summary is quickly reviewed for spurious data which are deleted, then the summary is telemetered to the ground, generally within two minutes of the condition end. The sequence of events is shown in Figure 6.

ADP-4 receives and immediately displays the telemetered summary (Figure 7), highlighting position or performance data indicating any unacceptable conditions. The meteorological information from the ten-meter tower will have been continuously monitored during the condition for unacceptable crosswind components, wind speed and gusts. The most recent upper atmosphere weather sounding information will have already been stored. This allows the site test director to make a quick condition acceptance decision based on complete and accurate information. A comprehensive condition summary, including weather, acoustic, and airplane performance data, is displayed, generally within four minutes of the condition. The information is also stored in permanent memory and is easily recalled.

In summation, the advantages of the new system are as follows:

- Positioning and performance condition summaries are more accurate.
- Telemetry allows more data to be transmitted with greater confidence.
- ADP-4 and telemetry allow a complete condition summary to be displayed at one location, thus centralizing test coordination.
- The extensive use of computers decreases human error and speeds up data acquisition.
- MAPS provides precise pilot guidance and real time positioning with final data-type accuracy.
- ADP-4 provides more reliable and complete acoustic and meteorological data, plus increased memory capacity which allows access to that data at any time.

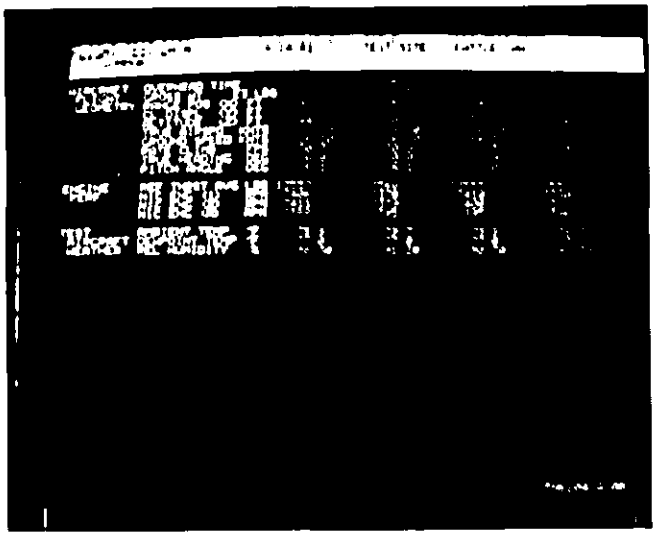

Figure 7 - Condition Summary

Conclusion

Overall performance of the system on the Boeing Models 757 and 767 noise certification tests was excellent. Minor software bugs and telemetry transmission errors were the only problems encountered, and they were corrected on-site with little or no test delay. Statistics showing some of the performance and timing aspects of the system are listed below:

- Acoustic Data Acquisition
  Total Conditions: 152
  Missed Conditions (hardware failure or operator error): 2
  Efficiency: 98.7%
- Acoustic Data Analysis
  On-Site EPNL versus "Final" EPNL Values:
  Normally less than 0.25 EPNdB difference
  Average difference = 0.1 EPNdB
- Non-Acoustic Data Acquisition
  - Airplane Performance
    Total Conditions: 152
    Total Received: 128*
    Efficiency: 84.2%
  - Weather
    Total Data Points: Over 10,000
    Missing Points (transmission errors): Less than 1%
  - Airplane Positioning (24 points sampled for on-site versus final data comparisons)
    Overhead Time within 0.1 Second
    Off-Center Distance within 1.3 Feet
    Altitude within 2.2 Feet
- On-Line Data Timing
  - Weather: Displayed almost immediately every five seconds
  - Performance: Displayed within 15 seconds of receiving telemetry (usually within two minutes of "data-off")
  - Noise
    As Measured: Displayed within three minutes after "data-off"
    Extrapolated: Displayed within five minutes after "data-off"

* Remainder were transmitted by voice over radio and manually entered.

The availability of such complete data during testing proved invaluable in the Models 757 and 767 tests. On occasion, the on-line data showed certain unexpected noise characteristics, and as a result, minor changes in test plans were made in order to more fully investigate these phenomena. Also, the extrapolated on-site data were very useful in showing whether or not certain flight criteria had any measurable effect on noise heard on the ground.

Output is "certification quality" data. The basic design philosphy has always included upward compatibility as a goal, so further enhancements may be added at a later date if a need arises. However, as presently configured, the system is capable of supporting most known airplane flyover noise test requirements in a very cost-effective manner.

Future plans for use of the system include noise research and noise certification of various Boeing airplanes as well as use of the system by other aircraft manufacturers on a contract basis.

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THE FLYOVER NOISE TEST MONITORING SYSTEM (FONTMS)

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ABSTRACT

The Flyover Noise Test Monitoring System (FONTMS) is a computer-based system used to display multiple data presentations in the field during the test. It is composed of three interactive subsystems which produce the following plots and tabulations: (1) noise spectra for one or two selected microphones, (2) A-weighted noise levels at all microphone sites displayed as a function of various independent parameters (e.g., engine power setting, aircraft altitude, engine power cutback point, and microphone site), and (3) surface and sound-path weather. FONTMS is used to verify data quality and display data trends, allowing the test director the visibility necessary to control the test.

A brief description is given of the test methods used in flyover noise testing, with emphasis placed on the reasons for detailed data monitoring and how the display data formats convey the required information. Manual techniques of processing the information for display which were used in the past are mentioned. Detailed descriptions are given of how the current computer technology was implemented in FONTMS to replace the inefficient manual methods. The benefits are presented in terms of visibility, speed, accuracy, flexibility, and cost.

INTRODUCTION

The flight development, design verification, and certification of new commercial jet transport airplanes include flyover noise testing to assure public acceptability of the aircraft. Conventional flyover noise testing by Douglas Aircraft Company for certification or development purposes involves flying the test aircraft over an array of ground-based microphones at a remote airfield,[1,2] and recording the noise data at the airfield in the Noise Test Center (NTC) trailer. The tape-recorded information is provided to data centers at Douglas, in Long Beach, California, for spectral history analysis to calculate the noise metric of effective perceived noise level (EPNL). For certification tests, the test procedures and limits are strictly controlled by Federal Aviation Regulation (FAR) Part 36.[3] In addition to the noise criteria, FAR 36 specifies the procedures to be followed or allows approved equivalents to demonstrate to the noise limits, the equipment to be used, the site terrain conditions, and the meteorological conditions required for the test. Whether the tests are conducted for certification or development purposes, FAR 36 requirements are followed to ensure the uniformity and quality of data.

The test matrix, which includes test conditions and microphone locations, is designed to obtain a data base of ground noise measurements covering the operating envelope of flyover types, heights, and engine power settings to satisfy the FAR 36 requirements.

Flyover noise testing requires extensive and complex techniques and equipment to control, record, and monitor the test procedures and data. Ground- and aircraft-based systems include a noise-sensing array dispersed over a surveyed ground test range; a noise recording, test communications, and monitoring facility; recording stations for the surface weather and sound-path weather; a tracking facility for the precise continuous recording of the aircraft space position; and aircraft instrumentation for recording time-synchronized aircraft and engine operating parameters (Figure 1). Since all tests are conducted at remote sites, all equipment must be mobile. The equipment is usually in a vehicle that is towed to the test site.

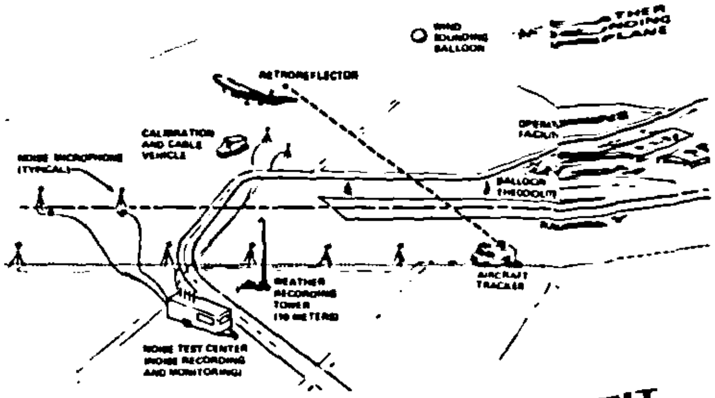


FIGURE 1. FIELD DEPLOYMENT OF FLYOVER NOISE MEASUREMENT EQUIPMENT

During testing, measurement parameters are monitored by the test director to assure the validity of data as well as to determine data characteristics and trends and effects of test variables.[4] The comprehensive data displays also permit sufficient visibility to assure adequate data sampling at minimum test costs. The monitoring tasks involve looking for invalid data, expected levels versus predictions, consistent trends (e.g., slopes and scatter), gaps in data, improper airplane test conditions, nonoptimized instrumentation (e.g., microphone locations), and equipment malfunctions.

In addition to supplying information to the test director, test data monitoring displays also provide information on test status and initial results to representatives of design engineering groups, program management, and the FAA.

TEST DATA MONITORING NEEDS

The specific displays will vary with the test, depending on the objective; however, the following types of information are usually required during the test in near-real time, regardless of the degree of automation.

COMPOSITE NOISE LEVELS

Composite noise levels are required to assess the noise status of the test aircraft relative to noise predictions, regulation limits, or noise data bases using a standard single-number value. Some noise level metrics used in flyover noise testing, in order of increasing complexity of calculations, are peak A-weighted sound pressure levels (L_A), perceived noise level (PNL), tone-corrected perceived noise level (PNLT), and EPNL. Ideally, for certification tests, the fully corrected and normalized EPNL would be calculated, allowing direct comparison to FAR 36 noise limits. However, this calculation requires immediate access to fully corrected and validated normalizing information, such as posttest calibrations, actual and reference aircraft performance, and weather data, as well as a large-scale computer, none of which are available in the field test environment. Therefore, only a preliminary EPNL can be calculated in real time. Presently, a less complex metric, PNL, is used with approximate normalization for only one or two microphone locations. However, the algorithms for EPNL are currently being incorporated in the calculations.

NOISE ONE-THIRD-OCTAVE-BAND SPECTRAL ANALYSES

Information on the broadband frequency and tonal distribution of the aircraft noise is frequently required to assess the components of the aircraft engine noise that control the composite noise level. This information is obtained from one-third-octave-band spectrum analyzer plots.

FUNCTIONAL NOISE TREND PLOTS

The data acquired during a flyover noise test forms a data base from which the certification noise levels are calculated. Information is therefore needed during the test to ensure that the data base is complete and unambiguous. Hence, a multiple cross-plot is maintained during the test to display composite (A-weighted) noise values as a function of the independent test variables such as engine power, aircraft altitude over the microphone, and cutback distance to indicate the following: (1) the range of the variables tested, (2) the continuity of the data in terms of the variables (e.g., no gaps in the data), (3) the data variation for repeated test conditions, and (4) the trends of noise level versus the independent variables. Typically, the A-weighted noise levels for 11 microphones must be plotted selectively against the parameters.

AIRCRAFT PERFORMANCE TABULATIONS

Aircraft performance data are continuously compiled at various times during a flyover to monitor actual versus planned test conditions in order to assess the acceptability of each run. Parameters logged include altitude, engine powers, airspeeds, gross weight, temperatures, and the flight surface configuration. Examples of unacceptable conditions are: setting power late, missing target power setting, power setting differences between engines, airspeed variations, missing target heights, and drifting off the runway centerline.

WEATHER CONDITIONS

Surface wind speed and wind direction must be monitored in near-real time, along with atmospheric noise absorption, temperature, and humidity measured over the sound path (ground level to airplane height). These data are needed to verify that the weather conditions are within FAR 36 limits so that the test can be started, and that they remain within limits during the test.

BACKGROUND OF FONTMS

The sources of the data to be monitored must be considered in the design of the acquisition and monitoring systems. Typical tests at remote sites use complex instrumentation systems deployed in two airborne and multiple ground-based sensor systems.[1,2,5,6,7] The sources and methods used before FONTMS was developed to transmit data to the NTC for recording and monitoring are shown in Figure 2. As indicated in the figure, initial methods included direct measurement, land line

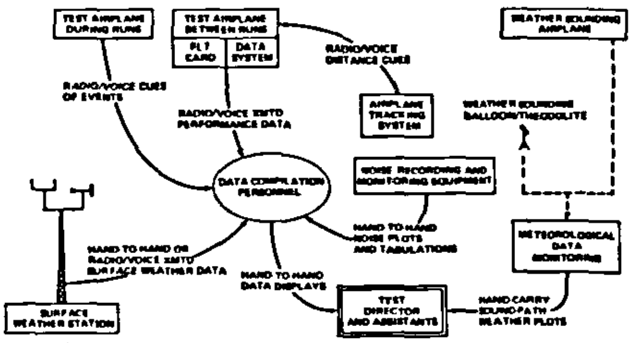


FIGURE 2. MEASURED DATA SOURCES FOR ON-SITE MONITORING

transmission, and voice/radio communication of aircraft and sound-path weather data for tabulation by hand.

Many parameters relating to the test systems and test environment affect the measured noise levels as shown in Figure 3. To increase the usefulness of the data displays, these effects are removed to the first order, as practical. Before displaying the information, the raw data received in the NTC must, of course, be converted to proper units and corrected for system response deviations using calibrations. Also, the noise data must be normalized to reduce scatter caused by variations of controlled parameters, such as altitude, lateral flight deviations, and engine power setting, and uncontrolled parameters such as weather and ambient noise.

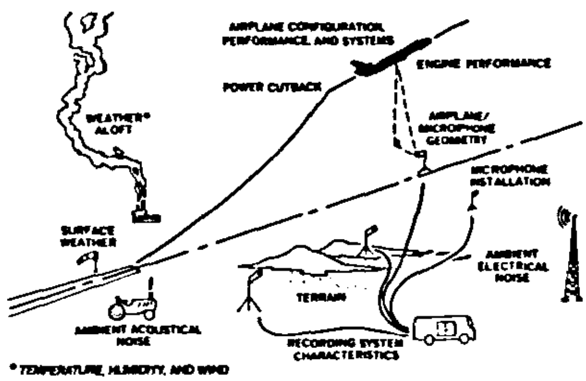


FIGURE 3. FACTORS AFFECTING MEASUREMENTS OF FLYOVER NOISE

Before FONTMS was developed, on-site data processing was done by manual tabulation of voice-radio transmitted cockpit readings, manual strip-chart noise readings, calculating and hand-tabulating the noise data, and final hand plotting of the relationships — all occurring in the NTC environment at the same time the radio communications and data acquisition activities were being conducted. The information was often generated by one monitoring or recording engineer and was presented orally or hand-carried to another. This produced a noisy and sometimes confusing environment. Also, during manual reading and plotting, the data displays were not available for evaluation by the test director and others. With additional test constraints and multiple corrections added over the years, the workload grew to involve as many as seven persons. And with the additional test criteria and sophistication, the correction, conversion, and display of this information became a more complex operation that interfered with the task of evaluating the data.

Additionally, due to the limited resolution and correction capabilities of the method, repeat runs were required to ensure that sufficient valid data were recorded. This could make for longer test schedules and higher program costs because of the high costs of aircraft flight tests.

Recent refinements in desktop computer technology, engineering instrumentation interfacing, and telemetry transmission yielded the solution. FONTMS grew from a calculator-based system to its current configuration over the past eight years. Most of the monitoring functions are now automated, allowing the test personnel to concentrate on the task at hand — evaluating the test results and assessing the success of the test.

DESCRIPTION OF FONTMS

FONTMS is based upon two Hewlett Packard HP9845T computer systems with extensive digital interfaces to data sources such as a noise spectrum analyzer, two systems for airplane telemetry data acquisition, and a microprocessor-based noise and weather processing system, as well as supporting peripherals such as disk drives, a four-pen plotter, and an interfaced desktop calculator.

The system is composed of three subsystems which are physically separated at three work stations, each having its own operator (Figure 4). The design was predicated on making each station as autonomous as possible to reduce extraneous verbal communication, while sharing certain information through the digital bus. Each operator was to be free to operate his system at his own pace, being restricted only by the speed at which data were transmitted from the source, which, in turn, is limited by the sequencing and rate of the flyovers. Also, the test witnesses were to have good visibility of the processed data.

The current configuration includes these subsystems:

1. Noise spectral analysis system (NSAS) — For noise spectral plots, spectral tabulations, and PNL tabulations.
2. Noise level multiplot (NLMP) system — For noise trend plots and performance tabulations.
3. Airborne meteorological telemetry system (AMETS) — For plots and tabulations of sound-path weather conditions.

A simplified description of each system in terms of hardware, user operation, and example data outputs is given in the following paragraphs.

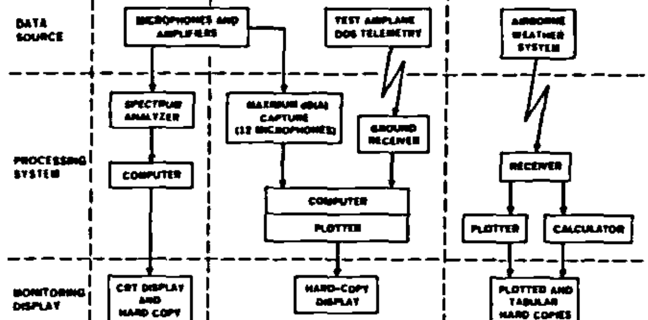


FIGURE 4. BASIC FONTMS SUBSYSTEMS AND DATA FLOW

NSAS

Hardware

The hardware for NSAS is shown in Figures 5 and 6. NSAS is built around the Bruel & Kjaer (B&K) Model 2131 one-third-octave-band spectrum analyzer, with the Hewlett Packard model 9845T computer as controller, calculator, display, tape storage unit, and printer. During the test, noise data from the signal-conditioning amplifiers are input into the B&K spectrum analyzer, filtered into one-third-octave bands, and detected using 2-second exponential averaging. The resulting spectra are transmitted via the digital bus to the computer at 0.5-second intervals over the significant period of the flyover, which lasts approximately 20 seconds. In order to adjust the data to absolute sound pressure levels (SPL), the gain information is also input to the computer from the microprocessor. Using the previously stored SPL calibrations, the measured spectral history is calculated together with a PNL history. The spectrum at the time of PNL maximum, PNLM (i.e., the noisiest time during the flyover), is then corrected for deviations from reference atmospheric absorption conditions, using ARP 866A methods,[8] and from a reference height over the microphone. After recalculation of the spectrum and PNLM values, the data are available for plotting, tabulating — on the CRT or hard copy via thermal printer/plotter — and storing on a disk.

FIGURE 5. NOISE SPECTRAL ANALYSIS SYSTEM (NSAS)

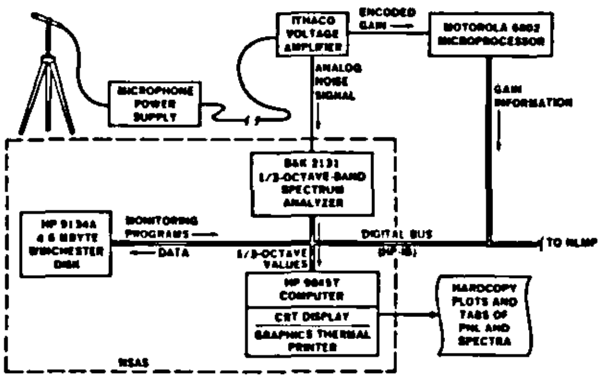


FIGURE 6. SIGNAL AND DATA TRANSMISSION OF NOISE SPECTRAL ANALYSIS SYSTEM (NSAS)

Operation

The system operates primarily through computer keyboard dialog, with some manual operation of the analyzer needed. A simplified flowchart of the BASIC computer program, FrA, used on NSAS is shown in Figure 7. Certain data (e.g., flight, test point, and microphone) are input by the operator through a series of answers to computer questions. Calibrations of sound level amplitude and system frequency response for four microphones can be held in computer memory and additional calibrations stored in disk files.

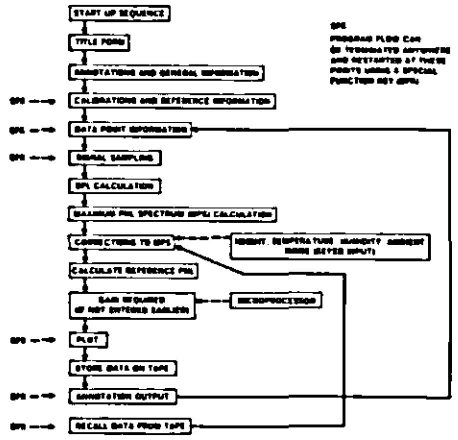


FIGURE 7. NSAS PROGRAM FrA BASIC FLOWCHART

Output

Figure 8 is an example of the hard-copy output of the CRT plot and tabulation of several PNLM spectra made during a recent test. Four line-codes are currently available to identify multiple spectra on a single plot.

NLMP

Hardware

In the NLMP system shown in Figures 9 and 10, noise signals from up to 12 microphones are filtered by an A-weighting filter, converted to SPL type units with a

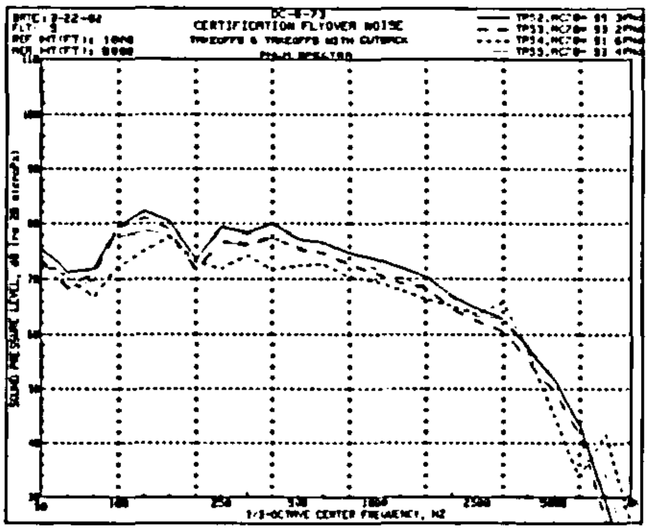


FIGURE 8. SPECTRAL PLOT PRODUCED BY NSAS COMPUTER SYSTEM

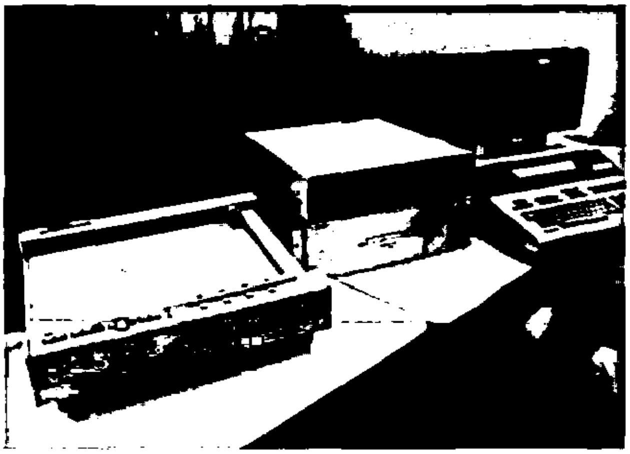


FIGURE 9. NOISE LEVEL MULTIPLOT (NLMP) SYSTEM

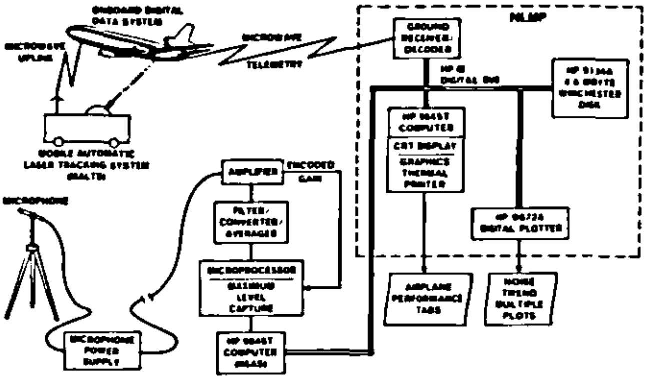


FIGURE 10. SIGNAL AND DATA TRANSMISSION OF NOISE LEVEL MULTIPLOT (NLMP) SYSTEM

logarithmic converter, and averaged to simulate a standard "slow" sound-level meter response. The data are next processed by the peak level monitor (PLM), based on a Motorola 6809 microprocessor system, which converts the fluctuating dc signal to digital values and selects and stores the maximum value during the flyover. The maximum A-weighted values, one for each microphone, are then furnished to the digital bus, where they are transferred to the NLMP computer via the NSAS computer.

Since the NSAS computer controls the common digital bus, the interrupt feature was employed to allow the microprocessor to provide A-weighted SPL data to the NLMP non-controlling computer. The interrupt feature allows the controlling NSAS computer to process these data as a background task without interfering with the spectral processing.

The airplane position in terms of X, Y, and Z coordinates is transmitted via microwave telemetry to the test aircraft from the mobile automated laser tracking system (MALTS).[5] This information is added to the approximately 400 parameters, including altitude, airspeed, engine settings, and temperatures, processed by the aircraft digital data system. The data are then transmitted via telemetry to a ground receiving station in the NTC. After conversion, the data are read by the NLMP system computer from a decoder/buffer.

After some adjustment of the A-weighted values to reference conditions and other manipulation, the data are plotted on a prepared form tailored to the specific test.

Operation

As with NSAS, the NLMP system is also operated primarily from the computer keyboard except for such activities as loading paper or changing pen numbers. The computer program used is also written in the BASIC language and is called "MPTM." Figure 11 is a simplified flowchart of program operation. The versatility of the program allows the operator to select from a multitude of possible displays, depending upon the particular test. Examples of program options available are shown in Table 1.

Output

Figure 12 is an example of a multiple plot from a recent test. The figure shows the four plot areas which are being updated and displayed during the test. Figure 13 is an example of a real-time tabular display of airplane

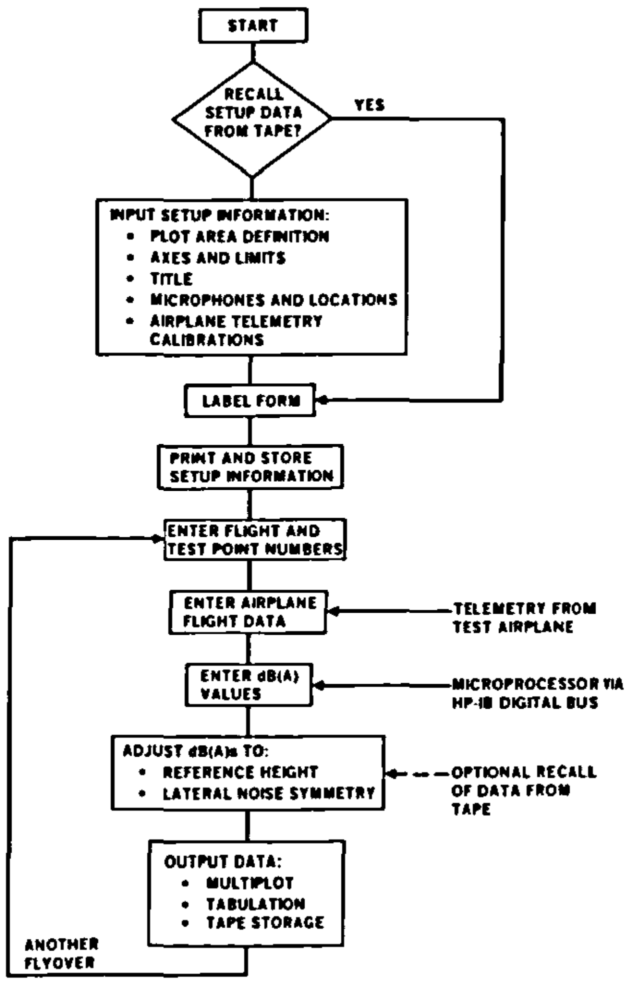


FIGURE 11. NOISE LEVEL MULTIPLOT (NLMP) PROGRAM MPTM FLOWCHART

TABLE 1. SOME NLMP PROGRAM CAPABILITIES AND OPTIONS

1 TO 12 MICROPHONES (STORAGE OF MICROPHONE NAMES AND LOCATIONS)
1 TO 4 PLOT AREAS (INDEPENDENT TITLING AND SCALING OF ALL AXES)
1 TO 4 ENGINES WITH N_1 (PERCENT OR RPM) OR EPR POWER-SETTING PARAMETERS
MICROPHONE AVERAGING OPTION
CORRECTED ENGINE FAN SPEED (FOR TEMPERATURE)
HEIGHT AND SYMMETRY ADJUSTMENTS OF NOISE VALUES
OPTIONS FOR EACH OF FOUR PLOT AREAS:

1. NORMALIZED CENTERLINE dB(A) VERSUS ENGINE SETTING (N_1 OR EPR)
2. AVERAGE (ADJACENT FOR SYMMETRY) MAXIMUM SIDELINE dB(A) VERSUS ENGINE SETTING (N_1 OR EPR)
3. SIDELINE dB(A) VERSUS AIRPLANE ADJACENT HEIGHT (ADJACENT FOR SYMMETRY)
4. NORMALIZED CENTERLINE dB(A) VERSUS CUTBACK DISTANCE
5. MULTIPLE CENTERLINE dB(A)s (HEIGHT CORR) VERSUS ENGINE SETTING (NOISE-POWER-DISTANCE)
6. FLIGHT PROFILE
7. ALTITUDES VERSUS N_1/√ THETA (NPD PLOT COVERAGE)

INDEPENDENT FLIGHT AND REFERENCE AIRPLANE PROFILE OPTIONS:

1. TWO HEIGHTS AT TWO LONGITUDINAL LOCATIONS (STRAIGHT LINE)
2. A HEIGHT AT A RUNWAY LOCATION, PLUS A GRADIENT (STRAIGHT LINE)
3. THREE HEIGHTS AT THREE LONGITUDINAL LOCATIONS (2-SEGMENT PROFILE)
4. TWO HEIGHTS AT TWO LONGITUDINAL LOCATIONS PLUS A GRADIENT (2-SEGMENT PROFILE)

performance data which is used to assess the validity of each flyover in terms of satisfying target conditions, and to supply data for the multiple plot.

AMETS

Hardware

Weather conditions are measured by sensors mounted on the exterior of a light aircraft.[7] Typical measurements include total air temperature, dew point temperature, barometric altitude, and turbulence. The measured data are telemetered to the receiver in the NTC. After conversion, the data are available in digital form on the digital bus and in analog form for output on the strip-chart recorder.

The calculator reads the data at 50-foot intervals, applies calibrations, makes adjustments (e.g., ram air effect on static air temperature), and calculates humidity values from temperature and dew point temperature information. The data are output on a printer in tabular form during the measurement and stored on a digital tape cassette for later retrieval.

Via another digital interface, the data can be transferred to a computer for final sound-path weather calculations and output.

The hardware in AMETS is shown in Figures 14 and 15.

Operation

During the test, sound-path weather is measured over the test site at approximately one-half-hour intervals. The AMETS aircraft makes a sounding maneuver consisting of a slow climbing spiral. As the aircraft ascends, the ground operators monitor the continuous display of raw data on the strip-chart recorder (Figure 16) and the tabular output, including calculated humidity, on the calculator's printout. The operation of the calculator and computer systems is indicated in the combined flowchart in Figure 17.

Output

Figure 18 is an example of a finalized tabulation and plot of measured and computed parameters of sound-path weather. These data are used in documenting acceptable test weather conditions per FAR 36, as well as in correcting the noise data for atmospheric absorption effects in the final processing.

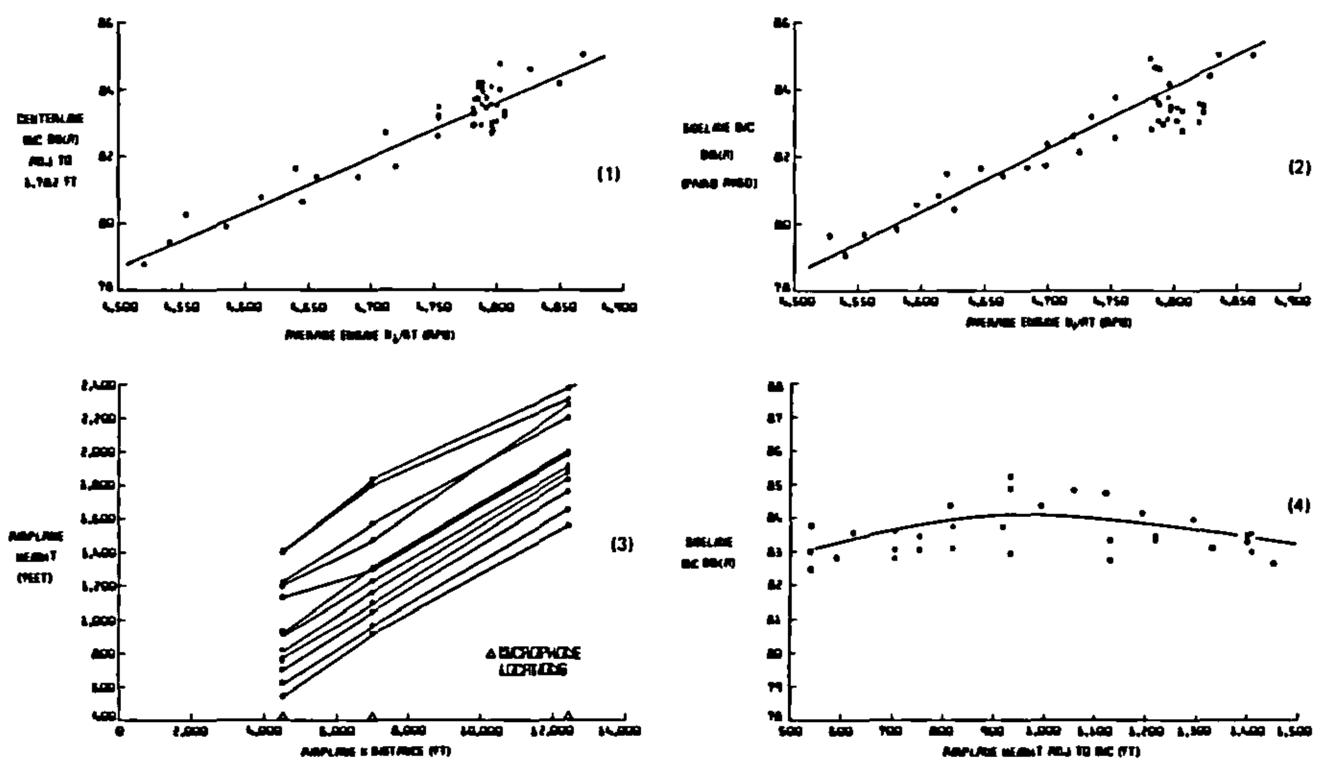


FIGURE 12. COMPUTER PLOT FROM NOISE LEVEL MULTIPLOT (NLMP) SYSTEM

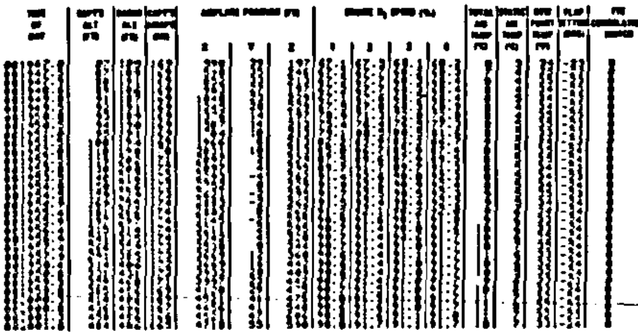


FIGURE 13. NLMP AIRPLANE PERFORMANCE OUTPUT

FIGURE 14. AIRBORNE METEOROLOGICAL TELEMETRY SYSTEM (AMETS)

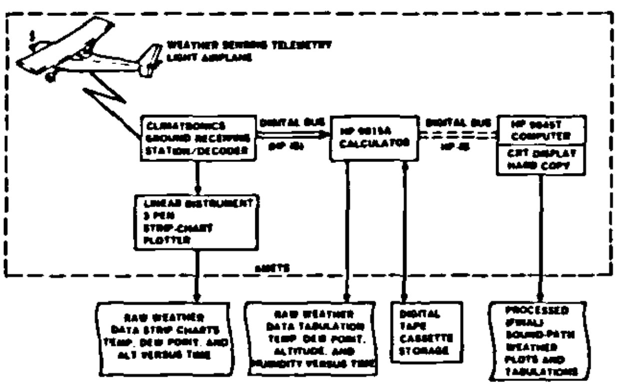


FIGURE 15. SIGNAL AND DATA TRANSMISSION OF AIRBORNE METEOROLOGICAL TELEMETRY SYSTEM (AMETS)

EVALUATION TO DATE

FONTMS has been used most recently for a certification flyover noise test as well as for several research tests in the last few years while it was being developed. Most of the expected benefits of the system have already been realized. Test costs have been lowered by eliminating approximately four data processing operators and by increasing real-time visibility and accuracy of important test validation parameters so that extra "insurance" flyovers are no longer necessary. Due to the independent operation of each work station and the availability of data on the digital bus, verbal communication and hand-carrying of tabulations and plots solely for transfer

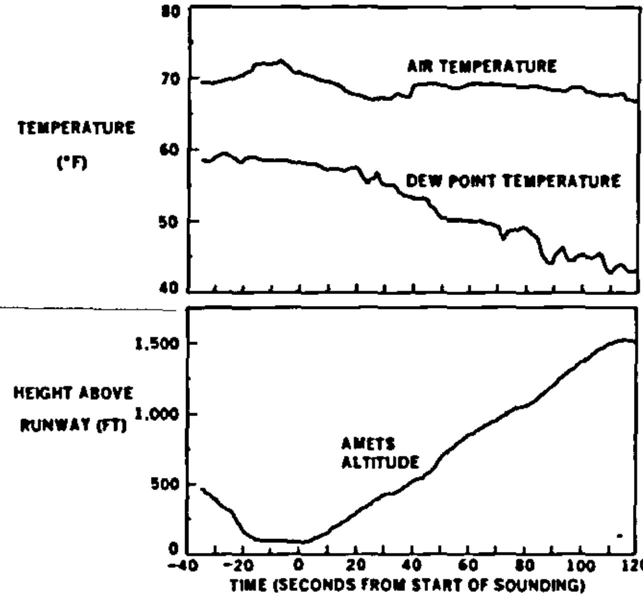
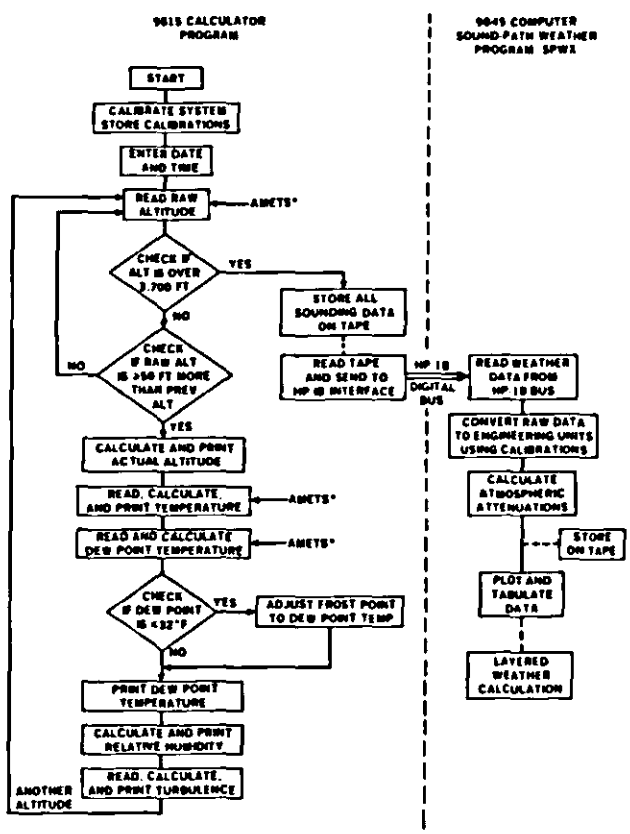


FIGURE 16. AMETS REAL-TIME OUTPUT DISPLAYS

of data have been eliminated. Some verbal radio communications of timing information and run assessment are still required, of course. However, the result is a more professional atmosphere in which data evaluation rather than data transfer is discussed.

The displays of data are now available for evaluation by the test director and test witnesses virtually 100 percent



* GROUND RECEIVING STATION

FIGURE 17. AMETS SOFTWARE FLOWCHART

SOUND-PATH WEATHER for CROWS LANDING, CALIF.

DATE=3/19/82 TIME=06:54 PST

ALTITUDE (FT,AGL)	AMB TEMP (DEGF)	REL HUMIDITY (%)	8-KHZ ALPHA (DB 100 ft)
30	36.9	90.0	7.7
65	38.4	89.0	8.2
270	43.0	69.9	8.9
424	45.3	64.5	9.3
644	45.9	67.7	8.7
939	44.7	68.6	8.9
1,300	43.5	62.9	9.9
1,510	42.6	63.3	10.1
2,042	40.5	64.0	10.4
2,507	38.1	67.6	10.5
3,000	36.3	65.7	11.1

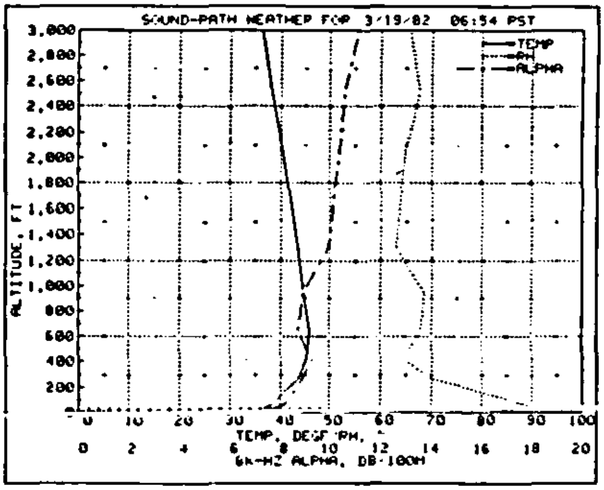


FIGURE 18. AMETS FINAL DATA

of the time. No displays are out-of-view as a result of hand processing of tabulations and plots since all is done by machine with the viewer present.

Due to the simplicity of BASIC language programming, the programs are easy to modify in the field, and special calculations, adjustments, and formats can be easily prepared for each test. Routines added so far calculate the difference between two spectra and the average of multiple spectra, and plot the results for each.

The increased calculation speed and accuracy reduce the operator's workload in manipulating the data, allowing more time for assessing the validity of the data.

It should be noted that despite the benefits of FONTMS hardware, there are some concerns. Provisions must be made for a backup mode, or Plan B, if some of the hardware fails during the test. The backup plan generally calls for reverting to the manual method used before FONTMS was developed. History indicates that certain items such as the digital bus, printer, and other interfaces can be very sensitive to ac power line voltage and ambient temperature which can fluctuate due to conditions at a remote site. It is expected that as experience improves, limitations will be better understood and methods of avoiding these problems will be found.

As a result of the good experience with the system in flyover noise testing, plans have been made to use a variation of FONTMS for monitoring displays of steady-state noise data and narrow-band noise and vibration data on the test aircraft.

CONCLUDING REMARKS

The current technology in small computers was utilized to develop FONTMS. The result is a system that processes and displays flyover noise and associated data during the test with improved speed, accuracy, and visibility, as well as reducing the crew workload and crew size. The system has been used successfully in recent tests, yielding valuable on-site information, and is likely to be useful for several more years.

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FIGHTER AIRCRAFT DYNAMIC PERFORMANCE

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Abstract

Flight testing in the area of performance has been considered for years to be straightforward. However, with the advent of ultra-performance fighter aircraft, performance flight test has had to deal with dynamic and diverse environments. In addition to the usual how far and how fast, there are questions to be answered in the areas of high response turning performance and expanded lift and drag regions. The advent of sophisticated high resolution, high response INS systems, has brought into its own the area of dynamic performance testing. Dynamic performance incorporates new maneuvers which not only give excellent results, but also can cut the number of performance test points required. Wind-up turns (WUTs), roller-coasters (RCMs) and split-s' (SS) are maneuvers considered as dynamic performance. These dynamic maneuvers were used to compute lift and drag with very good results. The quality of the data and the correlation with the conventional maneuvers (cruise, accelerations, climbs, and turns) was excellent. Obtaining this data from the sophisticated INS system required innovative analysis methods. Alone, the INS will provide excellent three-dimensional velocities, accelerations and rates. However, it will not provide the required wind axis flight path accelerations and angle of attack for dynamic performance testing. This paper addresses the methods used to obtain performance parameters from the INS as well as a break down of how dynamic performance testing can greatly reduce flight test time and effort.

Introduction

This paper describes dynamic performance data acquired during an AFFTC flight test program. These data were obtained from RCM, SS, and WUT maneuvers. For simplicity and brevity, only data at one Mach number are presented. Specific details on aircraft type and flight conditions are deleted in order to avoid classification problems. The data presented in this paper does not by any means represent the first application of INS data to performance testing. An early limited application at the AFFTC was in the late 60s on the SR-71 project. On that project the INS accelerations were used only during cruise testing. The general concept of using an INS was studied at the AFFTC in the late 60s and early 70s, but no funds were available for any testing. The first major application of an INS was on the YF-16 project in 1974. Mr. James Olhausen of General Dynamics, Fort Worth was primarily responsible for the success of that effort. The INS data was used also on the STOL transport projects in 1975. What the authors feel the data in this paper represents is the highest quality dynamic performance data obtained to date. Also, this paper represents the first unclassified documentation of the wind calculation algorithms.

Flight Maneuvers

The first maneuver presented is the RCM. The objective was to have a sinusoidal variation of load factor at a rate of 0.5 gs per second. Figure 1 is a time history of a RCM. The RCM maneuver extends over a g range of zero to two gs but other g ranges are possible. The problem with larger load factor ranges is the Mach number excursions accompanying the large g changes. These excursions can be very large. This is especially true in a pull-up maneuver. At high g levels in a pull-up the aircraft will decelerate substantially. However, this is not the case for the split-s maneuver. Therefore, the SS maneuver was devised to obtain aerodynamic data at high g levels while minimizing the Mach excursions. A time history of the SS is given in Figure 2. The SS is performed by rolling inverted then increasing the load factor at about one g per second until either the limit g or limit angle-of-attack is reached. As can be seen, there is very little Mach number loss. This is due to a combination of two factors. First, the maneuver is highly dynamic (one g per second) and the entire maneuver takes approximately 5 seconds, which typically is not enough time for large Mach excursions. Secondly, the maneuver is inverted and a trade-off is made between airspeed and altitude. The third dynamic maneuver is the WUT. This is a turn maneuver where g is increased at a rate of one g per second until limit g or limit angle-of-attack is reached. The WUT is primarily a flying qualities maneuver. However, the superior quality of the data from the INS has allowed us to apply WUT maneuvers to performance lift and drag curves. The range of data in a WUT is identical to the SS and for the first time the results were also comparable to the SS. The WUT time history is shown in Figure 3.

INS Data Reduction

The INS used on this project was a system containing two gyroscopes and three accelerometers connected to the aircraft by a gimballed platform. The three accelerometers are located at 90 degrees to each other on a platform stabilized by the two gyroscopes. These accelerometers sense the aircraft's velocity changes in the north, east, and down (or x, y and z) directions. The two gyro spin axes are also perpendicular to each other and free-floating. The vertical gyro's spin axis is in the vertical position sensing roll and pitch motion. The azimuth gyro's spin axis is in the horizontal position and senses yaw motion. These accelerometers and gyros allow the INS to measure any accelerations or angular rate changes and output these parameters at 50 samples per second. Data analysis for performance parameters utilizing this INS involved only nine parameters. The first three are the x, y, z components of inertial velocity. The second three are the x, y, z components of inertial acceleration. The inertial acceleration components were computed by differentiating the inertial velocities. The last three parameters are the Euler angles (roll, pitch and heading). These nine parameters from the INS are combined with true airspeed to compute angle of attack and flight path accelerations. In order to make the required axis transformation it is necessary to compute winds. During wings level constant altitude flight the INS computer will produce the horizontal components of winds. But these winds are computed using true airspeed which has not been corrected for pitot static position errors determined during the flight test program. Also, the winds are only valid for zero bank and zero vertical speed. It was necessary to develop computer algorithms to compute winds for any maneuver. The details of these equations are contained in the Appendix.

There are three basic transformation equations from the earth axis system to the flight path or wind axis system. Unfortunately, there are five unknowns for these three equations. The five unknowns are three components of wind, angle-of-attack and sideslip. To solve these equations it was assumed that the vertical component of wind and sideslip were zero. This algorithm worked quite well for maneuvers where bank angle was small. This includes climbs, accelerations, cruise, RCM and SS. But for turns and WUT maneuvers, the terms in the equation (equation three in the Appendix) which were set to zero in order to solve the equation became significant for high bank angles. The data became unusable and in many cases the equations were unsolvable.

A new algorithm was developed to reduce turn and WUT data. The assumption was made that the horizontal components of wind remained constant during the maneuver. This is a good assumption for constant altitude turns. For WUT data where altitude is lost in order to minimize Mach number variations, the winds can vary somewhat but if the altitude loss is small (less than 1,000 feet) then the constant wind assumption is approximately correct. The vertical component of wind was again assumed to be zero. A least squares solution for winds was found using the basic equation for true airspeed (equation 12 of the Appendix). This method worked quite well for the turning data but could not be applied to data at zero bank angle. The reason for this is that to acquire a statistically valid set of wind components requires that the x and y components of velocity have some significant relative variation. During maneuvers such as accels and cruise the x and y velocities are nearly linearly dependent. That is, the x velocity is approximately a linear function of the y velocity. In order to solve for x and y wind components it is required that the x and y velocities be linearly independent. This is not true for non-turning maneuvers. Therefore, we used two methods for determining winds. The first method incorporated the zero bank analysis for cruise, climb, accelerations, RCM, and split-s maneuvers. The second method used the least squares method discussed above.

Aerodynamic Data

Lift and drag coefficients were computed and standardized to a common set of reference conditions. The reference conditions were a standard altitude and center of gravity. The equations used will not be represented here since they are standard aerodynamic equations and details of the corrections to reference conditions have no relationship to the purpose of this paper. The corrections were very small since most of the data presented herein was flown near the reference conditions. All of the data presented were flown at or near the same Mach number. This Mach number was well below the transonic drag rise so no Mach corrections were made. Only one Mach number is shown since this is all that is necessary to illustrate the data quality and data correlation.

A substantial amount of conventional maneuver data was collected during the tests. Figures 4 and 5 present the drag polar data for accelerations, climbs, cruise and turns. Figures 6, 7 and 8 present the drag polar data for single RCM, SS and WUT maneuvers, respectively. In order to facilitate comparison of the various maneuvers the same fairing is shown on each plot. The data is also presented with the same scales. As can be seen there is excellent agreement between all the maneuvers. A similar set of plots for angle of attack is presented in Figures 9 through 13.

Conclusion

Every paper on dynamic performance testing usually comes to the same conclusion. That is, that the data contained in the paper demonstrates conclusively that dynamic performance techniques can save tremendous amounts of flight test time. We could easily reach the same conclusion. However, there is more to a performance evaluation than obtaining the lift and drag data. It is still necessary to evaluate the thrust and fuel flow characteristics of the engine. For this it is necessary to do climbs, accelerations and cruise testing. Still, by incorporating dynamic performance techniques into a performance evaluation we can vastly improve the ability to define the drag polar and lift curves. We can cover the entire lift coefficient range of the aircraft at one Mach number with just two maneuvers (RCM and SS). Dynamic performance techniques can reduce flight time significantly, but the main advantage of the method is the improvement in the overall performance definition which it affords.

It is now up to a very conservative flight test community to reconsider the methods of obtaining lift and drag. This not only means a re-evaluation of the maneuvers involved, but also the type of instrumentation used. The INS system we were involved with has proven to be an excellent navigation device and an invaluable tool for determining basic performance parameters with a high level of accuracy.

Appendix

Flight Path Calculations From Inertial Platform Measurements and True Airspeed

A calculation of wind velocities was necessary in order to translate inertial accelerations in the earth-axis-system (north-east-down) to accelerations in the wind-axis-system. Two different wind calculation routines were used, depending upon the type of maneuver.

The first wind method was designed for maneuvers where bank angle was near either 0 degrees or 180 degrees. The assumptions made were that sideslip (β) and vertical wind (V_{W_Z}) were zero.

The exact equations for airspeed components in the earth axis system are as follows:

$$V_{A_X} = (\cos\psi \cos\theta \cos\alpha \cos\beta + \cos\psi \sin\theta \sin\phi \sin\beta + \cos\psi \sin\theta \cos\phi \sin\alpha \cos\beta - \sin\psi \cos\phi \sin\beta + \sin\psi \sin\phi \sin\alpha \cos\beta)\, V_T \quad (1)$$

$$V_{A_Y} = (\sin\psi \cos\theta \cos\alpha \cos\beta + \sin\psi \sin\theta \sin\phi \sin\beta + \sin\psi \sin\theta \cos\phi \sin\alpha \cos\beta + \cos\psi \cos\phi \sin\beta - \cos\psi \sin\phi \sin\alpha \cos\beta)\, V_T \quad (2)$$

$$V_{A_Z} = (-\sin\theta \cos\alpha \cos\beta + \cos\theta \sin\phi \sin\beta + \cos\theta \cos\phi \sin\alpha \cos\beta)\, V_T \quad (3)$$

where

$$V_{A_X} = V_{I_X} + V_{W_X} \quad (4)$$

$$V_{A_Y} = V_{I_Y} + V_{W_Y} \quad (5)$$

$$V_{A_Z} = V_{I_Z} + V_{W_Z} \quad (6)$$

V_{A_X} = X component of earth axis airspeed

V_{A_Y} = Y component of earth axis airspeed

V_{A_Z} = Z component of earth axis airspeed

V_{I_X} = X component of earth axis inertial speed

V_{I_Y} = Y component of earth axis inertial speed

V_{I_Z} = Z component of earth axis inertial speed

V_{W_X} = X component of windspeed

V_{W_Y} = Y component of windspeed

V_{W_Z} = Z component of windspeed

V_T = true airspeed

ϕ = bank angle

θ = pitch angle

ψ = heading angle

α = angle of attack

β = sideslip angle

By incorporating the assumption of $\beta = V_{W_Z} = 0$ and combining equations (3) and (6) yield the following:

$$V_{A_Z} = V_{I_Z} = (-\sin\theta \cos\alpha + \cos\theta \cos\phi \sin\alpha)\, V_T \quad (7)$$

Equation (7) is readily solved for α using a one dimensional Newton-Raphson iteration scheme. Once α was calculated the wind components were computed using equations (4), (5) and (6) by substituting into (1), (2) and (3). The set of equations (1) through (6) reduced to three equations with five unknowns. The unknowns were the three components of wind and α and β. In order to solve these equations it was necessary to make the zero β and zero V_{W_Z} assumptions. Then, once the winds were computed, the accelerations could be transformed into the wind axis using the following equations which are simply the inverse of equations (1) through (3).

$$\begin{aligned} A_X = {} & (\cos\beta \cos\alpha \cos\theta \cos\psi + \cos\beta \sin\alpha \sin\phi \sin\psi + \cos\beta \sin\alpha \cos\phi \sin\theta \cos\psi - \sin\beta \sin\psi \cos\phi + \sin\beta \sin\phi \sin\theta \cos\psi) \cdot A_N \\ & + (\cos\beta \cos\alpha \cos\theta \sin\psi - \cos\beta \sin\alpha \sin\phi \cos\psi + \cos\beta \sin\alpha \cos\phi \sin\theta \sin\psi + \sin\beta \cos\phi \cos\psi + \sin\beta \sin\phi \sin\theta \sin\psi) \cdot A_E \\ & + (-\cos\beta \cos\alpha \sin\theta + \cos\beta \sin\alpha \cos\phi \cos\theta + \sin\beta \sin\phi \cos\theta) \cdot A_D \end{aligned} \quad (8)$$

$$\begin{aligned} A_Y = {} & (-\sin\beta \cos\alpha \cos\theta \cos\psi - \sin\beta \sin\alpha \sin\phi \sin\psi - \sin\beta \sin\alpha \cos\phi \sin\theta \cos\psi - \cos\beta \sin\psi \cos\phi + \cos\beta \sin\phi \sin\theta \cos\psi) \cdot A_N \\ & + (-\sin\beta \cos\alpha \cos\theta \sin\psi + \sin\beta \sin\alpha \sin\phi \cos\psi - \sin\beta \sin\alpha \cos\phi \sin\theta \sin\psi + \cos\beta \cos\phi \cos\psi + \cos\beta \sin\phi \sin\theta \sin\psi) \cdot A_E \\ & + (\sin\beta \cos\alpha \sin\theta - \sin\beta \sin\alpha \cos\phi \cos\theta + \cos\beta \sin\phi \cos\theta) \cdot A_D \end{aligned} \quad (9)$$

$$\begin{aligned} A_Z = {} & (-\sin\alpha \cos\theta \cos\psi + \cos\alpha \sin\phi \sin\psi + \cos\alpha \cos\phi \sin\theta \cos\psi) \cdot A_N \\ & + (-\sin\alpha \cos\theta \sin\psi - \cos\alpha \sin\phi \cos\psi + \cos\alpha \cos\phi \sin\theta \sin\psi) \cdot A_E \\ & + (\sin\alpha \sin\theta + \cos\alpha \cos\phi \cos\theta) \cdot A_D \end{aligned} \quad (10)$$

where

A_N = acceleration in the north direction

A_E = acceleration in the east direction

A_D = acceleration in the down direction

A_X = X flight path acceleration (longitudinal)

A_Y = Y flight path acceleration (lateral)

A_Z = Z flight path acceleration (normal)

The wind calculation method as described above seemed to degenerate during turning maneuvers. The reason for the problem can be seen by examining the term which was deleted in equation (3). The term is $\cos\theta \sin\phi \sin\beta$. For small bank angles and small sideslip the term vanishes, but as bank angle gets large (60 degrees or more) and sideslip becomes nonnegligible (on order of 1 degree) the term becomes significant in comparison with other terms in the equation. In many cases, the errors were so large that equation (7) became unsolvable. An alternative wind calculation method was required. The method developed was simply a solution of the basic velocity equation as follows:

$$V_T^2 = (V_{I_X} + V_{W_X})^2 + (V_{I_Y} + V_{W_Y})^2 + (V_{I_Z} + V_{W_Z})^2 \tag{11}$$

The basic problem with equation (11) is that it is one equation with three unknowns. Theoretically any three unique data points could be used to produce three equations in three unknowns. Practical data considerations required that the equation be reduced to two dimensions by assuming $V_{W_Z} = 0$.

There is simply not enough information in the Z direction during a constant altitude turn to determine the Z wind component. Equation (11) reduces to the following:

$$V_T^2 = (V_{I_X} + V_{W_X})^2 + (V_{I_Y} + V_{W_Y})^2 + V_{I_Z}^{\ 2} \tag{12}$$

The above one equation and two unknowns were expanded to N equations where N is the number of data points in the data run. The solution was obtained by minimizing the sum of the squared residual error of equation (12) with respect to each of the unknowns. The resulting two equations in the unknowns are as follows:

$$\sum_{i=1}^{N} \left[V_T^2 - (V_{I_X} + V_{W_X})^2 - (V_{I_Y} + V_{W_Y})^2 - V_{I_Z}^{\ 2} \right] \cdot (V_{I_X} + V_{W_X}) = 0 \tag{13}$$

$$\sum_{i=1}^{N} \left[V_T^2 - (V_{I_X}^{\ 2} + V_{W_X})^2 - (V_{I_Y} + V_{W_Y})^2 - V_{I_Z}^{\ 2} \right] \cdot (V_{I_Y} + V_{W_Y}) = 0 \tag{14}$$

Equations (13) and (14) are solved by a two dimensional Newton-Raphson iteration scheme. In order to utilize the winds computed above, it was first necessary to compute airspeed velocities in the body axis as follows:

$$V_{B_X} = (\cos\theta \cos\psi)(V_{I_X} + V_{W_X}) + (\cos\theta \sin\psi)(V_{I_Y} + V_{W_Y}) - \sin\theta\, V_{I_Z} \tag{15}$$

$$V_{B_Y} = (-\cos\phi \sin\psi + \sin\phi \sin\theta \cos\psi)(V_{I_X} + V_{W_X}) + (\cos\phi \cos\psi + \sin\phi \sin\theta \sin\psi)(V_{I_Y} + V_{W_Y}) + \sin\phi \cos\theta\, V_{I_Z} \tag{16}$$

$$V_{B_Z} = (\sin\phi \sin\psi + \cos\phi \sin\theta \cos\psi)(V_{I_X} + V_{W_X}) + (-\sin\phi \cos\psi + \cos\phi \sin\theta \sin\psi)(V_{I_Y} + V_{W_Y}) + \cos\phi \cos\theta\, V_{I_Z} \tag{17}$$

To compute α and β the following equations were used.

$$\alpha = \tan^{-1}(V_{B_Z}/V_{B_X}) \tag{18}$$

$$\beta = \sin^{-1}(V_{B_Y}/V_T) \tag{19}$$

The problem was completed using equations (8) through (10) to transform accelerations from the earth axis system to the flight path axis system.

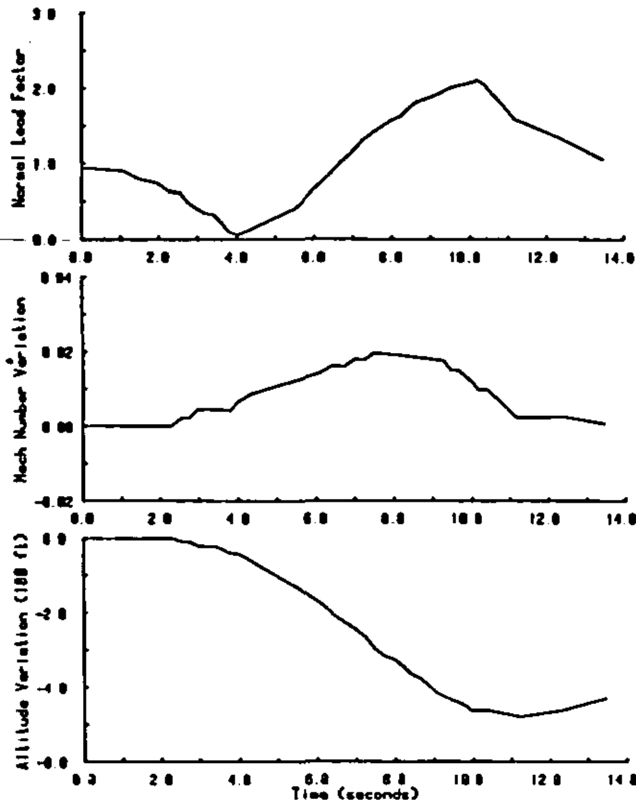

Figure 1 Roller Coaster Time History

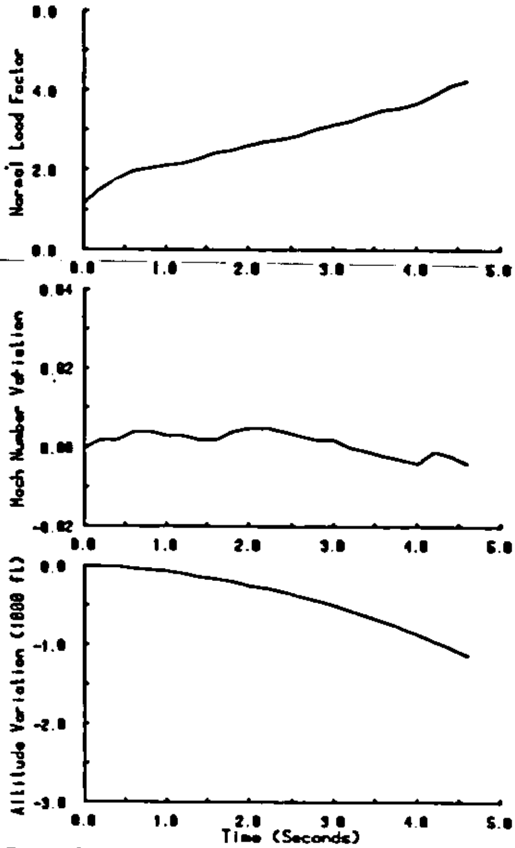

Figure 3 Split S Time History

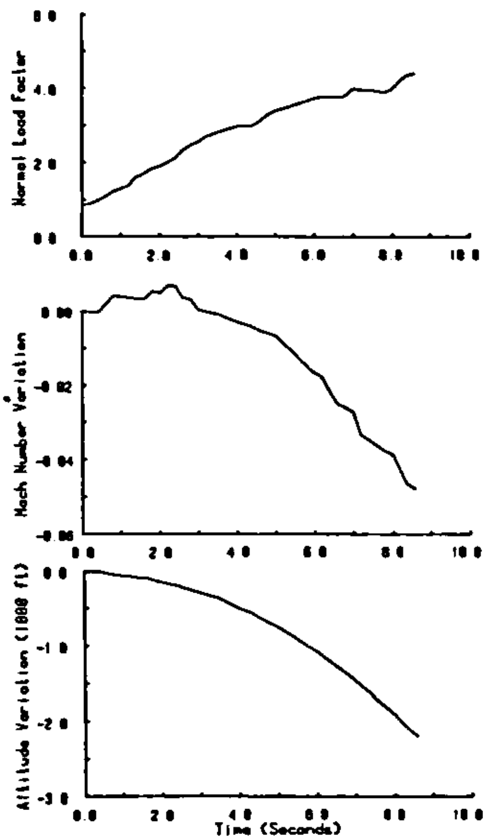

Figure 2 Wind Up Turn Time History

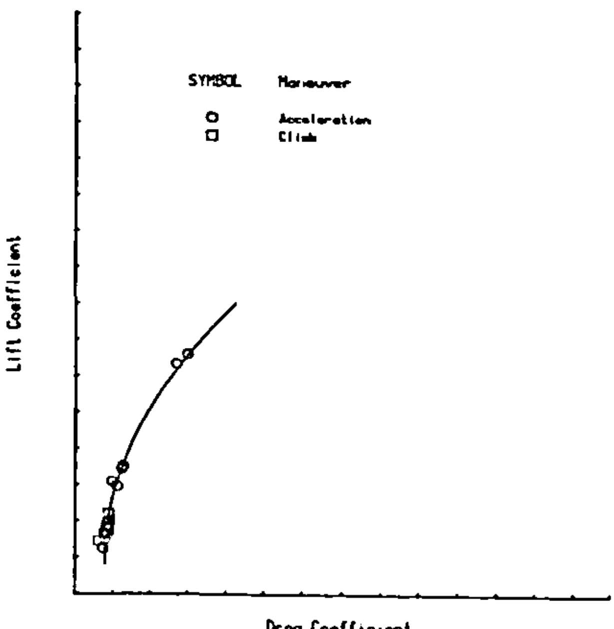

Figure 4 Acceleration and Climb Drag Polar

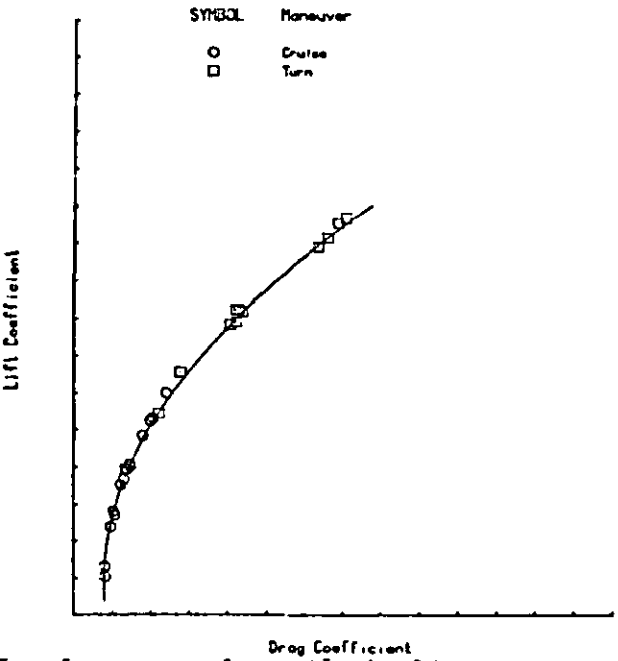

Figure 5 Cruise and Turn Drag Polar

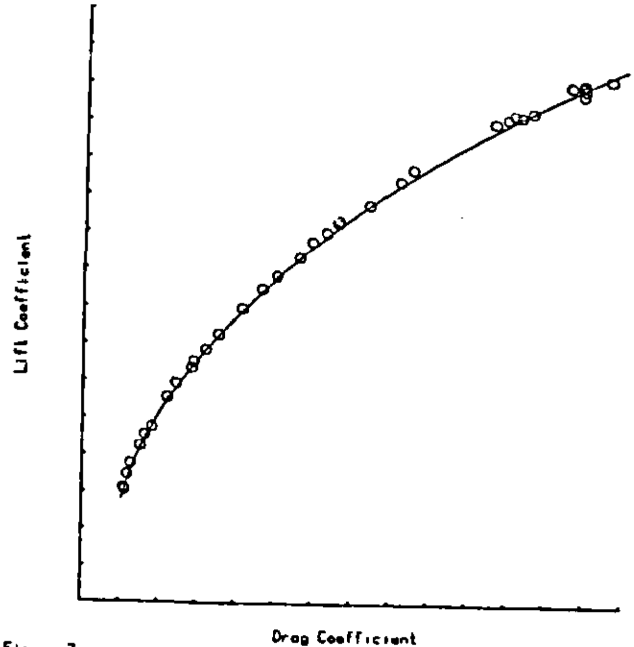

Figure 7 Wind Up Turn Drag Polar

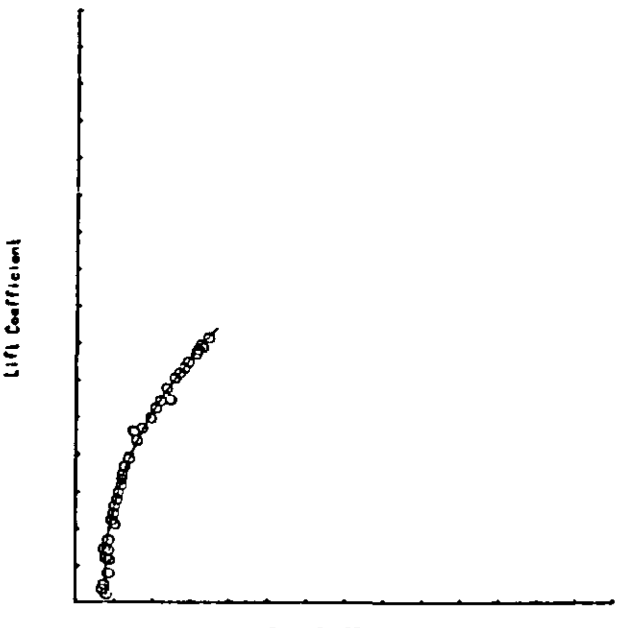

Figure 6 Roller Coaster Maneuver Drag Polar

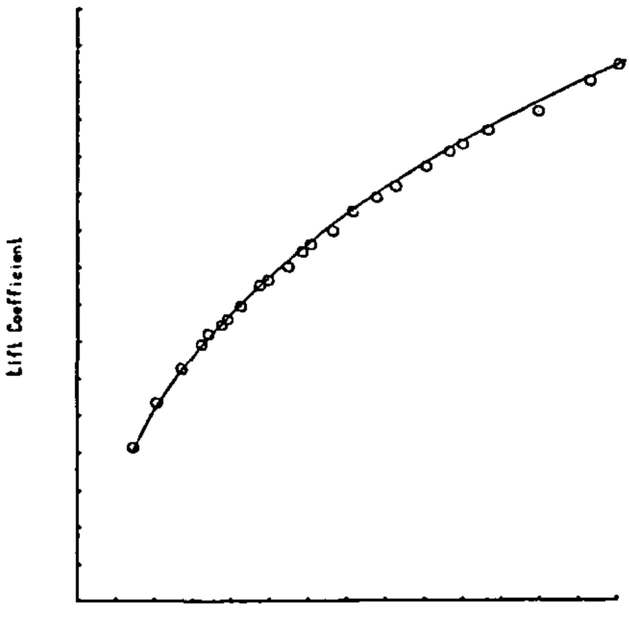

Figure 8 Split S Drag Polar

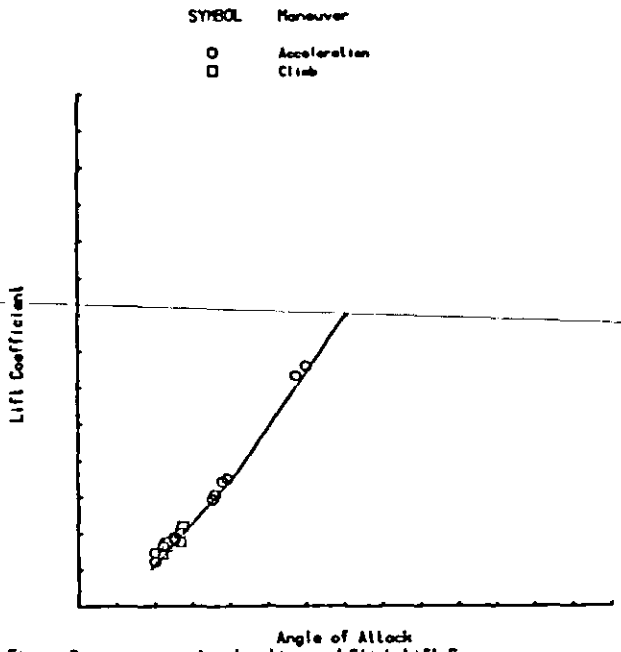

Figure 9 Acceleration and Climb Lift Curve

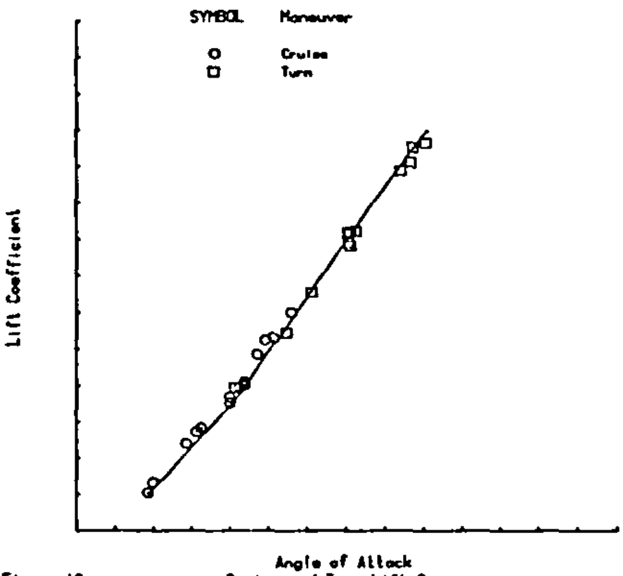

Figure 10 Cruise and Turn Lift Curve

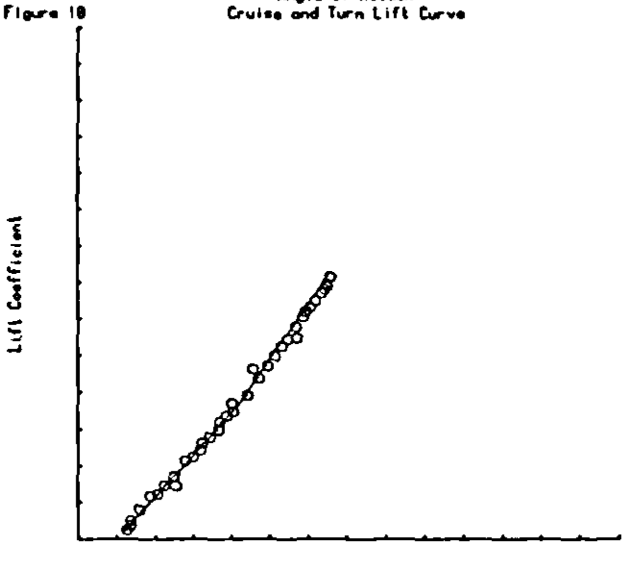

Figure 11 Roller Coaster Maneuver Lift Curve

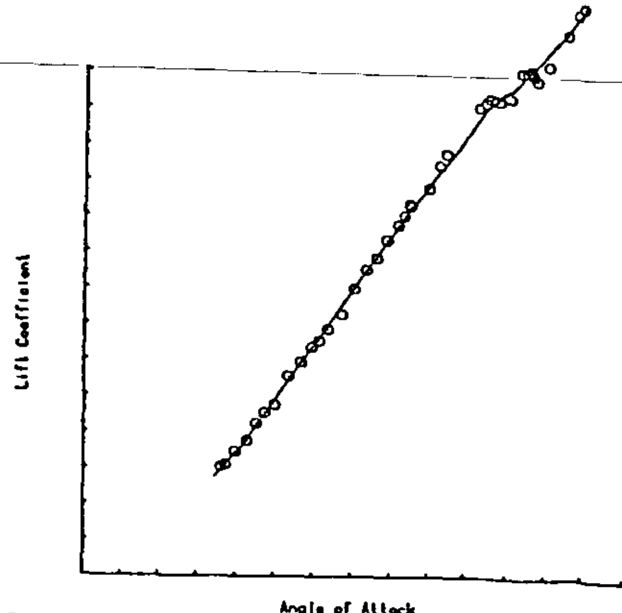

Figure 12 Wind Up Turn Lift Curve

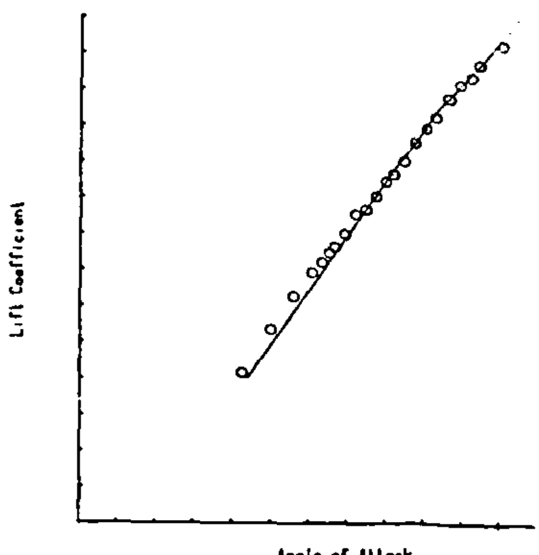

Figure 13 Split S Lift Curve

THE EVOLUTION OF FLUTTER EXCITATION AT MCDONNELL AIRCRAFT

by
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Abstract

Flight flutter testing involves exciting an aircraft's natural modes of vibration under various flight conditions, and measuring the response to this excitation. The critical flutter speed is then predicted from measurements acquired at subcritical speeds.

Many techniques have been employed to produce in-flight excitation of an aircraft's natural modes. Until the early 1960's, the primary technique involved pilot-induced excitations in the form of stick raps and rudder kicks. In the mid-1960's, the McDonnell Aircraft Company (MCAIR) began using electronic function generators that provide forcing functions to the aircraft control surfaces via the flight control system. These generators have incorporated features such as cockpit control panels, various shutdown mechanisms to ensure flight safety, and the use of telemetry to allow fast, real-time decision making. The first flutter generators were primarily analog devices, but the latest ones are easily used microprocessor-based units which offer flexibility and precise control of excitation.

Introduction

An important factor in the design of a new aircraft is freedom from flutter throughout the flight envelope. With trends toward higher speeds, lighter materials, and diverse wing configurations, flutter considerations have become even more critical. In spite of significant advances in theoretical flutter analysis and wind tunnel testing, flight flutter testing remains the conclusive way to verify that an aircraft is free from flutter. This is especially so in the case of military aircraft, where variations in external stores may alter the flutter characteristics considerably, thus complicating theoretical analysis and modeling.

When a wing or other structure in an airstream is subjected to a disturbance, the resulting oscillatory motion induces unsteady aerodynamic forces on the structure. At low speeds, these forces tend to oppose the motion, thus providing positive damping to the system. As speed increases, though, the phase relationship between the forces and the motion changes. This may provide additional damping for some modes, but in other modes, the effective damping may be decreased because of the energy extracted from the airstream. At the flutter speed, the net damping of the system is reduced to zero, thereby sustaining a neutrally stable oscillation. Any further increase in speed may produce negative net damping, resulting in divergent oscillation and possible structural failure. A flutter mechanism typically involves the coupling of two or more natural vibration modes.[1,2]

In a flight flutter test, the frequency and damping of the important modes are determined under various flight conditions. To do this, it is necessary to excite the aircraft, measure its response, and often, to measure the excitation as well.

To accurately characterize the flutter behavior of the aircraft, good quantitative frequency and damping data must be obtained for all modes of interest.[3] Also, since flutter testing is very costly and paces the expansion of the flight envelope, rapid and accurate techniques have been developed to minimize the number of test flights. The potentially destructive nature of flutter makes flight safety another paramount consideration; the flight envelope must be expanded cautiously, systematically, and only after thorough analysis of available data. All modes of interest must be adequately excited, and closely spaced modes must be separated, either by excitation or analysis. The excitation level must be high enough to be distinguished from noise due to turbulence or buffeting. Finally, the aerodynamic and structural characteristics of the aircraft must be altered as little as possible to minimize any effects on flutter behavior.[4]

At MCAIR, we've been working on these problems since the early 1950s. Figure 1 shows some of the excitation techniques we've used. For us, the modern age of flutter excitation began with the advent of the F-4 Stabilator and Aileron Exciters in the mid-1960s.

Early Techniques In Flutter Excitation

The first flutter excitation technique at MCAIR was simple impulse excitation produced by the pilot's striking the aircraft controls with stick raps and rudder kicks. This technique was successfully used through the Fifties on such aircraft as the XF-88, the F2H Banshee, the F3H Demon, and the F-101 Voodoo. It even saw some use in the early flutter testing of the F-4 Phantom II. Its use in flutter testing of the F2H-2 is described in detail in Reference 5. Symmetrical impulses were induced by sharp forward or aft stick raps. Antisymmetrical impulses were induced by sharp lateral stick raps and by rudder kicks. Testing followed an empirical procedure of exciting the aircraft at small increments in speed and fuel loading and measuring

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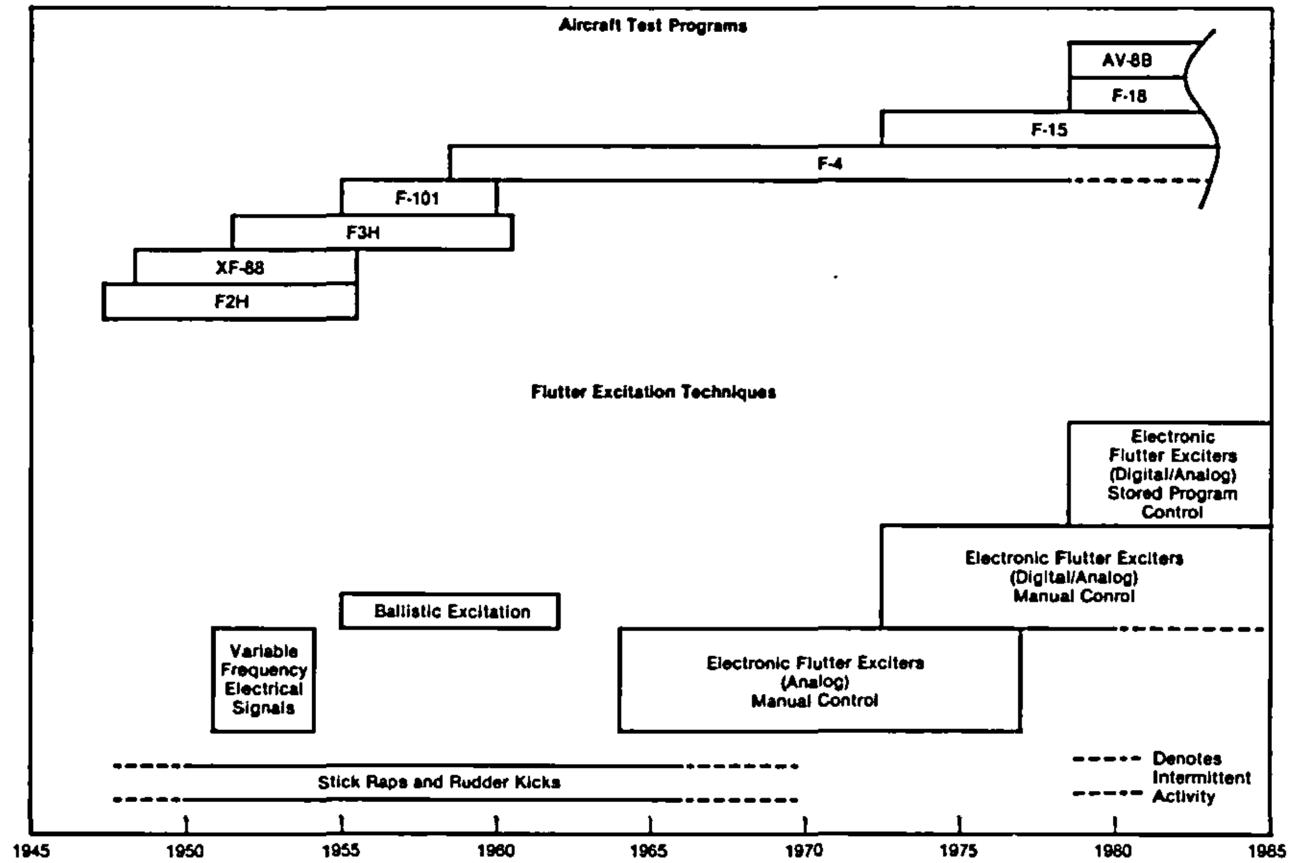

FIGURE 1. EVOLUTION OF FLUTTER EXCITATION TECHNIQUES AT MCAIR

the transient response with accelerometers. The responses were recorded on oscillographs and examined for mode, frequency, and rate of decay. This data was qualitatively evaluated to demonstrate adequate damping and freedom from flutter under the test conditions. The flight envelope was expanded cautiously after the extrapolation of data collected from previous flights. The manual excitation technique was safe enough for the F2H-2 because theoretical analysis had predicted a very gradual transition to instability (mild flutter). This prediction was substantiated by the test results.

The manual excitation technique has the obvious merit of simplicity. No special equipment or alteration to the aircraft is required. Test duration for each manual impulse is short, thus allowing many impulses to be applied during a single flight. But manual excitation has its disadvantages. First, the excitation is limited to low frequency ranges, usually with an upper limit of about 10 Hertz depending on the surface and the aircraft. Second, the applied force is not precisely repeatable because it is produced by the pilot. This results in variations in both the amplitude and frequency content of the response. Finally, unless a mode is very poorly damped to start with, this technique does not provide enough excitation energy for modes with natural frequencies above 5 or 6 Hertz to give valid quantitative data for those modes.

As early as 1952, a unique frequency response technique was used during transonic flutter tests of the XF3H-1 to investigate limit-cycle buzz of the fin-rudder system. Buzz is flutter with a single degree of freedom which is usually observed on trailing edge control surfaces. This unexpected buzz had been encountered while using the manual excitation technique. During use of the frequency response technique, stability was added to the system through higher control system stiffness and a damper on the rudder control rod. Excitation was provided by oscillating the rudder with the introduction of variable frequency electrical signals into the rudder servo of the autopilot system. The rudder hydraulic actuator was controlled by a pilot valve which was driven sinusoidally by an electrical exciter motor. Frequency was varied over the range of 5 to 35 Hertz by an automatically operated rotary switch in the cockpit which incremented the frequency every 3 seconds. Deflections of up to 1.5° were produced on the rudder. The excitation force and the rudder deflection were recorded on an oscillograph. Data obtained through the frequency response technique defined the margin of stability introduced to the originally unstable rudder by the design changes and verified freedom from buzz in the new configuration. References 6 and 7 describe this frequency response technique in detail and show that artificial stability may be added to a system for test purposes without complicating the interpretations of test results of the original system configuration.

The frequency response technique was a success because it provided adequate excitation to test the mode of interest and demonstrated freedom from buzz in the modified configuration. The excitation employed in this technique was the forerunner of later electronic function generators and marked MCAIR's first use of a form of swept sinusoidal excitation.

Flutter testing of the F-101 relied primarily on stick raps to excite the aircraft, but ballistic exciters called thrusters were also used. The thrusters were basically small rocket engines discharged on the wing tips to induce an impulse excitation. The rockets were powered by a sheet form of nitroglycerine, which was cut according to the size of impulse desired. Each wing tip accommodated six thrusters, three on top and three on bottom, which could be detonated individually by a 28V electrical signal. The thrusters were not very successful, primarily due to a difficulty in detonation at high altitudes (above 25,000 feet) where the propellant froze, a problem which might have been corrected by using a different propellent. The use of thrusters or similar ballistic exciters has several good points. Thrusters are small and simple, and do not disturb the vibrational characteristics of the aircraft. They can be placed as desired so as to best excite the modes of interest. Also, thrusters produce excitations and responses of short duration, which can be especially important under transient test conditions such as in a dive. Among their disadvantages is the fact that each thruster can only be fired once. Attempts to fire thrusters simultaneously met with only limited success because slight variations in timing of firing disturbed the phasing and symmetry of the resulting excitation. Finally, control of the amplitude and frequency content of the impulse is limited and may be affected by environmental conditions.

The use of inertial excitation, in the form of a rotating mass with a variable eccentric, was considered for testing of the F-101, but was rejected for two reasons. First, a very large mass would be required to produce enough excitation. Such a mass would require a lot of power to drive and could significantly affect the vibrational characteristics of the plane. Second, it is difficult to bring an inertial exciter to an abrupt halt. This could present a flight safety problem and would also hinder the technique of applying, then abruptly removing, harmonic excitation to produce a decaying response.

Flutter testing of the F-101 saw MCAIR's first use of telemetry for flutter data. Aircraft responses were telemetered to the ground station and plotted on oscillographs to provide time histories of the responses. This provided the first limited means of real-time flutter analysis.

Flutter testing of the F-4 in its many versions and configurations of stores spanned a period of about 20 years. Early testing continued to rely on stick raps and rudder kicks. Ballistic techniques also saw limited use when charges were detonated on the stabilators to excite certain modes. Data reduction and analysis still depended on obtaining frequency and decay data from the oscillograph-recorded transient response.

Introduction of the Electronic Flutter Exciter

In the mid-1960s, significant changes took place in the flutter testing of the F-4. Central to these changes was the use of electronic function generators to excite the ailerons and stabilator through the introduction of excitation signals to the servos of the auto pilot system. Since the introduction of these units, almost all flutter excitation at MCAIR has been produced by some type of electronic flutter exciter driving the aircraft control surfaces via the flight control system (see Figure 2). Table 1 lists significant characteristics of the generators used on the F-4 Phantom II, the F-15 Eagle, the F-18 Hornet, and the AV-8B Harrier.

F-4 Flutter Exciters

The F-4 flutter exciters were the Stabilator Exciter and the Aileron Exciter. These units were analog devices, fabricated entirely from discrete components. Their designs were based upon two prototype units (the Aileron Flutter Generator and Stabilator Flutter Generator) which had been developed in 1964 and 1965. The Stabilator Exciter was designed in 1965 and saw use during flutter testing of the stabilator with a slotted leading edge in early 1966. The Aileron Exciter was designed in 1967 and was used from that time until the mid-1970s for periodic wing/store flutter testing on the F-4. These exciters were very similar in most respects. The output characteristic, dictated by the input requirements of the autopilot servos, was a 400 Hertz sine wave carrier, amplitude modulated by a low-

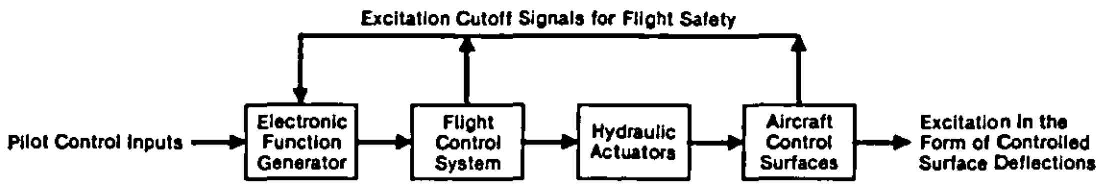

FIGURE 2. FLUTTER EXCITATION USING AN ELECTRONIC FUNCTION GENERATOR

TABLE 1. CHARACTERISTICS OF ELECTRONIC FLUTTER EXCITERS

	F-4 Stabilator Exciter	F-4 Aileron Exciter	F-15 FECU	F-18 FECU	AV-8B FECU
Control Surfaces Excited	Stabilator	Ailerons	Ailerons Stabilators	Ailerons Rudders Stabilators	Ailerons Rudder Stabilators
Electronic Device Technology	Discrete Components (Transistors, Etc.)	Discrete Components (Transistors, Etc.)	TTL, Analog IC's	CMOS (Digital and Analog)	CMOS (Digital and Analog)
Excitation Frequency Range and Accuracy	8 Hz to 30 Hz ±0.33 Hz	2 Hz to 16 Hz ±0.33 Hz	3 Hz to 55 Hz ±0.055 Hz	2 Hz to 60 Hz ±0.01 Hz	2 Hz to 70 Hz ±0.01 Hz
Excitation Amplitude Range	No Information Available	Ail: ±1.1V	Ail: ±1.00 V Stab: ±10.0V	Ail: ±2.86V Rud: ±3.33V Stab: ±5.80V	Ail: ±2.86V Rud: ±3.33V Stab: ±5.80V
Excitation Amplitude Accuracy	No Information Available	About ±10%	±5% Full Scale	±1% Full Scale	±1% Full Scale
Full Scale Surface Deflection	±2°	About ±1° or ±2°	Ail: ±4° Stab: ±1.8°	Ail: ±10° Rud: ±10° Stab: ±10°	Ail: ±2° Rud: ±5° Stab: ±3°
Forms of Excitation	Square Wave	Square Wave	• Sinusoidal • Bump (Pulse)	• Sinusoidal • Random	• Sinusoidal • Random • Bump (Step)
Sweep Rate Control	Internally Set to Fixed Rate	Internally Set to Fixed Rate	Slow and Fast Rates Internally Set	EPROM Programmable	EPROM Programmable
Control of Excitation	Manual	Manual	Manual	Stored Programs	Stored Programs
Visual Cues for Pilot	7	8	10	28	13
Adjustable/Programmable Parameters	9	11	18	22	21
No. of Stored Programs/ Method of Program Entry	N/A	N/A	N/A	16 Programs Entered Via Front Panel Keyboard Before or During Flight	99 Programs Stored In EPROM Module as Needed Before Flight
In-Flight Excitation Setup (Typical Time Required)	Potentiometer and Switch Adjustments (30 Seconds)	Potentiometer and Switch Adjustments (30 Seconds)	Potentiometer and Switch Adjustments (30 Seconds)	Auto-Increment (1 Second) Program Recall (5 Seconds) New Program Entry (30 Seconds)	Program Recall (5 Seconds) Thumbwheel Freq. Selection (5 Seconds)

frequency square wave. The excitation amplitude and frequency were equal to the modulation amplitude and frequency. As long as the units were powered, they presented the 400 Hertz carrier to the excitation outputs. In the Aileron Exciter, this carrier served as a bias signal to statically deflect the ailerons, thus removing freeplay.

Both exciters were located in the cockpit and were controlled by the pilot from switches and potentiometers on the front panel. Figures 3 and 4 show the front panel layouts of these units. Features controllable from the front panel were selection of manual (dwell) or automatic (sweep) mode, initiation and reset of a run (excitation applied), excitation amplitude (from 0 to a calibrated maximum), dwell frequency, sweep start frequency, and sweep stop frequency. The excitation amplitude was usually set to produce the desired deflection as part of the preflight procedure.

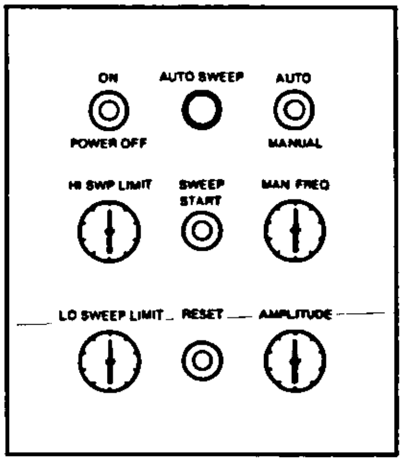

FIGURE 3. FRONT PANEL OF F-4 STABILATOR EXCITER

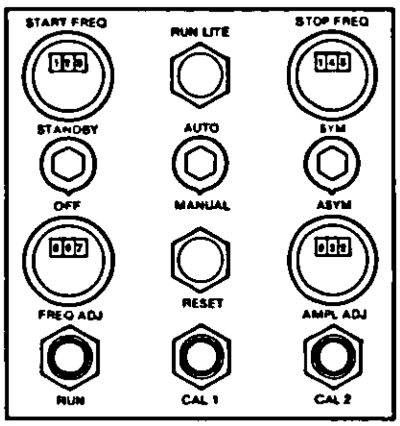

FIGURE 4. FRONT PANEL OF F-4 AILERON EXCITER

Internally calibrated parameters included bias amplitude (Aileron Exciter only), maximum deflection, dwell mode duration (1 to 10 seconds), and sweep rate (from about .067 Hertz/second to .133 Hertz/second). Frequency sweeps were linear and proportional in duration to the sweep range and rate. The reset switch on the front panel could be used to force an instant, premature termination of a run. This feature provided a measure of flight safety.

There were several significant differences between the Stabilator and Aileron Exciters. Some of these differences are apparent in the comparison chart of Table 1. The Stabilator Exciter had an excitation frequency range of 8 to 30 Hertz, whereas the frequency range for the Aileron Exciter was 2 to 16 Hertz. Deflections produced on the stabilator were up to $\pm 2°$. No records could be found of the maximum deflection for the ailerons, but it is thought to have been about $\pm 1°$ or $\pm 2°$. The actuators for the ailerons and stabilator had to be modified to provide adequate gain and frequency response.

The Aileron Exciter had the added feature of being able to produce symmetrical or antisymmetrical excitation. This was done by providing separate excitation to the port and starboard ailerons, and setting the phase relationship between the two outputs to 0° (for antisymmetrical) or 180° (for symmetrical) as desired. Symmetry control on the Stabilator Exciter was precluded by the fact that the stabilator of the F-4 was a single surface. A later version of the Aileron Exciter incorporated an inverse-tapered amplitude characteristic to compensate for roll-off in the aileron frequency response.

In addition to the excitation outputs, the exciters provided two telemetry outputs: a square wave at the excitation frequency and an analog output which was voltage proportional to the excitation frequency. This frequency information and the responses from appropriate transducers were passed via the telemetry link to a transmissibility plotter (T-plotter). The T-plotter produced a real-time plot of gain vs. frequency, but provided no phase information. The gain information was not absolute since the T-plotter could not be calibrated to a reference. Rather, the T-plots gave qualitative information such as resonant frequencies and points of low damping. To eliminate noise outside the frequency of interest, the T-plotter incorporated a tracking filter controlled by the telemetered frequency information. On the Aileron Exciter, a two-point verification of frequency calibration for the telemetry outputs could be obtained by depressing the CAL 1 and CAL 2 switches on the front panel, one at a time.

In spite of their limitations, the Stabilator and Aileron Exciters were a big step forward. They provided limited (though imprecise) control of excitation amplitude and frequency. The dwell mode provided a high degree of selectivity of excitation and guaranteed a concentration of excitation into the mode of interest. This, in turn, raised the signal-to-noise ratio of the response. The sweep mode provided a relatively quick way to provide excitation over a broad frequency range. The use of the T-plotter made some real-time data reduction and analysis possible.

The excitation of higher frequency modes using the exciters was difficult because of the rapid roll-off in the frequency response of the control surfaces. This roll-off had one positive effect in that the square wave modulation of the excitation produced near-sinusoidal deflections. This occurred because the higher order harmonic components in the square wave were attenuated significantly by the frequency response of the control surface actuators.

Flutter Excitation On The F-15

Flutter testing of the F-15 witnessed considerable advancement over earlier techniques. Most of these innovations were dictated by a full commitment to the flutter prediction technique developed by N. H. Zimmerman and J. T. Weissenburger, which is described in detail in Reference 3. This analysis technique predicts the critical flutter speed based on quantitative frequency and damping data acquired at subcritical speeds. Previous techniques had been based on the extrapolation of damping vs. velocity and the verification of adequate damping throughout the flight envelope. The decision to proceed to higher speeds was based on the extrapolation of test data from lower speeds. The danger of this method is that some configurations are susceptible to abrupt degradation of damping and loss of stability without warning from prior test results.

The new technique derives a flutter margin parameter from measured frequency and damping data. The margin parameter is a more fundamental indication of system stability than damping. Its curve changes in a predictable manner, not being subject to the abrupt reversals sometimes encountered in damping data. It not only provides a basis for safe decisions to expand the test envelope, but allows measurement of the flutter margin throughout the flight envelope and reasonably accurate prediction of the critical flutter speed. This technique became the basis for the F-15 flutter test program, which is described in detail in References 8 and 9.

Confidence in flutter margin analysis permitted the pursuit of a lightweight airframe design with minimum allowable flutter margins for the F-15. Use of the technique also established many of the requirements for flutter excitation, instrumentation, and data reduction. Central to these requirements is the need for deterministic excitation and accurate, quantitative data.

The source of flutter excitation for the F-15 was an electronic function generator called the Flutter Exciter Control Unit (FECU). This unit supplied sinusoidal excitation to the ailerons and stabilators through the flight control system, which was slightly modified to accept inputs from the FECU. The FECU was designed in 1971 and made extensive use of integrated circuits, both digital (TTL) and analog.

Like the F-4 Aileron and Stabilator Exciters, the FECU resided in the cockpit and was controlled from switches and potentiometers on the front panel (see Figure 5). However, the F-15 FECU design improves on the F-4 exciter designs in many respects. It has an expanded frequency range (3 to 55 Hertz) and a scheme for providing accurate setting of frequency. The start and stop frequency settings are measured internally and appear in 3-digit displays on the front panel to provide a means of correlating the potentiometer settings with the generated frequencies, which are within .055 Hertz of the displayed values. Excitation amplitude can be set from 0 up to 1 volt

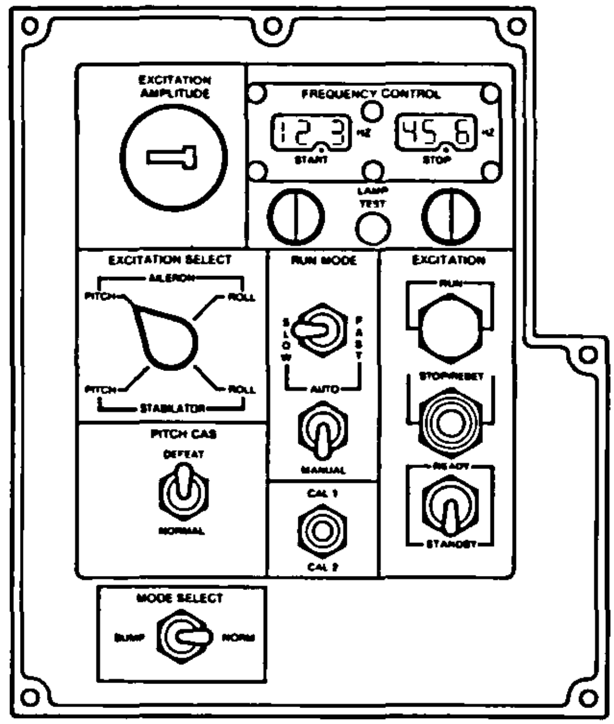

FIGURE 5. FRONT PANEL OF F-15 FLUTTER EXCITER CONTROL UNIT

±5% for the stabilators and up to 10 volts ±5% for the ailerons. This setting is done on a potentiometer with dimensionless digital readouts from 0 to 999 for repeatability of setting. In addition to the dwell and sweep modes, the FECU has a bump mode in which a single positive, half-cycle, sinusoidal pulse is output. A switch to disable the Pitch Control Augmentation System (CAS) allows flutter data to be acquired both in the normal CAS-on condition and with the CAS off. Internal adjustments to the FECU permit calibration of dwell duration time (0 to 10 seconds), slow and fast sweep rate (between .25 Hertz/second and 2 Hertz/second), minimum excitation frequency, a 10.000 volt reference, and reduction of total harmonic distortion of the sinusoids (to about 1%). The CAL switch on the front panel provides for verification of the frequency calibration at 5.0, 25.0, and 50.0 Hertz.

The aileron excitation outputs are applied to the Aileron Servo Control Unit (ASCU) which drives the ailerons via the Hydraulic Lateral Control Series Servo (HLCSS) to produce up to ±4° peak deflections. A modified pitch computer of the Automatic Flight Control System (AFCS) accepts the stabilator excitation outputs and commands the stabilators to produce maximum deflections of about ±1.8°. In addition to the excitation signals, the FECU supplies four 28 VDC bi-level lines indicating surface (aileron/stabilator) and symmetry (pitch/roll) to the ASCU and AFCS to control application of the excitation signals to the control surfaces with the appropriate phasing.

To ensure safety-of-flight, the FECU monitors four input signals to detect abnormal conditions. In the event of any abnormal condition, the excitation outputs are cut off and the FECU is returned to the standby status. The four abnormal conditions monitored on these inputs are excessive bending moment on the stabilators, greater than ±6° deflection on the stabilators, an over or under-G condition, or an abnormal condition on the HLCSS detected by the ASCU. As added precautions, the flight control system incorporates provisions to prevent a hardover condition and a paddle switch on the control stick which disengages the FECU.

The telemetry outputs of the FECU consist of bi-level outputs which completely define FECU operating status and two frequency outputs, one analog and one digital. These allow continuous monitoring of FECU operation at the ground station. The analog frequency output consists of a 4.8 volt peak-to-peak constant amplitude sinusoid that is synchronous in frequency with the excitation signal. This output is used to control tracking filters. The digital frequency output is a 16 bit NRZ-L binary serial word (with three dummy bits) containing a measured value of excitation frequency with .01 Hertz resolution.

In addition to the telemetry outputs of the FECU, 32 selected strain gage outputs are telemetered for real-time analysis. Strain gages are placed to measure both control surface deflection and aircraft response. The ground station supports strip chart plotting of strain gage outputs. Furthermore, 16 lissajous displays may be used to beat pairs of outputs together as an indication of potential flutter mechanisms. During a frequency sweep, a transmissibility plot (T-plot) presentation of gain and phase versus frequency can be displayed in real-time on a CRT. In addition, six near real-time T-plot presentations for selected modes can be produced on an electrostatic plotter. These T-plots represent a considerable advancement over those produced during testing of the F-4. Accurate plots of both magnitude and phase are computed from digitized data. Additionally, the response characteristics are normalized to measured control surface deflections, thus producing quantitative results and more repeatable plots. These data reduction techniques provide considerable capability for real-time analysis and evaluation of aircraft stability.

In addition to telemetered data, all flutter data is recorded on magnetic tape for post-flight processing. This processing includes the generation of T-plots for all parameters of interest. These T-plots provide much of the frequency and damping data upon which the flutter margin calculations are based. Additional frequency and damping data is acquired using the dwell/decay technique and by applying the Kennedy-Pancu vector plot technique to sweep data.[10] Frequency and damping are derived both manually and through computer processing of data.

In addition to use of the FECU, the aircraft is sometimes flown through turbulence to provide excitation across a wide frequency spectrum and allow determination of critical frequencies. This technique is particularly useful during transient flight conditions such as dives.

During integration testing, several minor adjustments had to be made to the flutter excitation setup. Aileron freeplay proved to have a detrimental effect on aileron excitation. This was resolved by generating a bias signal external to the FECU, statically deflecting the ailerons ±3°, thus aerodynamically preloading them to remove freeplay. Testing also revealed that the desired deflections on the ailerons and stabilators required drastically different settings on the amplitude potentiometer. A circuit modification was made to produce the desired deflections (about ±0.8° maximum for the ailerons and ±0.45° maximum for the stabilators) at nearly identical settings on the dial, thus reducing pilot workload.[8]

Initial flight flutter testing of the F-15 was performed between November 1972 and March 1974,[8] but testing continues on an intermittent basis for certification of modified versions of the aircraft and various stores configurations. The new excitation and data reduction techniques provided the quantitative data necesary to calculate flutter margins. Flutter margins of 15% or more were demonstrated for all modes.[9] One of the main shortcomings of the excitation techniques used on the F-15 is a rapid fall-off in magnitude of excitation at higher frequencies. This problem is especially apparent for the stabilators and is due to the frequency response limitations of the control surface actuators. Also lacking is a quick and accurate means of obtaining test results for transient flight conditions.

Present-Day Flutter Exciters

General Design Characteristics

The most recent flutter exciters at MCAIR are the F-18 FECU and the AV-8B FECU. These units are similar in many respects. Considerable improvements to F-15 techniques have been made by exploiting advances in integrated circuit technology. The most important example of this is the use of a microprocessor (the CDP 1802 CMOS microprocessor from RCA) to control overall FECU operation.

The FECU design philosophy delegates functions such as excitation signal generation, front panel interfacing, telemetry outputs, safety-of-flight monitors, and signal measurement to independent hardware blocks. These blocks are treated as I/O peripherals of the microprocessor with the overall operating characteristics of the FECU determined by the processor's firmware. This architecture makes the FECU easy to use and flexible, with significant modifications possible through changes to firmware.

Another significant improvement in the current flutter exciters is the digital amplitude and frequency control employed in the generation of excitation signals. This is made possible through the use of monolithic phase-locked loop (PLL) and digital-to-analog converter (DAC) technology. The excitation produced is much more accurate than was possible with previous flutter exciters. This improvement in accuracy is due not only to the use of better components, but also to the elimination of errors inherent in manual adjustment of potentiometers. The resulting accuracy in excitation is $\pm$.01 Hertz in frequency and $\pm$1% in amplitude.

The frequency synthesizer uses a PLL in a frequency-multiplying configuration with a stable input from a crystal oscillator to produce a high-rate reference frequency. The high-rate frequency clocks an up-down counter driving a DAC to generate a triangle wave at the excitation frequency. The triangle wave is then wave-shaped to produce the required sine wave.

In addition to sinusoidal excitation, the current FECUs are capable of generating a bandpass filtered pseudo-random output. This form of output is generated by feeding a pseudo-random pattern into a DAC. The bandpass filter is programmable with cutoff frequencies in the range of 2 to 63 Hertz.

The sinusoidal and random waveforms undergo several stages of amplitude scaling before being output. For the sine wave, the first stage consists of amplitude versus frequency scaling, programmable independently for each surface. This scaling is performed by feeding the output of an erasable programmable read-only memory (EPROM), programmed with the desired amplitude versus frequency characteristic, into an 8-bit multiplying DAC. The address to the EPROM is the binary representation of the excitation frequency, thus selecting the desired amplitude scaling factor. Since each surface has a separate EPROM, this feature allows compensation for the surface frequency response, producing a nearly flat surface deflection versus frequency characteristic for each of the surfaces.

The next stage of amplitude scaling is applied to both sinusoidal and random waveforms. This stage scales the signal by a programmable factor controlling percent of full scale excitation. Allowable values range from 5% to full scale with a resolution of 1%. The final stage applies a surface scaling factor to the waveform to produce the appropriate deflection on the selected surface. These scaling factors are set with resistive scaling networks and may be calibrated by internal potentiometer adjustments.

The FECU also incorporates an EPROM which controls the sweep rate characteristic in the sweep mode. This EPROM is addressed by the binary excitation frequency value (in .1 Hertz increments) to retrieve a value which controls the amount of time spent at the current frequency increment. Thus, variation in the contents of this EPROM can produce virtually any desired sweep rate characteristic, from linear to logarithmic to irregular. The characteristics typically used are logarithmic.

In addition to the AC excitation, the FECU generates a DC bias excitation to remove freeplay in the control surfaces by aerodynamically preloading them. This bias is programmable in a range of $\pm$5.0 degrees, as long as the maximum allowable surface deflection is not exceeded. On the F-18, the stabilitors are not provided with bias excitation, since they are generally loaded sufficiently and because the impact of stabilator bias on the flight path is unacceptable.

Like the F-15 FECU, the F-18 and AV-8B units monitor several inputs for abnormal conditions. Any detected abnormality disables the excitation output. The sources of the shutdown inputs vary between the F-18 and the AV-8B and will be described later. In addition to these inputs, the microprocessor can detect a number of internal fault conditions, both hardware and firmware. A detected fault disables the excitation outputs.

The excitation outputs are applied to the aircraft's flight control system which in turn drives the selected surfaces to produce the desired deflection. In the F-18, the heart of the flight control system is a pair of Flight Control Computers, each of which receives identical excitation signals. In the AV-8B, the excitation signals drive the Stability Augmentation and Attitude Hold System computer.

The telemetry outputs of the FECU include both an analog output and a set of PCM words. The analog output is a 5.0-volt peak-to-peak sine wave synchronous in frequency with the excitation (in sinusoidal modes). The PCM outputs consist of eight 16-bit serial words retrieved from the FECU by a time division multiplexed data system used for flight test instrumentation. These eight words define FECU operating status for display on the ground.

Data reduction and analysis techniques for flutter testing of the F-18 and AV-8B are basically the same as those used for the F-15. The most noteworthy difference is the use of a Hewlett-Packard Fourier Analyzer. This device uses Fast Fourier Transform techniques to convert the time histories of responses to the frequency domain in near real-time. This is especially useful in processing response data from the sweep and random modes, which provide excitation across a broad frequency spectrum. Application of the Fourier Analyzer speeds extraction of data, but the data is still used to calculate the flutter margins of the aircraft.

The AV-8B and F-18 FECUs have many characteristics in common. However, they differ in some respects, the most significant of which is the selection and storage of excitation parameters. Differences between the two units are discussed below.

F-18 Flutter Exciter

The F-18 FECU was designed in 1978 primarily using CMOS integrated circuits, both digital and analog. This unit supplies excitation to the ailerons, stabilators, and rudders via the Flight Control Computers. The excitation may be swept sinusoid, constant sinusoid, or a bandpass limited random signal. Excitation to the ailerons and stabilators may be symmetrical or antisymmetrical.

The FECU resides in the cockpit and is controlled from the front panel (see Figure 6). Excitation parameters are selected in complete sets called "programs". Each program contains all the characteristics required to completely specify an output excitation. Programmable parameters are surface (aileron, rudder,

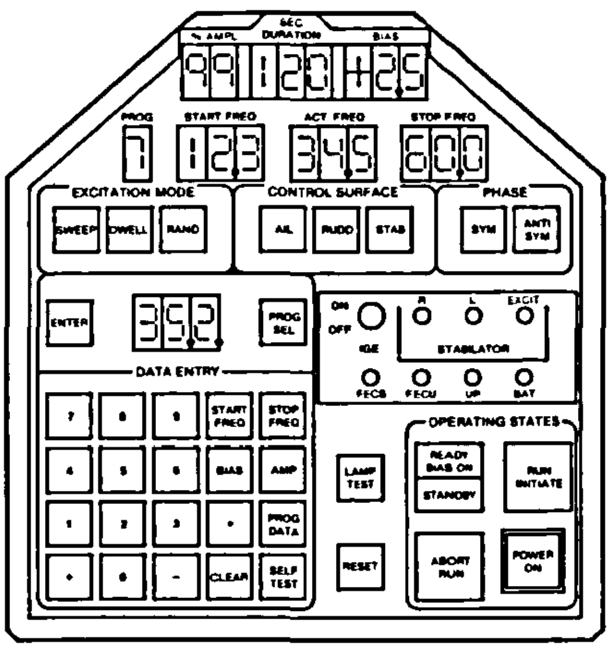

FIGURE 6. FRONT PANEL OF F-18 FLUTTER EXCITER CONTROL UNIT

stabilator), mode (sweep, dwell, or random), start and stop frequency (2.0 to 60.0 Hertz), percent amplitude, symmetry, bias deflection, and run duration. Programs are entered, stored, and recalled from a front panel keyboard. The selected program parameters appear in light emitting diode (LED) displays on the front panel. Up to 16 such programs can be stored in a nonvolatile memory within the FECU. In the sweep mode, the duration is computed as a function of start and stop frequencies and displayed. The FECU has three operating states: standby, ready, and run. In the standby state, no excitation is output. In the ready state, the programmed bias is applied at 1 degree per second and held. In the run state, the programmed excitation is applied. Runs can be halted at any time by pressing the ABORT RUN switch. The FECU also can be disengaged by a paddle switch on the control stick. Safety-of-flight inputs to the FECU include shutdown signals from each Flight Control Computer and strain gage outputs to indicate excessive loading of the stabilators.

The FECU supports a signal measurement circuit which may be used to monitor various signals, both output and internal. The FECU performs a self test function on demand which uses this circuit to make extensive voltage and frequency measurements. Although the measurement circuitry comprises less than 10% of the total components in the FECU, its capabilities provide a high degree of confidence in system integrity. Limited frequency measurements are made on a continuous basis to prevent the output of an erroneous signal.

The storage of excitation parameters in the form of programs considerably alleviates the pilot workload by doing away with the task of setting front panel potentiometers, thus allowing more testing per flight. Typically, the programs anticipated for use on a test flight are entered as part of the preflight procedure. Programs are then executed sequentially under the desired flight conditions. However, additional programs are often decided upon during the flight, or more than 16 programs are used. This requires the entry of programs inflight (at an off-speed flight condition), a procedure the pilots found cumbersome. As a result, a firmware modification has been made which provides a simplified programming sequence. This experience has had a considerable bearing on the AV-8B FECU design. Full Scale Development Flutter Testing on the F-18 was performed between late 1979 and early 1982 with limited testing still being performed. FECUs virtually identical to the F-18 version except for slight differences in the telemetry interface have been built for the Canadian and Australian F-18 test aircraft.

AV-8B Flutter Exciter

The AV-8B FECU was designed in 1980 and like the F-18 unit, primarily uses CMOS integrated circuits. The AV-8B and F-18 units are similar to the extent that many circuit cards are interchangeable, but there are significant differences. One such difference is that limited cockpit space requires the AV-8B unit to be designed in two packages, a cockpit control panel and an electronics package located elsewhere in the aircraft. These two packages communicate with each other over a half-duplex serial link. The pilot enters control information into the control panel, which transmits it to the electronics package via the serial link. The electronics package sends display information to the control panel. Other than this, the separation of the design into two packages has no effect on the function of the FECU.

The most significant differences between the AV-8B and F-18 units is in the storage of programs. Instead of allowing control panel entry of programs, the AV-8B FECU has up to 99 programs pre-stored in an accessible EPROM module. This eliminates the sometimes laborious programming sequences required for the F-18 FECU. The result can be seen in the simplicity of the front panel in the AV-8B unit (see Figure 7). Thumbwheel switches on the control panel require the pilot to select only the program number, start frequency, and stop frequency, thus greatly diminishing the effort required to specify excitation in flight. This allows more flutter testing to be performed per flight, advancing schedules and reducing costs. During the first week of flutter testing on the AV-8B, a program set was established which has not been changed since. A set of 99 programs with frequency parameters variable from the control panel proved to be more than sufficient.

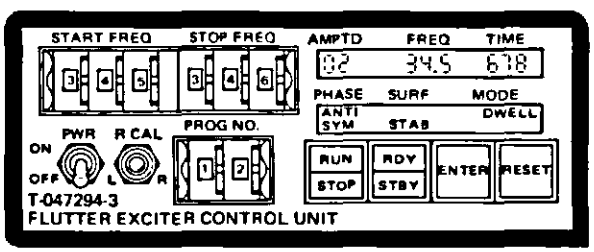

FIGURE 7. FRONT PANEL OF AV-8B FLUTTER EXCITER CONTROL UNIT

Like the F-18 FECU, the AV-8B unit has three operating states performing the same functions: standby, ready, and run. However, the AV-8B FECU makes transition between states simple and straightforward by blinking the pushbutton switch which normally comes next in the operating sequence.

Another new feature on the AV-8B FECU is the addition of the bump mode. The bump excitation consists of a step or ramp function to a programmed deflection with a rise time programmable from 0-99 milliseconds. The deflection is held for the programmed duration and then is decayed slowly over a period of 4.5 seconds. This mode is designed to approximate manual impulse excitation (such as the stick rap) and is included to expand the alternatives available to excite the modes of interest. The bump mode may not be used with bias because both deflections are generated using the same circuitry.

The AV-8B FECU supports a self test function performed by executing program number 0. This function uses the same measurement circuitry found in the F-18 FECU, comprising less than 10% of the total components. The AV-8B test function has been expanded through firmware modifications to provide a higher degree of fault isolation than available in the F-18 FECU.

Safety-of-flight inputs include a G-limit shutdown, an emergency disengage shutdown activated from a paddle switch on the control stick, and strain gage outputs to indicate excessive loading of the stabilators. A resistance calibration switch on the front panel allows verification of the stabilator strain monitors during self test. The final noteworthy feature of the AV-8B FECU is the use of custom liquid crystal displays (LCD) on the control panel. Experience with the F-18 indicated that the LED displays were difficult to read in bright sunlight, a problem which the LCD displays eliminated.

Full Scale Development Flutter Testing of the AV-8B began in 1982 and continues at this time. The FECU designed and built for the YAV-8B test aircraft is virtually identical to the AV-8B version except for minor packaging variations in the control panel.

Summary

Several conclusions can be drawn about the current FECUs in light of their use to date. The most significant improvements found in these units can be traced to their extensive use of digital electronics. This has not only improved the accuracy of the excitation, but has also made the flutter exciters more flexible and easier to use. This can be seen especially in the case of the AV-8B FECU, where pre-stored programs have greatly simplified control of excitation, thus allowing more tests per flight.

The programmability of sweep rate and amplitude versus frequency characteristics has proved to be useful as several sets of EPROMs were generated for each aircraft to be used under various test conditions. The random mode has been used to determine critical modal frequencies, particularly under transient flight conditions, similar to the way turbulence excitation was used on the F-15. However, extensive testing and statistical analysis of data are required to obtain quantitative data from random excitation.

The bump mode of the AV-8B FECU has not seen much use because it does not introduce a lot of energy into the aircraft structure. It was intended as an alternative to the other modes, but since these have been sufficient to excite all the modes of interest, the bump mode is not really needed.

At MCAIR, flight test instrumentation is currently undergoing a trend towards packaging miniaturization through the use of surface-mounted packages such as leadless chip carriers. These packaging techniques promise to provide as much as a threefold size reduction for future flutter excitation hardware. Furthermore, the advent of multipurpose interactive cockpit displays may eliminate the need for dedicated control panels on future flutter exciters, thus further reducing size and cost.

One significant problem remaining is adequate excitation of the modes of interest under transient flight conditions. The random mode held promise, but doesn't really provide the deterministic excitation necessary to produce quantitative results. One possibility is the impulse sine wave described in Reference 4. This form of excitation is a sinusoidal pulse with perfectly flat frequency content from 0 Hertz out to a variable frequency. The response produced resembles a decaying oscillation.

Another area of promise for excitation lies in the advances in avionics, particularly flight control systems. The next generation of aircraft will have increased maneuverability and require sophisticated closed-loop flight control systems. These systems may be capable of providing flutter excitation to the control surfaces simply by the way they are programmed, thus requiring no special hardware. The logical extension of this is the incorporation of active flutter suppression as part of the system.

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GRUMMAN'S DISPLAY AND CONTROL SYSTEM (DCS)
- WORK-IN-PROGRESS

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PAPER NOT AVAILABLE AT TIME OF PRINTING

THE MICROCOMPUTER IN
FLIGHT TEST DATA REDUCTION

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ABSTRACT

In support of flight test activities of the Coast Guard Short Range Recovery (SRR) helicopter, the HH-65A, Aerospatiale Helicopter Corporation (AHC) is utilizing a low-cost, high performance microcomputer system for flight test data reduction and analysis. A number of advantages have evolved with this system over the more expensive traditional large system approach. This paper provides an introduction to Aerospatiale's approach to automated data reduction and to the capabilities of this system.

Introduction

The advent of rapidly declining costs for computer hardware, due to the advances in the electronics industry, has made advanced computational capabilities available at a small fraction of the cost of obtaining such capabilities only a few years ago. These new systems are also easier to integrate with other electronic systems and more straight-forward to program than their predecessors. These price/performance breakthroughs now present a cost-effective alternate to the traditional approach of automated flight test data reduction.

When selecting computational hardware in the past, it has been necessary to estimate usage and data handling capacity on the high side to allow for unprojected growth or changes in program requirements. Also, because of the expense of the system, companies have had to procure a system which was capable of being shared across a number of flight test programs and also satisfy various engineering data processing requirements. It was not cost effective to have a system dedicated to a single flight test program, and at the same time, the overall system cost was increased even further in order to satisfy the diverse requirements of all of the users of the system. The net result was the procurement of a large, multi-million dollar system (i.e., an IBM or CDC mainframe).

The introduction of microcomputers with high data transfer and internal processing speeds offers a new approach. It is now economically feasible to acquire a complete computer system, tailored specifically to flight test requirements, to support a single flight test program.

System Description

The Ground Data Center (GDC) is a computer based electronic system for recovering flight test data which has been recorded on magnetic tape. The data may be displayed in graphic form on a computer Cathode Ray Tube (CRT) terminal, plotted graphically on paper, or printed in a numeric format. Figure 1 shows the major components of the GDC and how they are organized within the system.

The flight test data are recorded in flight on magnetic tape using Pulse Code Modulation (PCM) techniques. This is a means of sampling, digitizing, and outputting the analog flight test data as a serial stream of digital data. The airborne instrumentation equipment has the capability of sampling and recording up to 54 separate flight test parameters at a rate of 100 times per second. Each parameter is encoded using 10 binary digits (bits) which provides a resolution of 1024 discrete levels for full range. The 10 bits for each parameter taken together are referred to as "words". All 54 words (one for each parameter) are grouped together as "frames". The data is recorded serially on magnetic tape using a Bi-Phase Level (Bi-0-L) coding scheme. When a flight test maneuver is in progress, 57,000 bits of data are recorded each second (54 parameters and 3 timing and alignment words x 10 bits x 100 per second).

The recorded data is played back on the GDC instrumentation tape recorder and decoded using a bit sychronizer and format sychronizer (decommutator). The bit sychronizer reconstructs a clean digital signal from the recorded data, eliminating any noise or distortion introduced during the recording or playback processes. It also extracts a digital timing/alignment signal from the incoming data. The

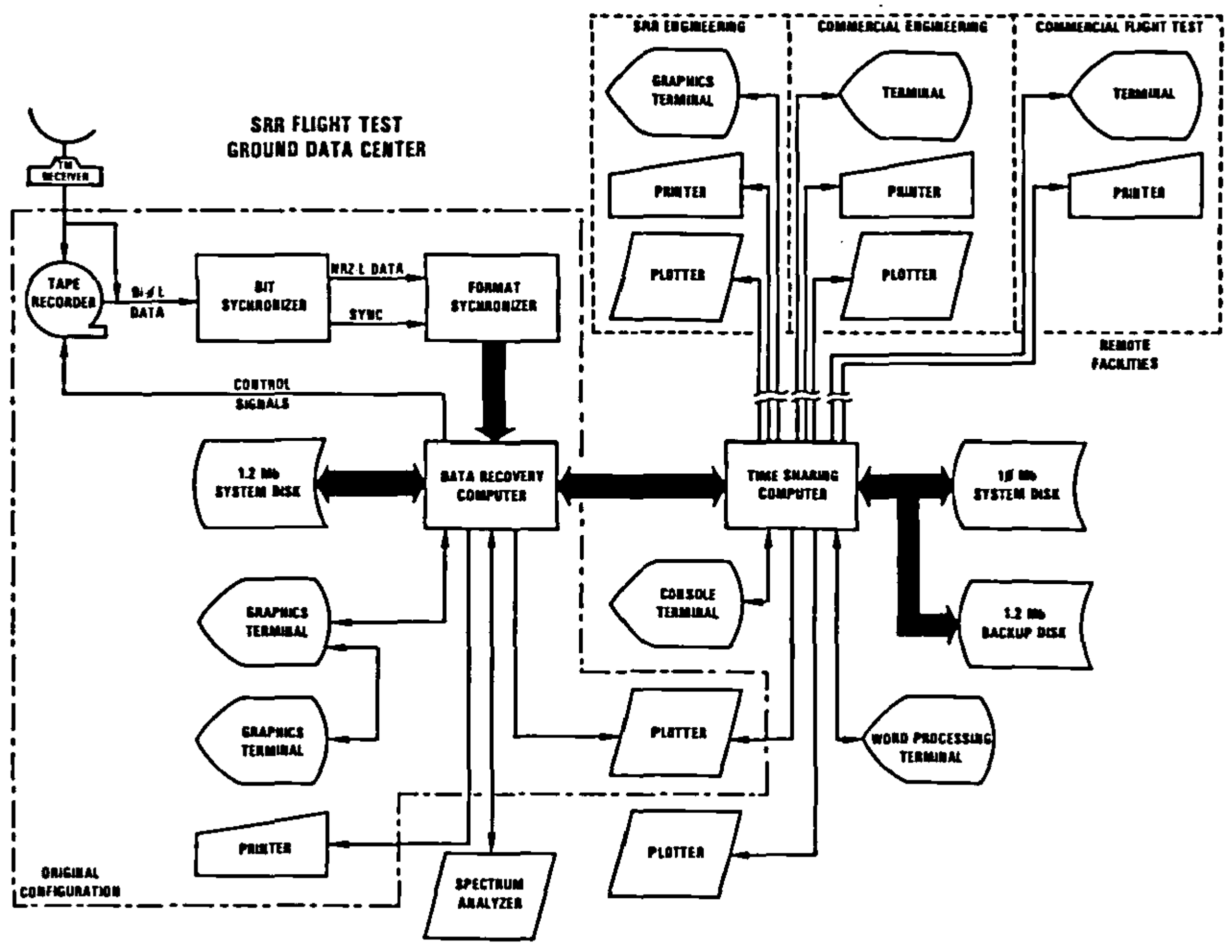

Figure 1

reconstructed data and timing signals are then received by the format sychronizer which divides the incoming bit stream into the original 10 bit words (a process referred to as "decommutation"). These data words are sychronized with a code word which signals the beginning of each frame and then transferred to the computer through a parallel Direct Memory Access (DMA) interface. This allows the computer to perform processing tasks concurrent with the transfer of new data from tape. Once the raw flight data is in the computer, the GDC software provides the data analyst with a great deal of flexibility in the selection and presentation of that data.

System Operation

The data analyst controls the operation of the GDC with the use of "menu" and fixed screen format entries through the keyboard associated with the CRT graphics display. Menus are used for selection of functions and subfunctions, and fixed screen formats are used for data entry. A menu is a list of alternative courses of action displayed on the CRT, each with an associated number or letter (Figure 2). The data analyst simply selects the desired course of action by typing the appropriate number or letter on the keyboard. After the data analyst has selected a function using a menu, it is always possible to terminate the selected function and return to the menu. This is useful when a mistaken keyboard input results in initiation of other than the intended function. The use of menus thus relieves the data analyst of the need to remember any codes or control sequences to perform the various system functions.

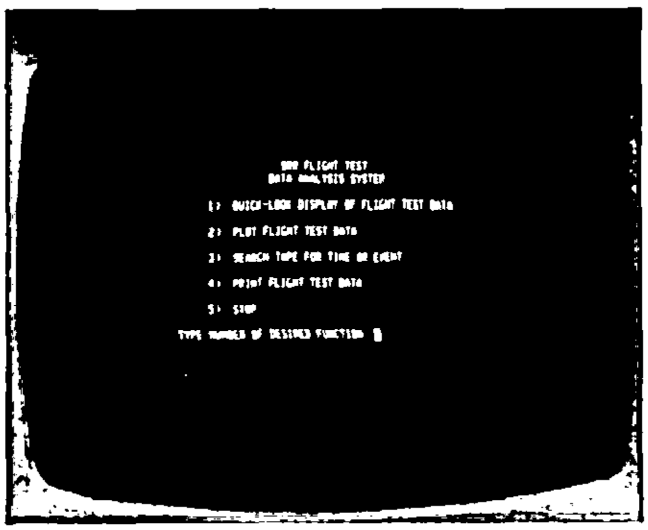

Figure 2 - Menu Display

Entry of setup information is facilitated in the GDC by the use of a fixed screen format data entry technique. This advanced technique is characterized by the use of a "questionnaire" style page displayed on the CRT with dashed lines for the answers or responses (Figure 3). The CRT cursor is positioned at the first response line and advances to each response line in order as the data analyst enters the information. If an incorrect or invalid entry is made, the cursor will not advance until it is corrected. It is possible for the data analyst to move back and forth, from field to field, as necessary until the desired information has been completely entered. Because the computer automatically positions and controls the cursor action, the data analyst is prevented from making entries which are too long, in the wrong place, or in an incorrect format. This allows the data analyst to spend maximum time on analysis tasks free from the need for special skills or requirements to operate the GDC.

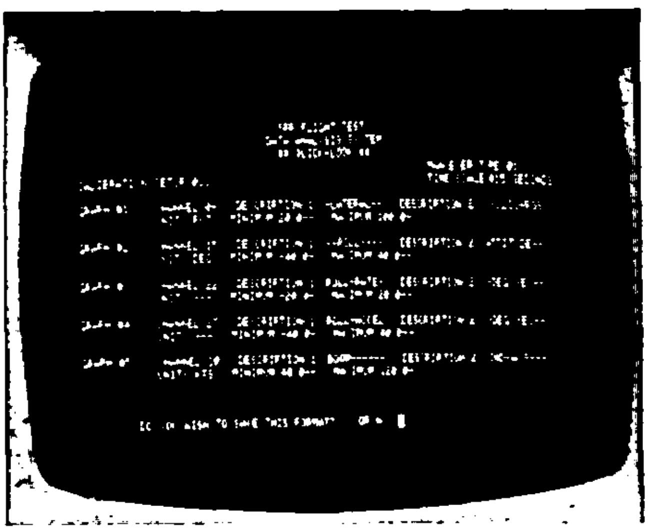

Figure 3 - Fixed Screen Format

The instrumentation tape recorder has a special purpose interface to the computer enabling the computer to initiate tape deck control functions directly without intervention by the data analyst. Once a tape has been mounted on the tape recorder, it is not necessary for the data analyst to have any further direct contact with the tape recorder until it is time to remove the tape. All tape control functions are initiated automatically by the computer in response to the functions being performed by the data analyst.

The basic philosophy underlying the entire system operation is simplicity and ease of use. It is always possible to correct mistakes easily or back up to previous steps from any point. The system keeps the data analyst informed of the exact system status and what the options are at all times.

System Functional Capabilities

The GDC has been designed to provide precisely the sorts of functions which maximize the productivity of the data analyst in performing analysis tasks. Each of the salient features of the data analysis process has been isolated and a corresponding function provided in the GDC for its support. The major functions performed by the data analyst in reducing flight test data are: 1) Parameter Quick Look, 2) Time History Plot, 3) Tabular Data Print, 4) Parameter Cross Plot, 5) Tape Search, and 6) Vibration Analysis.

Parameter Quick Look

Following a flight, the flight test engineer and data analyst are interested in reviewing the results of particular tests and manuevers conducted during that flight. It is sometimes necessary to confirm that certain test parameters are within the required tolerances as soon as possible after a flight to determine if the tests have to be flown again. To facilitate an immediate review of the parameter data and to aid the data analyst in scaling and formatting parameter data for plotting, the GDC can present a "Quick Look" of the parameter data in graphic form on the CRT in a psuedo-real-time, simulated stripchart display (Figure 4) as the tape is played back. The system is fast enough to present a real-time display of flight test data during flight and has been configured for the addition of telemetry equipment in the future if such a requirement should arise.

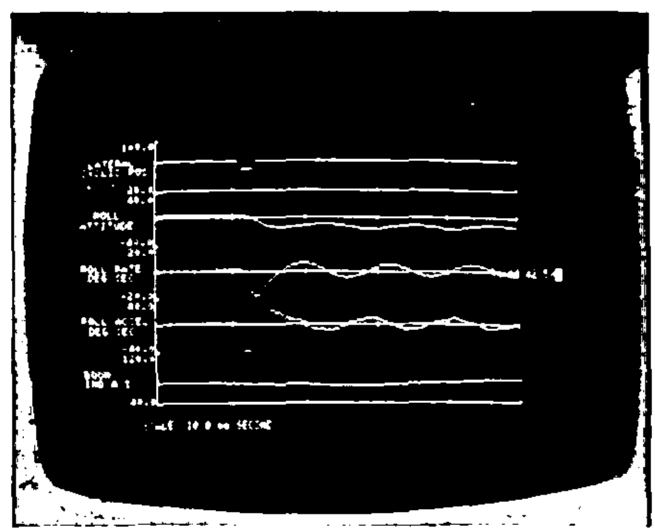

Figure 4 - Quick Look Display

Any five of the 54 recorded parameters may be displayed simultaneously during Quick Look. The data analyst sets up the display by selecting the five parameters, the ranges of values for their display, and the time scale using a fixed screen data entry page. This setup information may then be stored for future use, obviating the need to reenter it each time a particular set of parameters is to be displayed. The system is capable of storing up to 99 different setups. The range of values for display of a parameter is selected by specifying the minimum and maximum values in the engineering units of the parameter. All of the conversions and scaling are performed automatically by the GDC to display the data within the specified range.

Time scales from 1 to 270 seconds may be selected by the data analyst. When the CRT is filled, the cursor returns to the left side of the screen and begins replacing the old data with new. In this way, the data displayed is from the most recent period for the time scale selected. For example, on the 20-second time scale, the most recent 20 seconds of data would always be on the screen. The data analyst has the option of continuing the scan, stopping, or reversing the scan.

The sensors used onboard the aircraft to monitor the various parameters do not always produce a linear output with respect to the parameter values they are monitoring. To produce data which correctly portrays the parameters being measured, it is necessary to compensate for these non-linearities prior to presenting the data. The output of each of the sensors onboard the aircraft is measured at a number of different points throughout the parameter's range following sensor installation. This calibration data is then stored by the GDC and used to compensate the recorded data prior to display to the data analyst. The analysts and engineers are always presented with properly scaled, fully compensated data.

After reviewing a flight using the GDC Quick Look mode, it is necessary to reproduce selected tests and manuevers in hardcopy form for inclusion in the flight test report. The GDC directly supports the preparation of the flight test report by reproduction of flight test data, both graphically and in tabular form. Graphic presentation of the flight test data is performed using the system plotter. Tabular data and text generation utilizes the system printer which is capable of producing high quality solid character, proportionally spaced printed output, both left and right justified. The computer allows the data analyst to enter all of the headings, annotations, and legends to be placed on the plots and listings using the fixed screen format data entry technique.

<u>Time History Plot</u>

After the data analyst has used Quick Look to isolate a particular manuever and adjust the time scale and parameter display ranges to most usefully present the data, the next step is to plot the data in a "Time History" format. A Time History Plot (Figure 5) is simply a facsimile of the parameter data displayed on the graphics terminal during Quick Look. Headings and

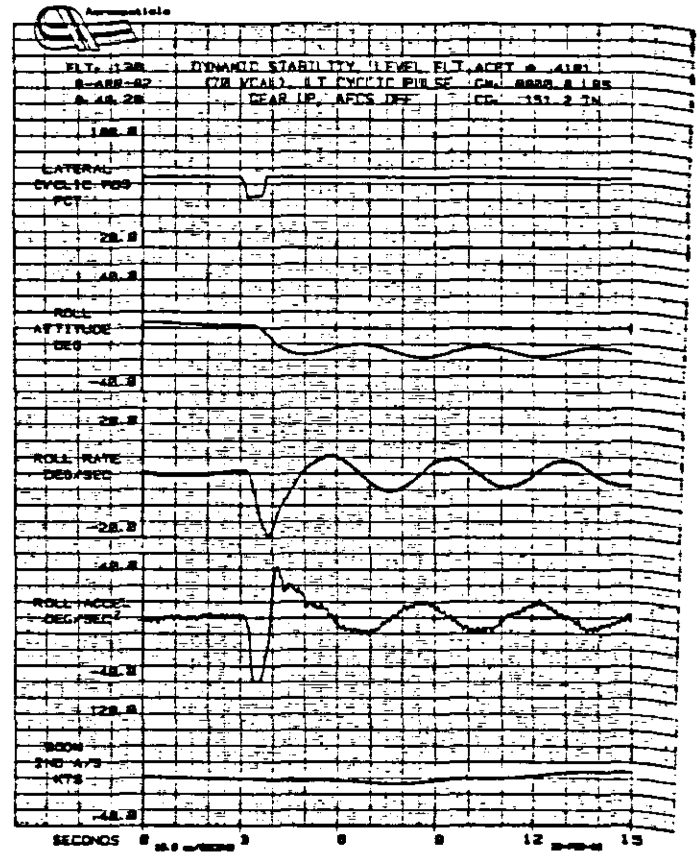

Figure 5 - Time History Plot

other annotations are added to the plot for documentation purposes and the plot is produced using a much higher resolution than is achievable during Quick Look on the graphics terminal. A high speed pen plotter is used with carefully aligned millimeter-grid graph paper to maximize the readability of the plot.

<u>Tabular Data Print</u>

If the precise values of the parameters through a particular test are of more interest to the flight test engineer than the overall time response inter-relationships of the parameters, the GDC can be used to print the actual values of the parameters at each sampled point. At a rate of 100 frames per second of recorded flight, it is easy to see the potential volume of the printed reproduction of even a short segment. However, for a very detailed look at a small segment of time, the printed listing of parameter data produces the most accurate level of detail possible. For a sort of "tabular Quick Look", it is possible for the data analyst to have the tabular data directed to the CRT display instead of the system printer (Figure 6). In this way, the data analyst may scan the actual data in much shorter time and with less waste of paper. As in Quick Look and Time History Plots, the data presented is in engineering units, compensated with the proper calibration corrections.

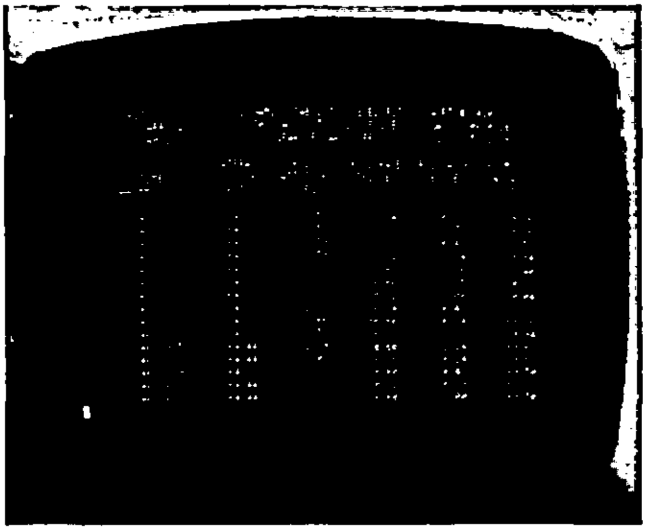

Figure 6 - Tabular Data Display

Parameter Cross Plot

Parameter Cross Plots enable the data analyst to plot parameter data as a function of another parameter instead of as a function of time. Two different forms of Cross Plot are produced by the GDC for flight test data reduction: 1) Calibration Cross Plots and 2) Stability Cross Plots. Calibration Cross Plots (Figure 7) are used to present the relationship between onboard sensor output and the measured parameter values. These are the calibrations which are used during data recovery to correct

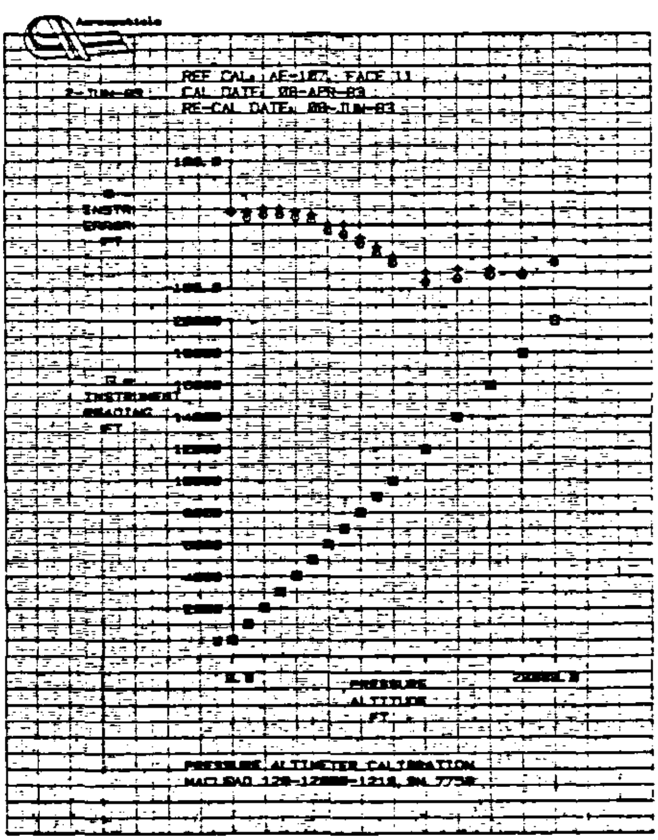

Figure 7 - Calibration Cross Plot

the raw sensor data. Stability Cross Plots (Figure 8) are used to show the static stability characteristics of a test aircraft. These characteristics involve the relative control positions and attitudes of the aircraft throughout various aircraft configurations and airspeeds.

Data to be plotted on a cross plot can be selected dynamically from the parameter data recorded on tape or entered manually through the graphics terminal. When selecting cross plot data dynamically, the data analyst uses a data display mode very similar to the Quick Look display to scan for the desired data (Figure 9). Any time the data analyst stops the scan, the numeric values of each of the displayed parameters at the current cursor position is presented on the CRT for the analyst to review and/or select for plotting. When all of the desired data points have been selected, the data is directed to the plotter. The data analyst also has the option of manually entering the parameter values through the graphics terminal (Figure 10) and then plotting this data in one of the two Cross Plot formats.

Tape Search

The GDC enables the data analyst to search a tape for a particular time or event and also to begin automatic data recovery at a specific time. Time code is recorded on the tape during flight in the same manner as parameter data and is thus accessible during data recovery.

To search for a particular time location, i.e., the start of a maneuver, the data analyst simply enters the time at which the manuever was initiated. The GDC then begins the search process. Since the time code is stored like parameter data, the tape recorder must be in playback mode for the GDC to read the time code. Also complicating matters is the fact that the tape recorder is not run continuously from startup to shutdown, but only during the actual flight test maneuvers. Therefore, it is necessary for the computer to compare the desired location with the current location, move a computed distance toward it, then stop, and read the new current location. If the new location is the desired location, the tape stays where it is. If it is not, the process continues until the desired location is found. The actual scanning algorithm used by the computer is an optimized, binary search routine which locates the desired position in the least number of steps. The important feature of this system is that the computer performs these searches completely autonomously.

Automatic data recovery can also be performed by the GDC, enabling the data analyst to get a segment of parameter data beginning precisely at a specified time. This is used when it is necessary to plot more than five parameters for a particular manuever. Plotting groups of five

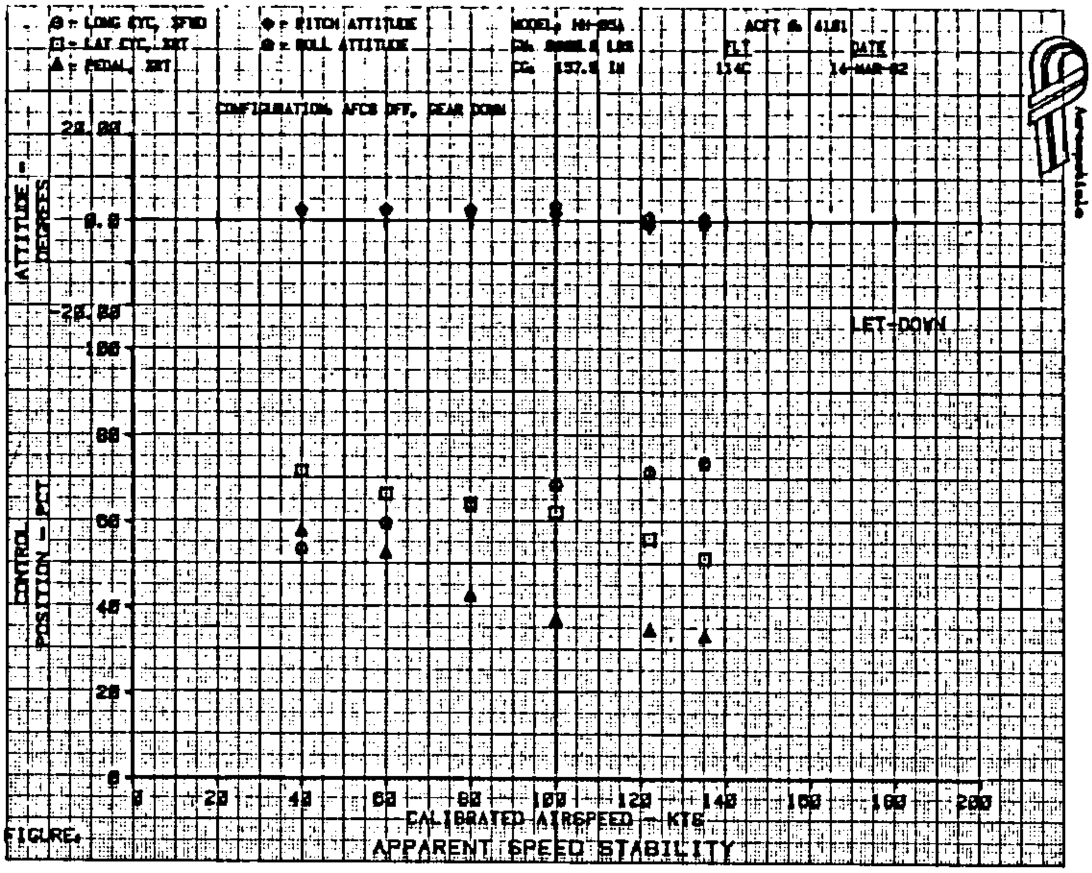

Figure 8 - Stability Cross Plot

parameters at a time, all sychronized to exactly the same time code is performed automatically after the data analyst specifies the desired parameters and the time code for the segment start.

Vibration Analysis

Vibration analysis is used extensively in helicopter testing to identify and isolate problems in the many rotating components which make up an aircraft. Vibration sensing on the aircraft is performed by special accelerometers which output an electrical signal recorded using frequency modulation (FM). This vibration data is recorded on different tracks of the same tape recorder used to record the other flight parameters. During playback, the vibration signals are fed into a spectrum analyzer which is controlled automatically by the GDC. After a frequency/amplitude distribution has been isolated for a particular flight segment, it can be

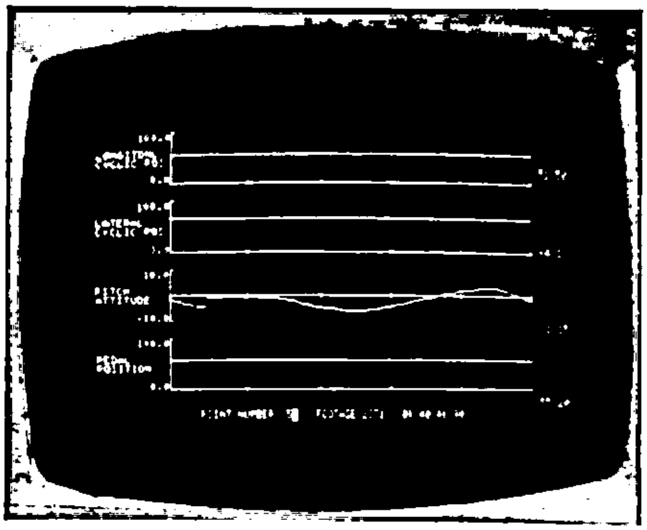

Figure 9 - Point Selection Display

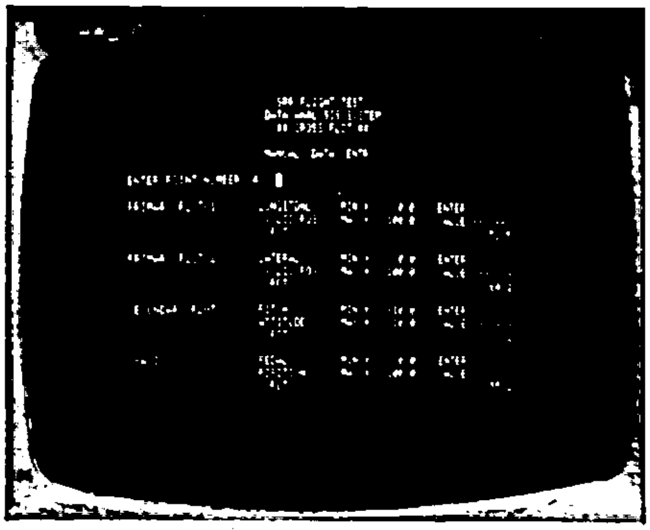

Figure 10 - Manual Point Entry

plotted on the system plotter (Figure 11).

Output Spooling

As flight test activities grew and data reporting requirements burgeoned beyond the capacity of the "original" GDC (Figure 1), AHC began looking at alternatives for increasing their data production capabilities. Although it takes a data analyst about four minutes to recover a particular flight segment and prepare it for plotting, the actual plotting process takes more than seven minutes. This enabled the data analyst to produce only four to five plots per hour. To take advantage of the free time the data analyst has while the plotter is operating, a second computer system was added to off-load the plotting task from the data recovery computer. When a data segment is ready for plotting, the data recovery computer transfers the data to the second computer which then drives the plotter. This addition almost doubled our plotting capacity to nine plots per hour. A second plotter has recently been incorporated in the system, bringing the plotting rate up to over 17 plots per hour.

Timesharing System

A timesharing operating system was installed on the second computer permitting utilization of the GDC for a number of auxilliary computing activities by the flight test department and several other departments, as well, concurrent with the data plotting functions. These activities include engineering computing, weight and balance computing, maintenance tracking, and word processing. The overall system still has significant capacity for further growth when additional requirements arise.

Weight and Balance Computing

The GDC has been designed to provide weight and balance computation capabilities to meet some of the unique requirements of flight test activities. During aircraft handling qualities testing, it is necessary to configure an aircraft for very specific weight and center of gravity (cg) values by careful loading of fuel, ballast, and test personnel. As fuel quantity decreases during a flight, it is also necessary to know what effect this is having on the aircraft weight and cg.

The GDC enables the flight test engineer to enter the aircraft number, crew loading, minimum acceptable fuel load, and the desired aircraft weight and cg (Figure 12). The weight and balance system then manipulates the fuel and/or ballast loading to achieve the desired weight and cg. The ballast stations and their respective capacities are stored by the GDC, as well as crew weights and the empty weight and cg data for each aircraft. After computing the aircraft configuration, the ballast and

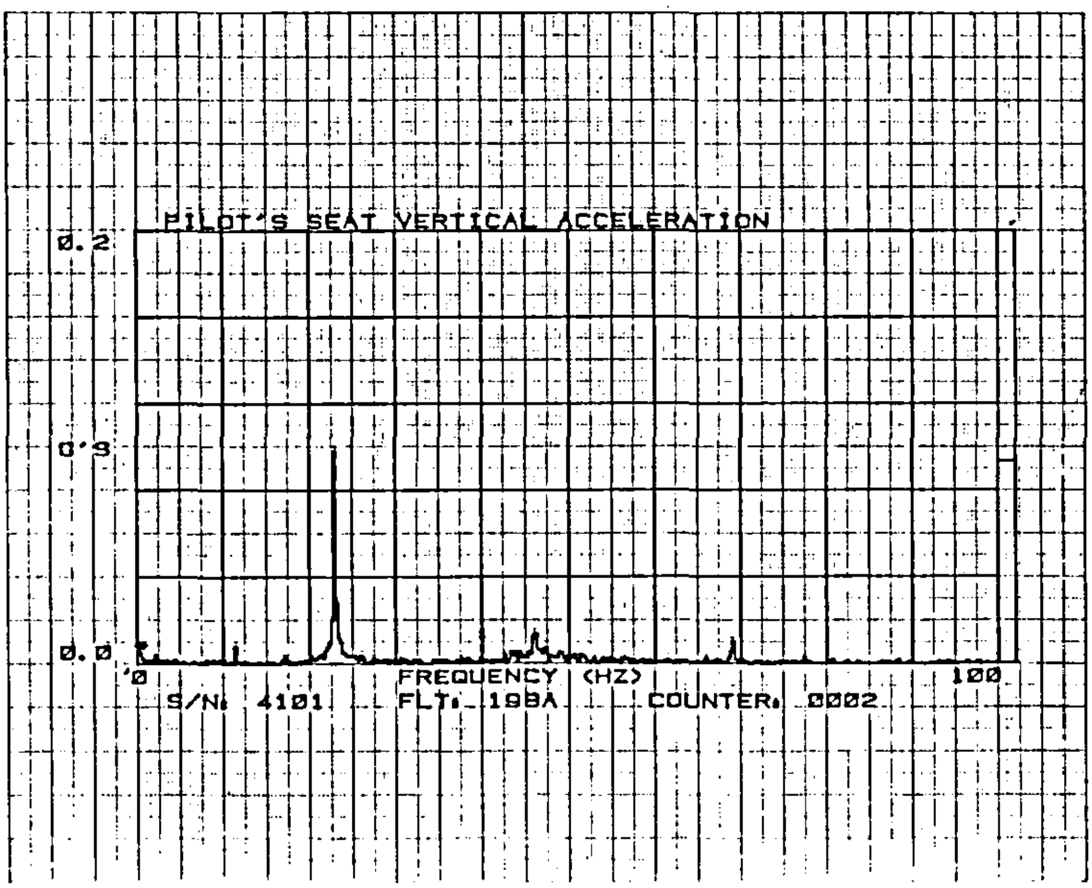

Figure 11 - Vibration Data Plot

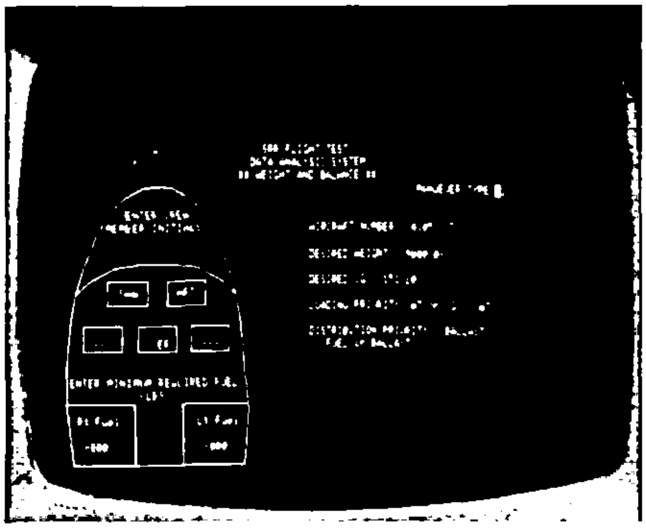

Figure 12 - Aircraft Configuration Entry

fuel loading data are presented to the flight test engineer (Figure 13). The GDC also plots the aircraft weight and cg beginning with the configured fuel load and ending with zero fuel (Figure 14).

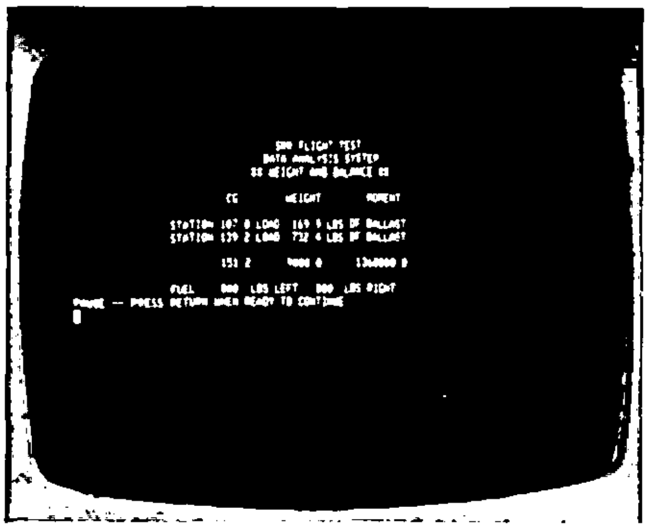

Figure 13 - Aircraft Loading Display

Maintenance Tracking

To assist in scheduling and tracking maintenance activities on the aircraft, the GDC incorporates a maintenance tracking system. All of the required inspections and maintenance tasks for each aircraft are stored, along with descriptions of the tasks and the intervals at which they must be performed. The flight time for each aircraft is logged on the GDC, and when the interval for a particular maintenance task is reached, the maintenance supervisor is notified that the task must be performed. The maintenance supervisor may also display a list of all of the maintenance tasks which will be required within a specified time period, including a brief description of the tasks and the dates they were last performed (Figure 15). The maintenance tracking system is also used to keep track of the time in service of many of the subsystems and components on the aircraft.

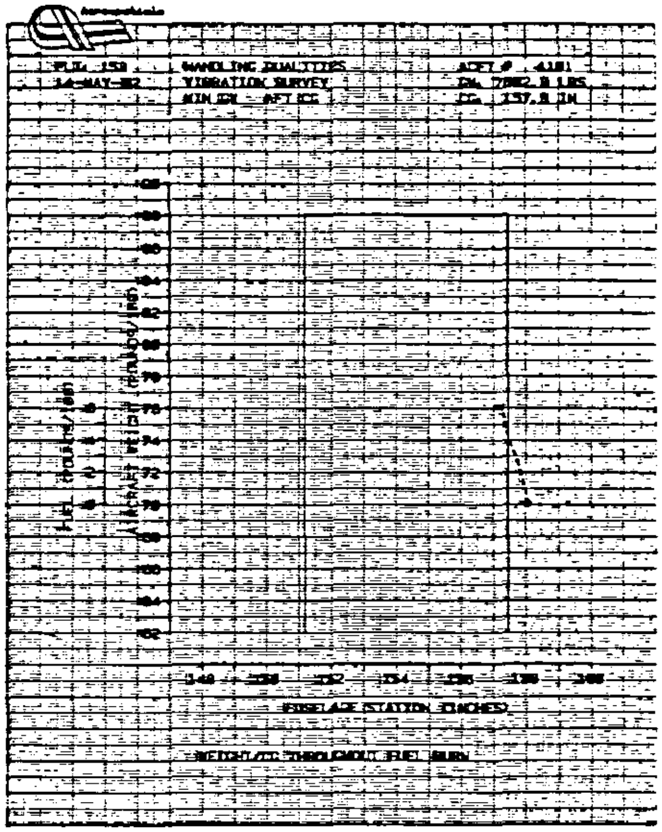

Figure 14 - Fuel Burn Plot

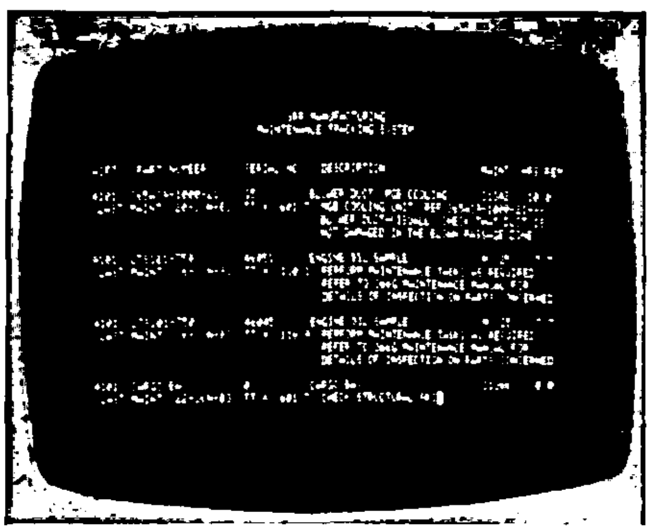

Figure 15 - Upcoming Maintenance Display

Conclusion

The AHC system offers a number of advantages over the traditional large system approach. First, the flight test department has exclusive use of the system. Post-flight data reduction can be performed immediately without delays from other programs or departments. Second, the system is easily expanded in the event of increased data monitoring requirements by the simple addition of processors in parallel with the current data recovery computer. This provides essentially unlimited expansion with uniform cost increases directly proportional to the increased performance. This is in contrast to large systems in which performance may easily be increased to a point beyond which further increases, however small, produce major cost increases.

A third advantage of the present system configuration is the ease with which it can be converted for applications requiring telemetry (TM) of data to the GDC while the aircraft is in flight. The airborne tape recorder may be supplemented by a TM transmitter, and a TM receiver arranged in parallel with the GDC tape recorder. This would provide recording of all 54 data channels while providing continuous real-time display of any five. With the TM equipment on hand, conversion between TM and tape-based modes would take less than five minutes.

The fourth, and probably biggest, advantage of AHC's approach is the initial cost of the system. A traditional data reduction system capable of satisfying the AHC flight test requirements would cost (on a purchase basis) several orders of magnitude more than the microcomputer system. Yet, the microcomputer system provides comparable performance in the computational areas which are critical to AHC's flight test requirements. AHC's computer application is characterized by a small amount of internal processing which must be performed on a large amount of data. Therefore, the data transfer rate of the computer is critical, while internal processing ("number crunching") is of secondary importance. The microcomputer selected by AHC has only about one-third to one-half the internal processing speed of a large system, but it actually has a higher data input/output (I/O) bandwidth. This allows AHC to achieve all of its flight test data reduction and analysis requirements for a fraction of the cost of traditional systems.

NOSEBOOM POSITION ERROR PREDICTION DATA BASE UPDATE

by
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Abstract

A data base update has been made to a NACA-developed algorithm for the prediction of noseboom static pressure position error. The original NACA study had used data obtained from small models; the update used data from actual flight test aircraft/noseboom installations. In addition, the algorithm was extended to include not only subsonic and transonic data, but supersonic data as well. Software was developed which allows easy data base creation, modification and enlargement. Additional software was written to predict position error curves for given aircraft/noseboom installations. The accuracy of the new data base and algorithm was confirmed by the use of an independent test case.

Nomenclature

- A_X - Maximum fuselage cross sectional area, including inlets but not wings or tail surfaces (ft^2)
- D - Diameter of fuselage at maximum cross sectional area (inches)

$$D = (2*12)\sqrt{A_X/\pi}$$

- L/2 - Distance from the aircraft nose apex to the point of maximum fuselage cross sectional area (inches).
- M' - Indicated Mach number.
- Ps - Corrected noseboom static pressure (PSF).
- Ps' - Indicated noseboom static pressure (PSF).
- Qc' - Indicated noseboom impact pressure (PSF).
- X - Distance from the aircraft nose apex to the noseboom static taps (inches).
- ΔP - Static pressure difference (PSF).

$$\Delta P = Ps - Ps'$$

Introduction

Flight test noseboom installations are used to obtain accurate total and static pressure measurements. No total pressure position error corrections are required. Static pressure position error corrections are necessary for the entire aircraft speed range. Typically, the noseboom static source is calibrated either by the tower fly-by technique or by use of a trailing cone static source. However, before the static pressure position error calibration can be conducted, a prediction of the position error is often helpful.

A NACA algorithm for noseboom static pressure position error prediction has existed since 1949.[1] A more complete summary of the algorithm and methodology was presented in 1955.[2] When applied to AV-8B and F/A-18 noseboom installations, the NACA algorithm and data base produced rather poor predictions. Figure 1 shows the prediction error for the AV-8B. The original NACA study had used data obtained from several small models which had been mounted on the wing of a fighter aircraft. In order to improve the accuracy of future position error predictions, a new data base was developed using data obtained from actual noseboom installations on various fighter aircraft. In addition, software which allows additions to the data base and predicts the position error correction was developed for a Hewlett-Packard desk top computer.

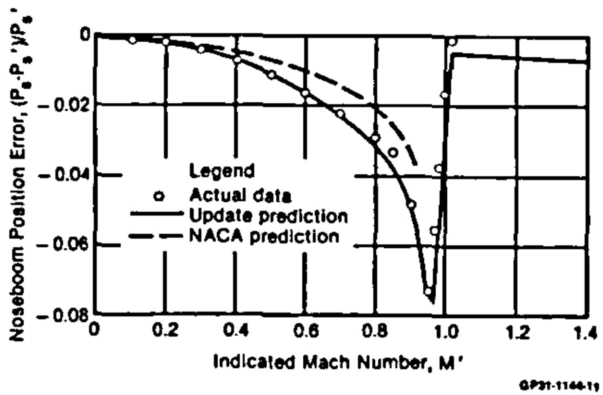


FIGURE 1. PREDICTED AND ACTUAL NOSEBOOM POSITION ERROR AV-8B HARRIER II

Development of the new position error prediction data base was accomplished using the following steps:

- Understanding the NACA algorithm
- Data base update
- Software development and use
- Test case and results.

The NACA Algorithm

The static pressure position error prediction algorithm is divided into three discrete Mach number ranges: (1) subsonic ($0 < M' < 0.9$), (2) transonic ($0.9 < M' < 1.05$), and (3) supersonic ($1.05 < M' < 2.0$). In addition, the algorithm uses several aircraft length parameters. The distance from the apex of the aircraft nose to the noseboom static taps is known as X. The maximum fuselage cross sectional area, A_X, is calculated using fuselage and intake areas but not wing or stabilator areas. Fuselage diameter, D, is calculated from A_X using the formula for the area of the circle:

$$D = 24\sqrt{A_X/\pi}$$

The distance from the apex of the aircraft nose to the point of maximum fuselage cross sectional area is L/2. Typically, L denotes aircraft length. However, the NACA study found that the position error was particularly sensitive to the shape of the forward fuselage. Therefore, the fuselage length parameter was altered to reflect this sensitivity. Figure 2 shows the algorithm parameters applied to a body of revolution as used in the NACA investigation. Figure 3 shows the algorithm parameters applied to a typical fighter aircraft/noseboom installation.

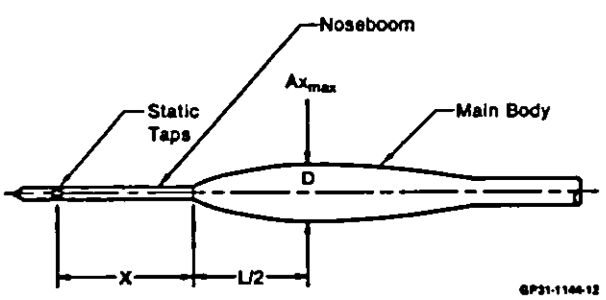


FIGURE 2. BODY OF REVOLUTION (NACA RM L9C25)

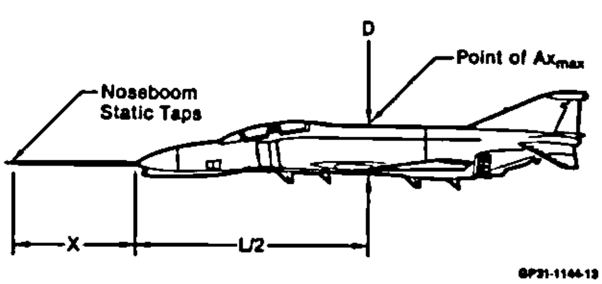


FIGURE 3. PREDICTION PARAMETERS APPLIED TO ACTUAL AIRCRAFT

Subsonic Prediction Algorithm

The subsonic prediction algorithm is set up as a simple two-dimensional table of $(Ps-Ps')/Qc(L/D)^2$ versus X/L. The NACA algorithm expresses position error correction as (Ps-Ps')/Qc' as opposed to the method used by McDonnell Aircraft which expresses it as (Ps-Ps')/Ps'. The NACA study found that the value of (Ps-Ps')/Qc' remains essentially constant subsonically for a given aircraft/noseboom configuration. Two nondimensional parameters are used to adjust the value of (Ps-Ps')/Qc'. The first is fineness ratio, L/D. The second is the ratio of the length of from the apex of the nose to the noseboom static taps and the length from the apex of the nose to the cross sectional area, X/L. For given indicated Mach number, M', and aircraft geometry, a value of (Ps-Ps')/Qc' is found. A simple conversion is then used to obtain the value of (Ps-Ps')/Ps' for the datum point. This will be explained in more detail below. Figure 4 shows the data for the subsonic data base.

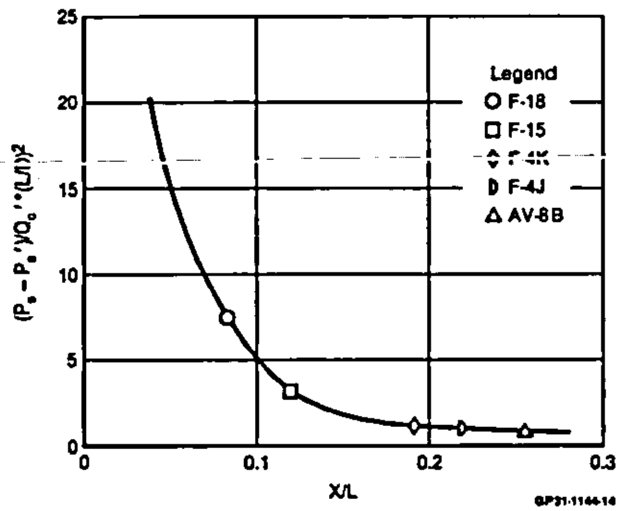


FIGURE 4. SUBSONIC DATA BASE

Transonic Prediction Algorithm

The transonic region presents a greater prediction challenge due to the large pressure variations associated with shock wave formation. In addition to the two parameters used in the subsonic data base (namely, (Ps-Ps')/Qc' $(L/D)^2$ and X/L), the transonic data base uses a third parameter, $(M'-1)(L/D)^2$. This $(M'-1)(L/D)^2$ parameter is used to account for the shock formation and movement past the noseboom static taps. Figure 5 gives the transonic data base.

Supersonic Prediction

The NACA prediction algorithm did not extend into the supersonic range. Therefore, a rough approximation was developed as part of the data base update.

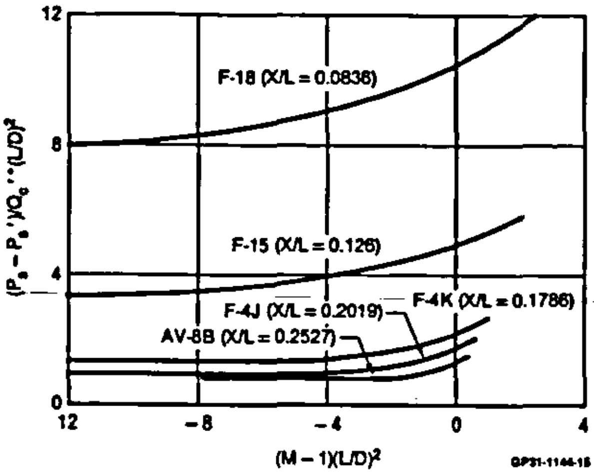


FIGURE 5. TRANSONIC DATA BASE

Data Base Update

The new noseboom static pressure position error prediction data base was developed using actual data from five fighter aircraft: the F/A-18, F-15, F-4K, F-4J and AV-8B. The NACA algorithm has been developed using position error expressed as (Ps-Ps')/Qc' versus M'. However, position error data obtained from the actual aircraft was in the form of (Ps-Ps')/Ps' versus M'. Therefore, a conversion from the (Ps-Ps')/Ps' form to the (Ps-Ps')/Qc' form was necessary in order to develop the data base. In addition, a conversion in the reverse order was necessary in order to apply the prediction data to noseboom installations on McDonnell-built aircraft.

The data conversion was conducted as follows: Mach number M' is defined as a function of Qc' and Ps'.

$$M' = \left[5\left(\left(\frac{Qc'}{Ps'}+1\right)^{0.2857143}-1\right)\right]^{\frac{1}{2}} \qquad [1]$$

Solving for Qc':

$$Qc' = Ps'\ [(1+.2M^2)^{3.5}-1] \qquad [2]$$

If (Ps-Ps') is defined as ΔP, then

$$\Delta P = Ps-Ps' \qquad [3]$$

$$(Ps-Ps')/Ps' = [\Delta P/Ps']$$

$$(Ps-Ps') = \left(\frac{\Delta P}{Ps'}\right)\ Ps' \qquad [4]$$

Finally, solving for (Ps-Ps')/Qc':

$$\frac{(Ps-Ps')}{Qc'} = \frac{\left(\frac{\Delta P}{Ps'}\right)\ Ps'}{Ps'[(1+.2M^2)^{3.5}-1]} \qquad [5]$$

$$\frac{(Ps-Ps')}{Qc'} = \frac{\left(\frac{\Delta P}{Ps'}\right)}{[(1+.2M^2)^{3.5}-1]} \qquad [6]$$

Equation [6] allows (Ps-Ps')/Qc' to be calculated as a function of the two parameters that are known for any given datum point, namely [ΔPs/Ps'] and M'. This equation may be rewritten to allow a conversion from (Ps-Ps')/Qc' to (Ps-Ps')/Ps' as well:

$$\left(\frac{\Delta P}{Ps'}\right) = \left[\frac{(Ps-Ps')}{Qc'}\right]\left[(1+.2M^2)^{3.5}-1\right] \qquad [7]$$

With the problem of data conversion solved, the data base was easily updated in the subsonic and transonic regions using the NACA algorithms. Aircraft geometry and position error data were put into the form of the NACA algorithms to create the new data base. However, because an algorithm for supersonic predictions was not available, one had to be developed. An examination of the actual flight test data from the four previously mentioned supersonic aircraft* revealed a similarity in the supersonic position error curves. The four supersonic position error curves were input into a second order curve-fit routine, which does not present an excellent position error estimate for any single aircraft, but does give a reasonable prediction of the error for all the aircraft.

Software Development

A Hewlett-Packard desk top computer, equipped with two disk drives and a plotter table, was used to develop both the data base and prediction software. A summary flowchart and explanation is shown in Figure 6.

Data Base Generation

Generation of a new data base was central to the development of accurate position error predictions. Therefore, a major goal of the programming effort was to allow easy data base modification and enlargement. This goal was accomplished through a series of programs which allows input of raw position error data, converts the data into a data base, predicts position error curves for new aircraft noseboom installation, and plots the results.

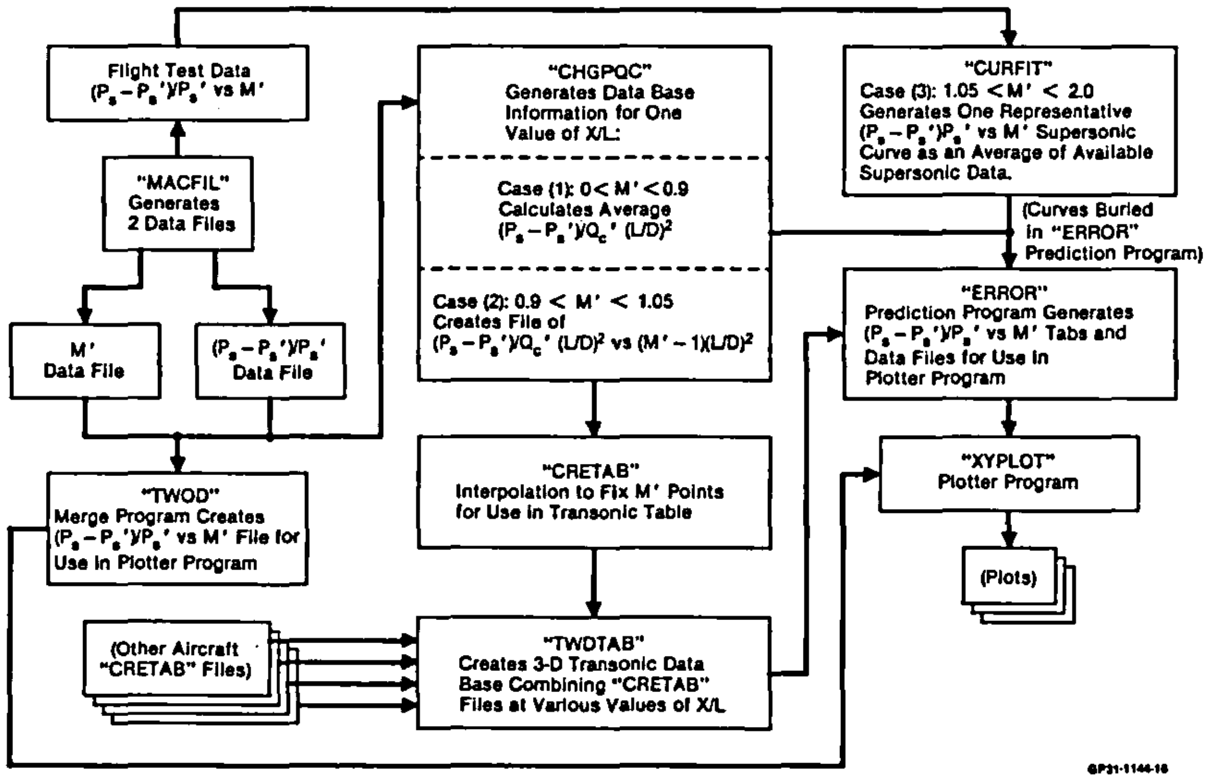


FIGURE 6. NOSEBOOM STATIC PRESSURE POSITION ERROR PREDICTION SOFTWARE DEVELOPMENT AND FLOWCHART

Flight Test data is input into the program "MACFIL" which generates 2 separate data files: the first file is indicated Mach number (M') and the second file is position error ((Ps-Ps')/Ps'). "TWOD" is a merge program which creates a single (Ps-Ps')/Ps' versus M' data file for use in the plotter program. The heart of the data base generation is the program "CHGPQC". This program uses the two "MACFIL" data files and applicable aircraft/noseboom geometry to create subsonic and transonic data base points. After "CHGPQC" has been run for all the aircraft/noseboom combinations to be included in the data base, a summary subsonic data base curve is buried in the prediction program "ERROR". Supersonic data is summarized as a second order curve-fit in the program "CURFIT". This data is also buried in the prediction program "ERROR".

The transonic data base is arranged as a three dimensional table. Therefore, individual transonic data files from the "CHGPQC" data base program are prepared for merging through use of the interpolation program "CRETAB". This program rearranges each "CHGPQC" data file so that a common set of $(M'-1)(L/D)^2$ data points is shared in each field. Once this interpolation has been accomplished, the resulting data files are merged into a single three dimensional table; this is accomplished in the program "TWDTAB". The data base is now complete and predictions may be made.

Noseboom Static Pressure Position Error Predictions

The program "ERROR" is used to predict the position error. It requires only a few simple inputs: the aircraft/noseboom geometry and a "TWDTAB" generated transonic data file. The NACA algorithm for subsonic and transonic conditions is used with the supersonic curve-fit to predict a position error for the given configuration over a Mach range up to M'=2.0. This data is stored in a data file which can later be plotted using the plotter program "XYPLOT".

Results

The data base was developed using five McDonnell aircraft models: the F/A-18, F-15, F-4K, F-4J and AV-8B. To test the validity of the data, a test case was obtained from outside the company. This data was obtained for a noseboom installation on the Northrop F-5E.[3] Figure 7 shows a comparison of the actual and predicted position error for the test case.

The test case data showed very good correlation with the predictions. The transonic area showed very good prediction of the "Mach jump". Both the minimum position error and the Mach number at which it occurs were predicted to be very close to the actual data. This was encouraging given the relative complexity of the transonic algorithm. Typically the predictions fell within 10% of the actual data.

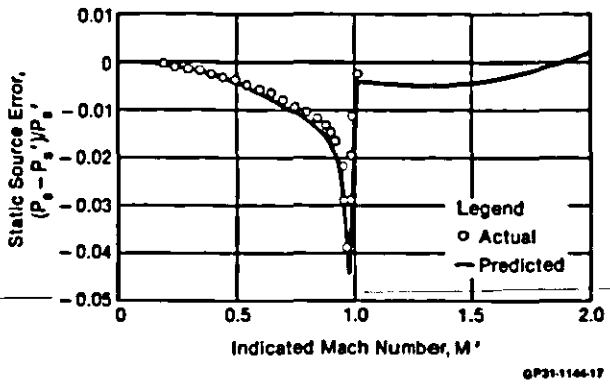


FIGURE 7. NORTHROP F-5E NOSEBOOM PITOT-STATIC CALIBRATION

Conclusions

The noseboom static pressure Position error prediction algorithm and data base was shown to have the following advantages over the old methodology:

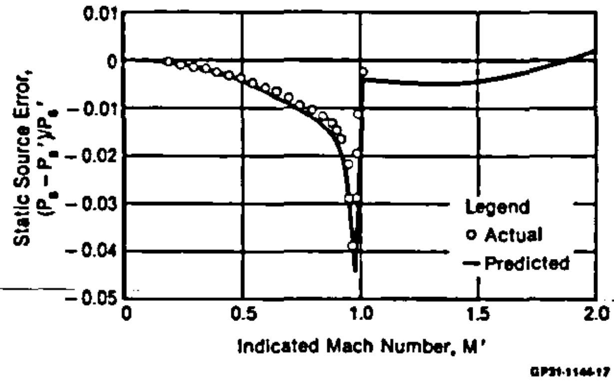
- o More accurate predictions. The F-5E test case showed very good correlation.
- o A data base built upon actual aircraft flight test information, not small models.
- o Easy data base development, modification and expansion.
- o Extension of the prediction algorithm to approximate supersonic data.
- o Easy prediction through user-friendly software.

References

1. Danforth, Edward C.B.; and Johnson, J. Ford; Error in Airspeed Measurement due to Static-Pressure Field Ahead of Sharp-Nose Bodies of Revolution at Transonic Speeds. NACA RM L9C25, August 19, 1949.

2. O'Bryan, Thomas C.; Danforth, Edward C.B.; and Johnson, J. Ford; Error in Airspeed Measurement due to the Static-Pressure Field Ahead of an Airplane at Transonic Speeds. NACA Rept. 1239, 1955.

3. Data obtained from M. Burne at Northrop Corporation.



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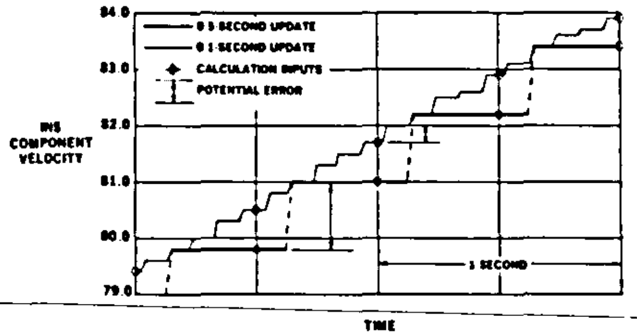

FIGURE 1. UPDATE RATE COMPARISON

Concurrent with the above INS tracking developments, a new procedure was developed for determining the mean flight path wind component during a takeoff or landing performance test run. It was adopted as standard Douglas procedure.[6] This procedure uses airspeed system pitot total pressure (PTO) and space position tracking speed to calculate an indicated instantaneous flight path wind velocity at the aircraft as it travels through the air mass. The mean PTO wind value with standard variation for height is used in takeoff performance calculations.

The stand-alone INS space position tracking procedure was refined and verified using test data from the DC-8 Series 71 aircraft.

STAND-ALONE INS PROCEDURE

The basic elements of the stand-alone INS space position procedure for takeoff performance are given below and summarized in Figure 2. They include items for aircraft instrumentation, test operations, and data calculations.

The aircraft instrumentation items recorded on the Douglas digital data system include: (1) component INS velocities at a 0.1-second update rate; (2) airspeed system total pressure; (3) total air temperatures, and (4) engine thrust data.

AIRCRAFT INSTRUMENTATION:
- COMPONENT INS VELOCITIES
- AIRSPEED SYSTEM TOTAL PRESSURE
- TOTAL AIR TEMPERATURE
- ENGINE THRUST DATA

TEST OPERATION:
- PERIOD OF ZERO SPEED
- INS BAROLOOP OPEN

DATA CALCULATION:
- ADJUST INDICATED INS VELOCITIES
- ADJUST AMBIENT TEMPERATURE
- ADJUST MEAN HEADWIND VELOCITY

FIGURE 2. ELEMENTS OF STAND-ALONE INS PROCEDURE

Test operation requirements included: (1) a defined period of zero speed prior to brake release, and (2) an open baroloop circuit in the INS during the test period.

Data calculations using the X2CC program on the Sigma 9 computer included: (1) adjustment of the indicated INS component ground velocities to provide zero velocity during the zero speed period; (2) adjustment of the INS indicated vertical velocity to provide height values parallel to the runway slope prior to the rotation maneuver in takeoff; (3) adjustment of the runway ambient temperature to provide a calculated total air temperature equal to the measured value at rotation; and (4) adjustment of the headwind to equal the mean value of the calculated PTO wind 5 to 10 seconds before rotation.

The data adjustments require at least two runs by the X2CC computer program to make calculations for each takeoff. The optimum values are unique for each test. The procedures for establishing the adjustment values are discussed in later sections of this paper.

The stand-alone INS procedure for space position tracking offers the following seven advantages:

1. Tests can be made from any runway where the runway slope is known.
2. Improved accuracy of airspeed values for events following rotation.
3. Improved accuracy of instantaneous acceleration data.
4. Reduced time from test to availability of tracking data.
5. Elimination of time correlation problems between the tracking and the onboard data systems.
6. Elimination of test condition limitations imposed by systems outside the aircraft.
7. Elimination of the requirement for an external tracking system.

TAKEOFF COMPUTER PROGRAM

The performance results of takeoff performance tests are calculated using the Douglas X2CC computer programs and the Sigma 9 computer. The X2CC program was developed from the YC-15 takeoff performance computer program.[3] It makes various calculations for analysis of takeoff performance. In addition to processing the INS tracking data, PTO wind measurements, and total temperature calculations, the X2CC program calculates distance corrected for wind, equivalent airspeed, indicated drag coefficient, and indicated rolling coefficient. The design concept of the X2CC program permits any desired calculation to be made with the data available. There are 150 output parameters available for tabulation or plotting. The test data

TAKEOFF PERFORMANCE DATA USING ONBOARD INSTRUMENTATION

by
Harold K. Cheney
Douglas Aircraft Company
McDonnell Douglas Corporation
Long Beach, California

ABSTRACT

A method has been developed to obtain velocity, distance, height, wind, and ambient temperature for takeoff performance calculations using self-contained instrumentation onboard a test aircraft. Space position data are obtained using an inertial navigation system. The headwind at the aircraft is determined using pitot total pressure, tracking altitude, and tracking velocity. Ambient field temperature is determined using the measured total temperature prior to rotation. The only information required from ground systems is the runway slope. The resulting performance data, corrected to zero wind conditions, are equivalent or superior to data obtained using external tracking, wind, and temperature measuring systems. Using the INS space position data, it is practical to calculate indicated instantaneous drag and rolling coefficients.

INTRODUCTION

This paper describes procedures for determining aircraft takeoff performance from flight tests without the use of data measurement systems outside the aircraft. The space ground position data are obtained by a Litton Industries inertial navigation system model LTN-51, with Litton Computer Program No. 101588. By using an INS, the procedure becomes similar to that used by the National Aerospace Laboratory NRL in the Netherlands[1] and by the Lockheed-Georgia Company. The primary features that separate this from previously reported procedures are the methods of determining wind and ambient air temperature. The term "stand-alone INS" is used to identify this procedure.

The stand-alone capability is possible through the use of (1) the modified INS for tracking, (2) magnetic tape for data recording, (3) a computer program for calculating results, (4) a new procedure for determining flight path wind component, and (5) total air temperature for determining ambient temperature. The INS equipment and basic computer equations were initially used at Douglas Aircraft Company during the YC-15 STOL aircraft flight test programs.[2,3] The stand-alone procedures were developed during the Super 80 and DC-8 re-engine test programs. Similar procedures are being developed for landing performance.

The development of the stand-alone INS procedure started in 1975 at Douglas Aircraft Company during the YC-15 STOL aircraft test program and was completed in 1982 during the DC-8 re-engine program. Two LTN-51 inertial navigation systems were obtained for the YC-15 program. These systems were modified by Litton to provide three-axis orthogonal velocities to the aircraft digital data system and other special features.[4] Equations were developed to calculate space position tracking information by integration of the component velocities. These equations were included in the computer programs used for calculating takeoff and landing performance. The output of these computer programs provided the ability to make a comparison of ground system tracking (laser or phototheodolite) with INS tracking data.

During the YC-15 program, it was determined that the INS data for distance, velocity, acceleration, and height compared well with laser and phototheodolite data. The potential for obtaining accurate onboard tracking data for FAA certification tests was recognized. A comparative analysis of INS and laser tracking indicated that the 0.5-second update rate of the component velocities provided by the INS was too slow to provide the accuracy desired for FAA certification data. A modified internal computer program for the LTN-51 units was obtained from Litton.[5]

The modified LTN-51 units now provide a 0.1-second transmission rate for the North-South and East-West velocities, and a 0.2-second transmission rate for vertical velocity. The effect of update rate on data accuracy is shown in Figure 1. It should be noted that the timing of the INS does not match the aircraft data system timing. The expected improvements from the 0.1-second update rate were obtained.

for the figures in this paper were prepared from output tapes of the X2CC program.

INS SPACE POSITION CALCULATIONS

The space position tracking data are calculated from the three-axis orthogonal component velocities using the equations presented in Figure 3.

The component velocity correction values, CV_N, CV_E, and CV_V, are required to adjust the indicated component velocities to true values. The correction values are normally less than 2 knots. They are needed because, during normal INS operations, the instantaneous indicated component velocities wander around the true values.

$$V_N = V_{NI} + CV_N \quad \text{— NORTH-SOUTH VELOCITY}$$

$$V_E = V_{EI} + CV_E \quad \text{— EAST-WEST VELOCITY}$$

$$V_V = V_{VI} + CV_V \quad \text{— VERTICAL VELOCITY}$$

$$V_G = \sqrt{(V_N)^2 + (V_E)^2} \quad \text{— GROUND SPEED}$$

$$V_{FP} = \sqrt{(V_N)^2 + (V_E)^2 + (V_V)^2} \quad \text{— FLIGHT PATH VELOCITY}$$

$$S = \Sigma\text{I-I (INT)}\,\frac{(V_{Gi-1} + V_{Gi})}{2} \quad \text{— GROUND DISTANCE}$$

$$H = \Sigma\text{I-I (INT)}\,\frac{(V_{Vi-1} + V_{Vi})}{2} \quad \text{— HEIGHT}$$

Where:

V_{NI}, V_{EI}, AND V_{VI}	=	INDICATED COMPONENT VELOCITIES
CV_N, CV_E, AND CV_V	=	COMPONENT VELOCITY CORRECTION VALUES
V_N, V_E, AND V_V	=	COMPONENT VELOCITIES
V_G	=	VELOCITY COMPONENT PARALLEL TO GROUND
V_{FP}	=	FLIGHT PATH VELOCITY
S	=	GROUND DISTANCE
H	=	HEIGHT
INT	=	CALCULATION INTERVAL

FIGURE 3. BASIC INS TRACKING EQUATIONS

The wandering of the North-South (V_{NI}) and East-West (V_{EI}) indicated component velocities from true component velocity is due to the inability of the INS to compensate perfectly for the rotational forces on or near the surface of the earth. The indicated velocity errors oscillate continually in an approximate 84.4-minute period. Typical component velocity errors during takeoff and landing operations are shown in Figure 4. These values were obtained while the aircraft was stationary before takeoff or after a landing. The corrections CV_N and CV_E are the values which bring these indicated values to zero. For the space position calculations, the corrections are assumed to be constant during the test run. This is true when the velocity errors are near maximum. As the error values swing from one extreme to the other, they may change as much as 0.2 knot in 60 seconds. This would result in an error of 10 feet in the calculated distance for that period. Figure 5

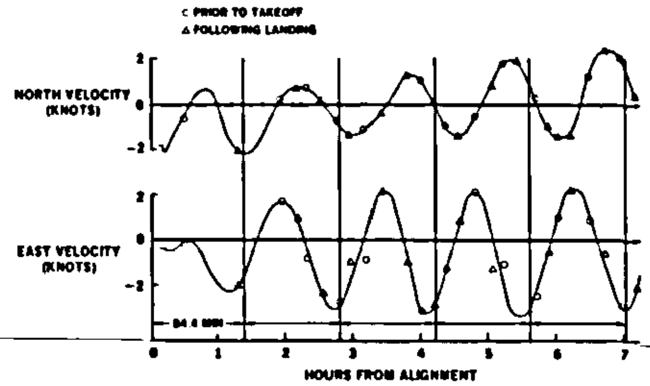

FIGURE 4. TYPICAL INS VELOCITY VALUES AT STATIC CONDITIONS

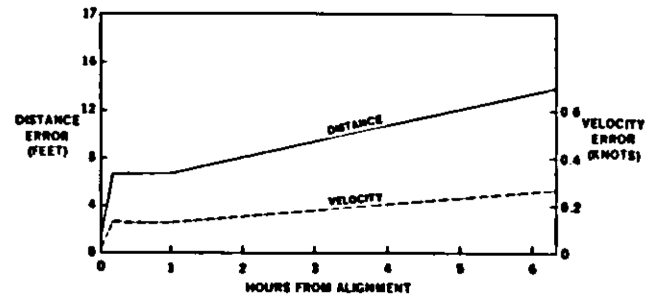

FIGURE 5. LTN-51 INS SPACE POSITION ACCURACY

presents the 1-sigma errors for distance and velocity calculated from basic characteristics of the LTN-51 INS. These errors are not considered significant in the determination of takeoff performance.

In the normal INS configuration, the difference between indicated vertical velocity (V_{VI}) and true vertical velocity varies due to the baroloop feedback circuit. This circuit adjusts INS computed altitude to equal a reference indicated pressure altitude. During the takeoff maneuver, the reference indicated pressure altitude varies due to the changes in static pressure position error. Large changes in static pressure error occur during transition from ground roll to free flight. To eliminate the effect of static position error changes on indicated INS vertical velocity, the test LTN-51 units incorporate a switch to open the baroloop circuit during the test run.

The vertical velocity correction, CV_V, is determined by referencing the calculated INS height to the calculated runway height determined from the known runway slope. A typical initial calculation of height on the runway at Yuma, Arizona, using a CV_V value of 0.00 is shown in Figure 6. Also shown in the figure are radio altitude height and calculated runway height above liftoff, using the runway slope of the liftoff area. The optimum vertical velocity correction is the value which provides a calculated height parallel to the runway prior to rotation. The procedure used to determine CV_V is shown in Figure 7. This determination of CV_V can be made after the initial calculation with the X2CC computer program using tabular data or in graphic form, as

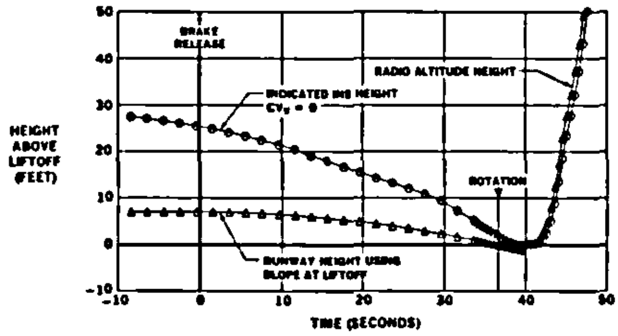

FIGURE 6. INITIAL HEIGHT CALCULATION

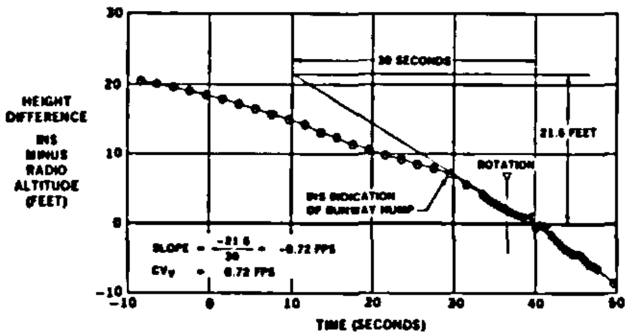

FIGURE 7. DETERMINATION OF VERTICAL VELOCITY CORRECTION

shown in the figure. By using the new CV_T value and the time the aircraft passed over the hump in the runway, one can obtain the results shown in Figure 8. The particular example selected was made with the baroloop closed. The change of height while the aircraft is stationary prior to brake release is an indication of change in INS vertical velocity error between brake release and rotation with the baroloop closed.

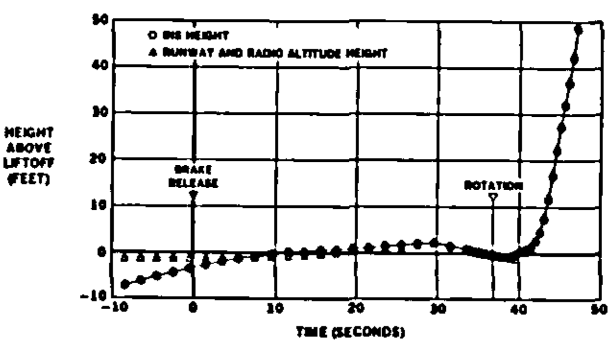

FIGURE 8. FINAL HEIGHT CALCULATION

WIND MEASUREMENT

The PTO wind calculation procedure enables a measured wind value to be obtained at all points along the actual flight path. PTO wind is obtained by subtracting flight path velocity, which is the aircraft speed relative to the ground, from PTO true airspeed. PTO true airspeed is obtained using pitot total pressure from an airspeed system and static pressure determined from altitude at brake release and tracking height. The results of PTO wind measurement for a relatively steady wind are shown in Figure 9. The maximum and minimum values are similar from 20 seconds before brake release until liftoff. For this run, a mean value of 7.5 knots was selected. The mean values after liftoff are calculated using the standard one-seventh power variations with height.

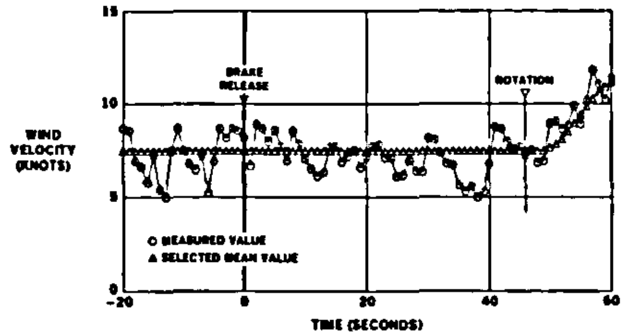

FIGURE 9. TYPICAL TOTAL PRESSURE (PTO) WIND

The equations used, problems in development, and typical results for the PTO wind measurement procedure will be given in a forthcoming Douglas paper.[6] For several years, the PTO wind was included in the takeoff and landing performance test calculations. The values obtained were used only to improve the equations and evaluate their potential application. Final acceptance has been gained because the use of PTO wind values improved the consistency of final performance results.

AMBIENT TEMPERATURE

The ambient temperature for a takeoff performance test has traditionally been obtained using a shaded thermometer along the side of the runway or an asperated total air temperature probe on the aircraft prior to brake release. During the analysis of the DC-8 test activity, an improved procedure was developed in which the ambient temperature is adjusted to a value that provides a calculated total air temperature equal to the measured total temperature during the rotation period in takeoff. A typical example of this procedure is shown in Figure 10. This method accomplishes in a more positive manner the temperature measurement improvement desired from special flight test asperated total air temperature probes.

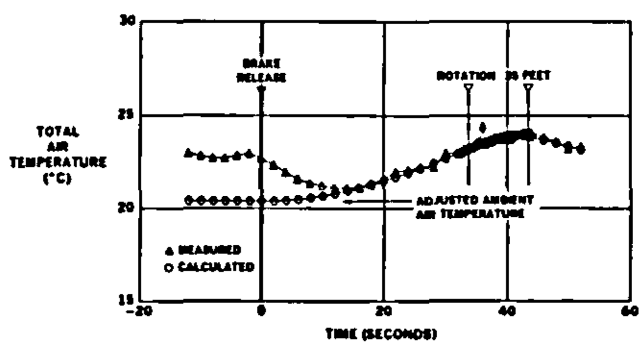

FIGURE 10. AMBIENT AIR TEMPERATURE CONFIRMATION

DISTANCE AND VELOCITY VERIFICATION

The accuracy of the stand-alone INS space position tracking procedure has been verified by comparing stand-alone INS and ground tracking system results. The comparison is possible using data provided by the X2CC computer program for takeoff performance. Results of a detailed comparison of 15 typical DC-8 performance takeoffs were used to confirm the stand-alone INS tracking procedure.

Distance and speed show good correlation. The INS stand-alone values average 0.5-percent less than tracking values obtained from a ground measurement system. The comparisons of acceleration demonstrate excellent agreement of mean values with less scatter of the INS values. Height values from INS and radio altitude systems demonstrate reasonable agreement; i.e., ±2 feet at 35 feet. Phototheodolite height values frequently vary from the mean of the INS and radio altitude values. Due to the use of the PTO wind measurement procedure, excellent agreement (±0.3 knot) exists for airspeed values obtained prior to the rotation-liftoff maneuver. Analysis of acceleration data indicates the INS values for airspeed are potentially more accurate after rotation.

To show the type of differences that occur, the poorest example of phototheodolite tracking is presented in this paper. It is not a typical case. Except for the consistent 0.5-percent difference in distance and speed, the data from the other takeoffs show monotonous agreement for distance, speed, and mean acceleration values. Figure 11 presents a comparison of flight path velocity for the selected takeoff. Before rotation, the nominal 0.5-percent difference exists. After liftoff, a marked increase in phototheodolite velocity develops, then decreases to equal the INS velocity after a 35-foot height.

The period from initiation of rotation to constant altitude climbout is the most difficult in which to obtain good data with ground tracking systems. Frequently, the tracking target is on the nose or tail of the aircraft and rotates around the wheels or center-of-gravity. The inertial navigation system units are normally located near the center-of-gravity.

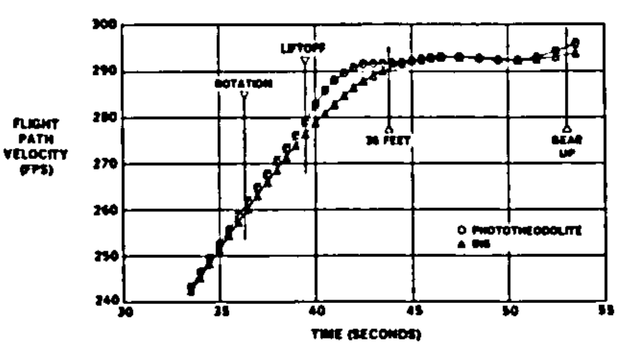

FIGURE 11. FLIGHT PATH VELOCITY

ACCELERATION

The instantaneous acceleration values from tracking data are calculated by the X2CC computer program using a seven-point numerical derivative of the flight path velocity values. The calculation interval normally used for takeoff performance is 0.5 second. The results of the example takeoff are presented in Figure 12. The

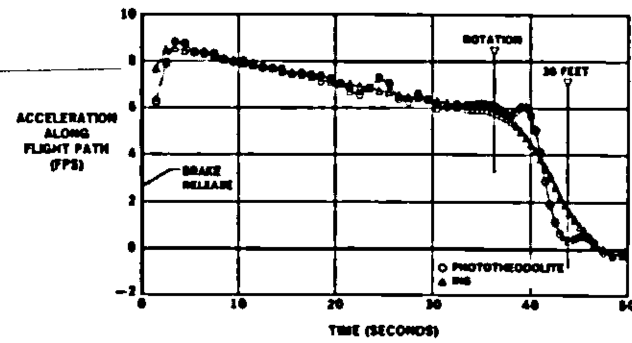

FIGURE 12. FLIGHT PATH ACCELERATION

agreement between the two tracking systems is excellent up to rotation, except for a little scatter of the phototheodolite values. After rotation, the phototheodolite values increase during a period of constant thrust and increased drag. Logic indicates that the INS values are closer to the true values. These acceleration values indicate that the INS flight path velocity values are probably more representative of the true values than the phototheodolite values.

HEIGHT

Stand-alone INS height data were verified primarily by comparing INS values with height values provided by the aircraft radio altitude system corrected for runway slope. The differences between INS height and radio altitude for six flights are shown in Figure 13. The INS values are 1 to 2 feet lower than the radio altitude values. The scatter of data is ±2 feet.

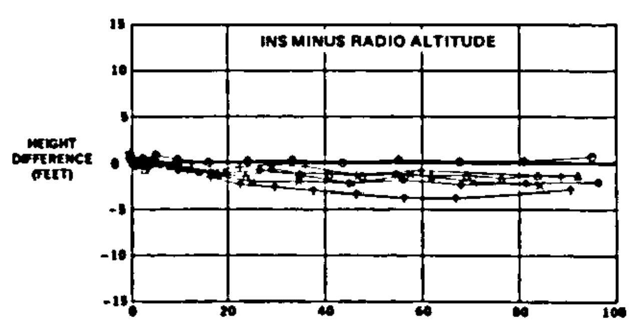

FIGURE 13. HEIGHT DIFFERENCE

Figure 14 shows greater variations in a similar comparison with initial phototheodolite data. Since these data were obtained, a procedure has been developed that uses indicated INS lateral distance to improve the

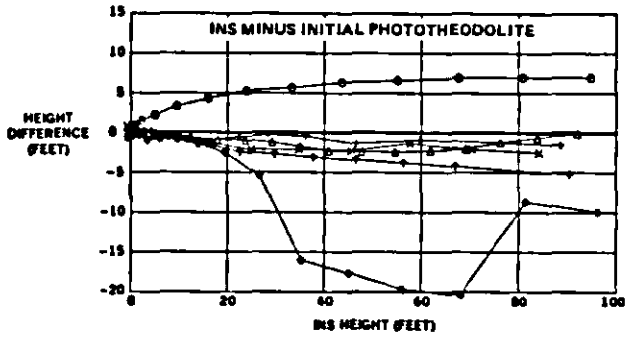

FIGURE 14. HEIGHT DIFFERENCE

quality of the phototheodolite data. The good agreement of INS and radio altitude indicates that they are superior to the phototheodolite method of determining height for takeoff performance tests. The X2CC computer program provides for use of either the INS height or radio altitude height.

AIRSPEED

Tracking systems determine flight path velocity in space. The speed of the aircraft in the air mass is presented as a function of airspeed. True airspeed is determined by adding the mean flight path wind component to the space speed determined by the tracking system. Operational takeoff performance data are normally referenced to equivalent airspeed (V_E), which is obtained by multiplying true airspeed by the square root of the ambient atmosphere density ratio. The accuracy of the V_E values is dependent on the mean component wind used to determine true airspeed.

The current Douglas flight test procedure for determining the mean component wind for takeoff test is to use the mean of the calculated PTO wind for 5 to 10 seconds before rotation.

The PTO wind values obtained using INS and ground tracking systems vary by the incremental difference between the flight path velocities. The PTO measured wind values for the example takeoff are shown in Figure 15. The INS PTO wind measured before rotation is 0.8 knot greater than the phototheodolite wind values. The mean values selected are shown in the figure. In this case, the measured values increased more rapidly than the standard one-seventh power variation with altitude.

The final equivalent airspeed values of the sample run are shown in Figure 16. The final V_E values are identical before rotation. The difference between flight path velocity values measured by INS and phototheodolite has been compensated by the difference of the PTO measured wind component. Through the use of PTO wind, the final results of various tracking systems become equal except for the variation of each system with respect to the true flight path velocity. For the 15 takeoffs studied, the maximum difference in V_E values before rotation was 0.3 knot. The zero-wind distance values were also very similar. The PTO wind values as used in the stand-alone INS procedure compensate for the minor differences between tracking systems. These results are evidence of verification of the stand-alone INS tracking procedure for takeoff performance.

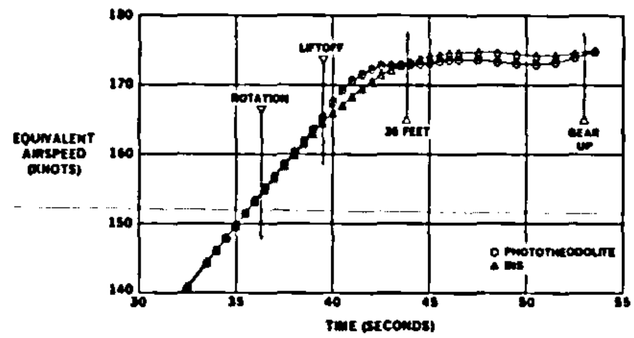

FIGURE 16. EQUIVALENT AIRSPEED

COEFFICIENT DATA

The X2CC computer program includes the capability to calculate indicated drag coefficient and indicated rolling coefficient. Indicated drag coefficient is determined using measured instantaneous acceleration, calculated thrust, and an estimated rolling coefficient. Results of the example takeoff are shown in Figure 17. The values at the start of the run are scattered due to the low airspeed. Indicated rolling coefficient is calculated using acceleration, thrust, and an estimated drag coefficient. Typical results are shown in Figure 18.

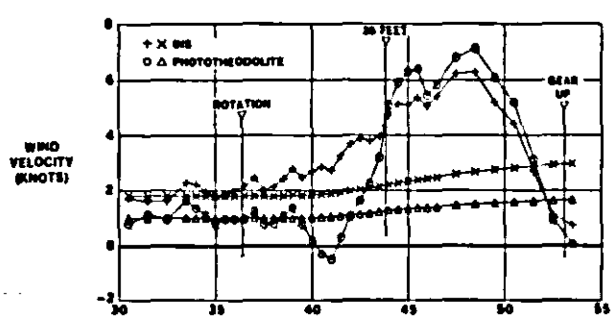

FIGURE 15. TOTAL PRESSURE (PTO) WIND

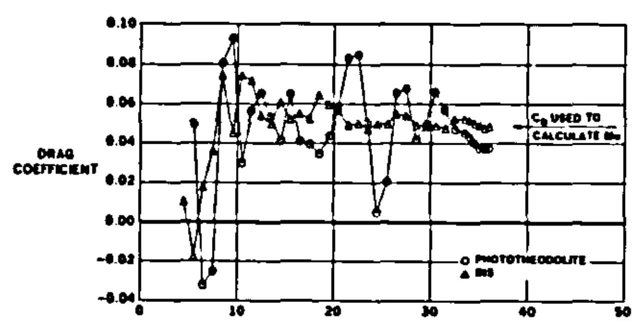

FIGURE 17. INDICATED DRAG COEFFICIENT

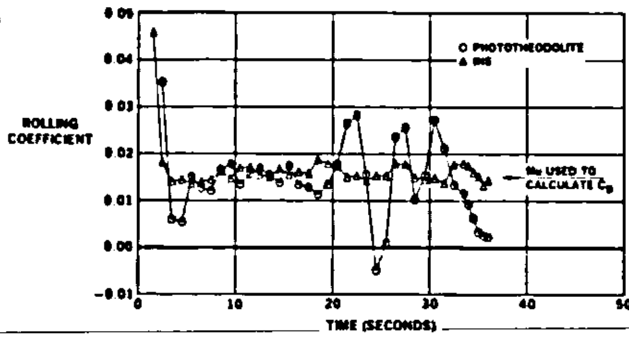

FIGURE 18. INDICATED ROLLING COEFFICIENT

By adjusting the estimated coefficient values to the mean of the calculated values, a set of coefficients is obtained to match the calculated thrust. In some cases, the calculated coefficient values indicate that the thrust values are not correct. The coefficient values are more sensitive to data errors than the other output parameters values. A primary advantage of the calculation of coefficients is to confirm the quality of the measured input parameters. If temperature, thrust parameters, acceleration, equivalent airspeed, or other items are incorrect, the coefficient values will differ from nominal values. Figure 19 shows typical coefficient values during a simulated engine failure test. The equivalent values before and after engine cut indicate that the calculated thrust and the estimated coefficient values are coordinated.

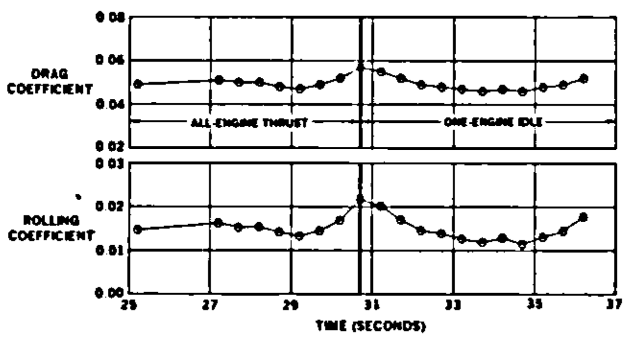

FIGURE 19. INDICATED COEFFICIENTS WITH ENGINE CUT

Coefficient values obtained using INS tracking are smoother than those obtained using ground tracking data.

CONCLUSIONS

The following conclusions have been made:

- The LTN-51 INS with a 0.1-second update rate of component ground velocities provides space position tracking data equivalent to current ground tracking systems for determination of takeoff performance.
- The PTO wind measurement procedure compensates for minor differences in distance and velocity between tracking systems.
- Either the INS or radio altitude system is superior to phototheodolite for measuring height.
- INS tracking data are superior to ground tracking data for determining instantaneous acceleration values.
- INS tracking data are better than ground tracking data for determining performance coefficients and airspeeds for events following rotation.
- The stand-alone INS procedure enables takeoff performance to be obtained without reference to ground system measurements.

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PADDS - A PORTABLE AIRBORNE DIGITAL DATA SYSTEM

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Abstract

The physical size of data acquisition and reduction equipment, while rapidly shrinking, has not kept pace with the demands for the quick response required for in-service testing and troubleshooting. In order to meet these requirements, Boeing built a system called the Portable Airborne Digital Data System, or PADDS. This system quickly interfaces with new generation aircraft digital avionics systems, acquires and records selected information from the Airborne Radio Incorporated 429 (ARINC 429) standard buses. It allows for the playback of the data (after the flight) in an office-type environment. It has the ability to obtain analog strip charts of the data. This paper describes the PADDS, its uses, and future plans.

Introduction

When you have a 747 all to yourself with no customer interior to contend with, the size and weight of onboard data acquisition and reduction equipment is of little consequence. Its an entirely different story, however, when that same 747 is in revenue service between London, England, and Johannesburg, South Africa, or when it is awaiting delivery to the customer. Then, quick response and physical size are of the utmost importance.

Boeing has attempted to meet the requirement for acquiring data from aircraft in service over the years with a variety of equipment. One of the requirements 20 years or so ago was to measure the fatigue life of B-52s while in service. In those days, the instrumentation consisted of a few accelerometers connected to an oscillograph. Boeing Customer Service representatives located at the various Air Force bases would load the paper in the oscillograph, and periodically check to see when the magazine was empty. When it was, he would replace the magazine and send the exposed roll back to Boeing for development and manual data reduction. Needless to say, the amount, accuracy and turn-around times associated with this type of data collection and reduction would hardly be considered sufficient with today's technology.

A more recent attempt to solve the "space-time" problem was the "suitcase" data system, which turned out to be a "steamer trunk" system. This system uses various parts of our large data acquisition and reduction system, packaged so as to be more easily transportable to the test area.

The advent of the Boeing 757 and 767 digital avionics systems was the catalyst that allowed development of the Portable Airborne Digital Data System (PADDS).

Digital Avionics Overview

The Digital Avionics System is the interconnection of various system components using the Mark-33 Digital Information Transfer System (DITS). This is a system in which an avionics system element having information to transmit does so from a designated output port over a twisted and shielded pair of wires to all other system elements having need of that information. Bi-directional data flow on a given twisted and shielded pair of wires is not permitted.[1] Each twisted and shielded pair of wires is commonly referred to as a "bus." Each information element on the bus is called a "word" and is given a number or "label." Each bus is given a name comprised of the acronym for the Line Replaceable Unit (LRU) and the bus from which the measurement is made. For example, bus IRULA1 LABEL 312 is the Inertial Reference Unit, Left, Bus A1. The label (312) is the octal number encoded into the first eight bits of the ARINC word and is the label for Ground Speed.

Much of the information contained on the various buses is information that, prior to the Digital Avionics Systems, had to be obtained by installing transducers, wiring, power supplies, signal conditioning and acquisition equipment. Now this same information is simply "picked off" the appropriate bus by the acquisition equipment and the wiring confined primarily to the airplane's electronics bay. For example, most of the airplane's surface positions, rudder, elevator, aileron, spoiler, and flap, are available on the avionics buses. This means that not only can the avionics systems themselves be tested, but also that the basic airplane data is available for other types of testing.

The basic information element is a digital word containing 32 bits. There are five application groups for such words: Binary (BNR) data, Binary Coded Decimal (BCD) data, Discrete data, Maintenance data and Acknowledgment, ISO alphabet #5 and Maintenance data (AIM).[1] Three of these, the Binary, Binary Coded Decimal, and Discrete, are of interest for data acquisition. In all three cases, bits 1 through 8 are the word label. Bits 9 and 10 are the Source/Destination Identifier (SDI) bits. The SDI function may find application when specific words need to be directed to a specific system of a multi-system installation or when the source system of a multi-system installation needs to be recognizable from the word content. The SDI bits are not available for this function when the resolution needed for data necessitates their use for valid data. Bit 32 is a parity bit for all groups.

For BCD data, bits 30 and 31 are the sign/status matrix (SSM) bits and have the following meaning:

Bit No. 31	30	Meaning
0	0	Plus, North East, Right To, Above
0	1	No Computed Data
1	0	Functional Test
1	1	Minus, South West, Left From, Below

For BNR data, bits 29, 30 and 31 are the SSM bits and have the following meaning:

Bit No. 29	Meaning
0	Plus, North East, Right To, Above
1	Minus, South West, Left, From, Below

Bit No. 31	30	Meaning
0	0	Failure Warning
0	1	No Computed Data
1	0	Functional Test
1	1	Normal Operation

For BCD data, bits 11 through 29 contain the data and is encoded bits 11 through 14 the least significant digit and bits 27 through 29 the most significant digit.

For BNR data, bit 28 is equal to one-half the stated range of the measurement, bit 27 equal to one-fourth, bit 26 equal to one-eighth, etc. down to the desired resolution of the measurement.

Discrete measurements may use single bits or groups of bits in bit positions 11 through 28. Single-bit discretes are used for on-off, up-down, yes-no type measurements, such as landing gear position, which is in one of two states. Groups of bits or packed discretes are used to convey various information depending upon the value of the group. For example, the Auto-Throttle/Speed Control look-ahead mode uses bits 17 through 20 to convey the following status information (in part):

Bits 20	19	18	17	Meaning
1	0	1	0	Standby Mode
0	0	0	0	Takeoff
0	0	0	1	Takeoff Reduced
1	0	0	0	Climb
0	1	0	1	Cruise
1	1	0	0	Go-Around
0	0	1	0	Hold

See Figures 1 and 2 for a summary of the word formats.

ARINC 429 Binary Coded Decimal Word Format

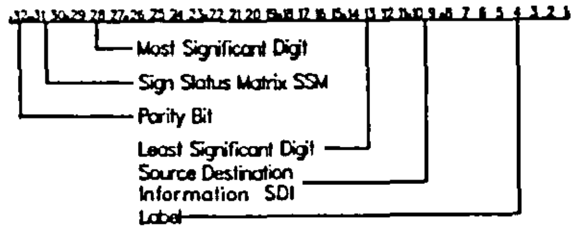

Figure 1

ARINC 429 Binary Word Format

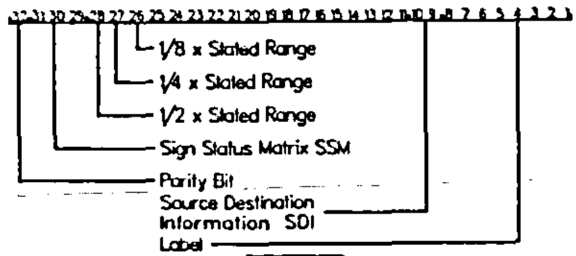

Figure 2

PADDS General Description

PADDS is used to collect and analyze data from production avionics systems ARINC 429 buses. It extracts data from the buses, stores it for later analysis, and produces displays of real time or previously collected data. The system consists of a PADDS Acquisition Unit (PACQ), PADDS Recording Unit (PRUN), PADDS Analysis and Control Monitor (PACMON), PADDS Analog Display Assembly (PANDA), PADDS Remote Operators Panel (PROP), and PADDS Field Setup Terminal (FIST). See Figures 3 and 4 for system schematics.

PADDS is used on two different types of test operations: Boeing-operated test flights and customer-owned and operated airplanes during normal revenue flights. On customer-operated flights, the PACQ and PRUN are installed in the electronics bay and the PROP installed in the flight crew compartment. Prior to flight either PACMON or FIST is connected to PACQ to perform system initialization and pre-flight, and is then removed before takeoff. After the flight, the data tape is removed from the recorder unit and played back in a ground-based system comprised of a PRUN, PACMON and PANDA. For Boeing-operated tests, the system may be configured the same as for a commercial flight, or the PRUN, PACQ, and PROP may be mounted in the passenger cabin. PACMON and PANDA are mounted in the cabin for real-time data monitoring and analysis as well as post-test data reduction.

PORTABLE AIRBORNE DIGITAL DATA SYSTEM

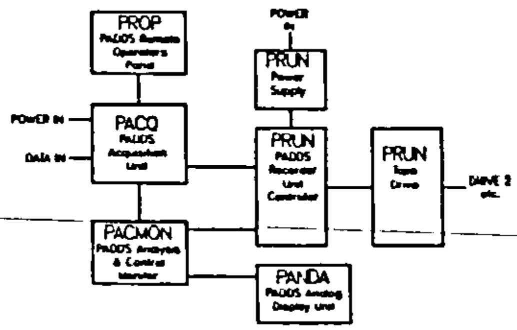

Figure 3 — Full System Diagram

PORTABLE AIRBORNE DIGITAL DATA SYSTEM

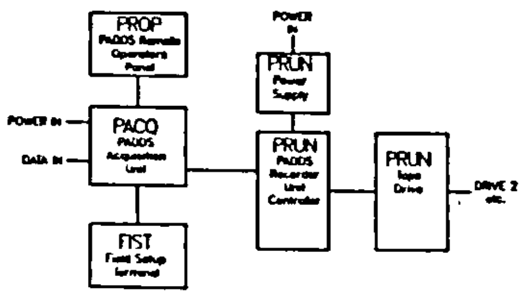

Figure 4 — Record Only System

Data Acquisition System

The PADDS Acquisition Unit (PACQ) can select up to 255 labels each from up to eight ARINC 429 buses. The information that specifies which labels the PACQ is to select is transmitted to it from the PADDS Analysis and Control Monitor (PACMON) and is retained by PACQ until a new set of selection information is received. The selection information consists of the bus number, the ARINC 429 label, and a sampling ratio for each parameter to be output by the PACQ. The bus number (1 through 8) corresponds to the bus input connector number J1 through J8. PACQ continuously outputs the selected data to the PADDS Recorder Unit (PRUN) at a rate specified by the sampling ratio. A sample ratio of "n" specifies that every nth sample the specified label on the ARINC 429 bus is to be recorded. "n" may range from 1 through 16.

In addition to acquiring data, PACQ maintains the current time of day and outputs this information to both the PADDS Remote Operators Panel (PROP) and to the PADDS data bus at a once-per-second rate. Resolution is one second and accuracy one second per 24 hours.

PACQ is housed in a four-modular concept unit (MCU) size enclosure comprised of 14 circuit card slots, a motherboard, and 14 connectors for power and system interface. The circuit cards consist of eight Front-End ARINC (FEAR) interface cards, a Time Interface Card (TIC), a Tiny Processing Unit (TPU) card, a First-In/First-Out (FIFO) buffer card, a CMOS memory card, and a Central Processing Unit (CPU) card. The CPU and CMOS memory cards are purchased and the others built in-house.

The FEAR card is used to receive data from an ARINC bus and determine if the current label is to be recorded. Ths is accomplished by using the ARINC label, including the SDI bits, to address information stored in a 1K word by eight-bit (1Kx8) non-volatile random access memory (RAM) called the Measurement Information RAM or MIR. The information in the MIR is obtained from the CPU card during setup. The MIR contains the following information:

1) Four bits to determine the sample ratio (0 to 15).
2) One bit that if true extends the label to utilize the SDI.
3) One bit that if true says the measurement is to be time tagged.
4) One bit that if true means the current word is to be processed.

A 1Kx4 RAM is used as a "count-down" counter to determine when data is to be output for a given label. Each time a label to be recorded is input, the corresponding RAM address is decremented until it becomes zero. When this occurs, the information is output and the RAM count reset with the information from the MIR.

The Time Interface and Control (TIC) card contains a time-of-day clock and miscellaneous unrelated functions. Besides keeping track of the time-of-day, the TIC card also performs as a once-per-second interrupt driver for the CPU card.

The Tiny Processing Unit (TPU) card is a programmable controller having 256 16-bit words of PROM memory. It reads data from the FEAR or TIC cards and outputs data to the FIFO buffer card. It also reads data from the FIFO and outputs it to the recorder unit if the recorder unit is ready to accept data.

Recording Equipment

The PADDS Recording Unit (PRUN) is comprised of three units: the Controller, Tape Drive, and Power Supply. These units are housed separately, each in a four MCU enclosure.

Controller Unit

The Controller unit is a 3M HCD-75 controller-formatter with a Boeing-built interface card.

All PRUN control functions and data transfer are done through the Controller via the PADDS Tape Bus, which is a parallel differential bi-directional bus that carries five control and eight data signals used by the PRUN.

The Controller utilizes an Intel 8748 micro-processor to control the operation of the recorder unit. The 8748 is a single-chip processor with 1K bytes of EPROM, two eight-bit latched input/output ports and an eight-bit data bus.

The unit decodes the input command lines and sends the appropriate commands to put the recorder in the required mode of operation. It controls the positioning of the tape during normal operation, but also allows the Host to position the tape during "Host" mode.

Power Supply

The PRUN Power Supply contains a +5 volt, 15 ampere, power supply, a +12 volt, 6.8 ampere, power supply, and a +24 volt, 1.0 ampere, power supply. Nickel-Cadmium (NICAD) batteries are used to back up the +5 volt and +12 volt power supplies in case of input power failure. The +24 volt power supply is used to provide charging current for the NICAD batteries.

The unit is designed to provide the primary power for the PRUN subsystems and to provide battery backup for a period of five minutes after input power failure.

Tape Drive

The Tape Drive is a 3M HCD-75 unit, modified to be compatible with the system. It utilizes a 3M DC600HC high-capacity, preformatted tape cartridge as the recording media. A single cartridge can hold 67 megabytes of data.

The drive moves the tape in one of three modes: (1) the search mode at 60 inches per second (ips); (2) the streaming mode at 30 ips; or (3) the start-stop mode. The search mode is used during load and unload modes as well as when searching for a specific tape block. The streaming mode is used during recording and playback when the average data transfer rate is 17.5K bytes/second. Below 17.5K bytes/second, the drive functions in the start-stop mode.

Remote Operators Panel

The PADDS Remote Operators Panel (PROP) is a hand-held display and control device normally connected to the PADDS Acquisition Unit. It consists of a six-digit, numeric, liquid-crystal display, six light-emitting diode (LED) indicators, four pushbutton switches, and a toggle switch. All switches are guarded to protect against accidental activation.

PROP receives messages from PACQ which activates the display and indicators and sends messages to PACQ defining the state of the switches, excluding the reset switch. The reset switch is connected directly to the PACQ hardware reset input and causes a processor restart when pressed.

The LED's show system not ready, beginning of tape, end of tape, error, monitor control, and low battery.

The switches control run/stop, tape rewind, error reset, PACQ reset, and display tape remaining.

Data is received and transmitted as asynchronous, eight-bit bytes at 9600 baud. American Standard Convention for Information Interchange (ASCII) conventions are followed, but the system is not restricted to ASCII characters.

Analysis and Control Monitor

The PADDS Analysis and Control Monitor (PACMON) is a self-contained, portable computer system consisting of a processing unit, floppy disk, keyboard, CRT display, power supply, and interfaces to the other PADDS units. PACMON provides the operator with the capability to display and edit the PADDS support data, initializes the acquisition unit, controls the recording unit, and displays data input from the acquisition unit and the recording unit. PACMON may be operated in a stand-alone mode to perform data base maintenance. It may be operated in a playback mode with the recorder unit only to play back and display previously recorded data. It may also be operated in a monitor mode to monitor and display data in "real-time."

Hardware

PACMON hardware consists of a keyboard unit and a display unit. The display unit houses the CRT, which is a Ball TV-90, nine-inch diagonal monochrome CRT with a display capability of 80 columns by 25 lines of text. The display unit also houses a Shugart SA-450, double-sided, double-density, 5.25-inch floppy disk drive. A Cherokee International Model QX2 power supply (providing +5 VDC, ±12 VDC, and +24 VDC) is also housed in the display chassis.

The processing unit contained in the display chassis consists of an I/O processor, an application processor, a floppy disk controller, and a non-volatile program memory.

The I/O processor card is an 8086 microprocessor-based card that has the primary job of providing the interface logic to connect PACMON to the PADDS data bus. In bus monitor mode, the processor receives data from the bus and stores it in a 4K byte FIFO buffer. In bus master mode, PACMON may read data, write data, read status, and write commands to the data bus.

The applications processor card is also 8086 microprocessor-based and provides the keyboard interface logic, the CRT display control, and an eight-channel, 0 to 5 VDC digital-to-analog converter.

The floppy disk controller card gives the applications processor (AP) card the ability to read from or write to the floppy disk. It transfers data betwen the IEEE-796 bus memory in the AP and the disk with no intervention required by the AP other than to initiate the operation via appropriate commands.

The non-volatile program memory card contains EPROMs which contain the firmware required for the system.

Software/Firmware

The software installed on the PACMON consists of Input/Output Processing (IOP) firmware and Application Program (AP) firmware and software. The IOP firmware consists of the IOP Control program and the IOP Debug utility. The firmware is stored in EPROMs installed on the I/O Processor card. The AP firmware is comprised of the Nucleus and the Executive which are stored in EPROMs on the non-volatile memory card, and the AP Bootstrap and the AP Debug utility which are stored in EPROM's installed on the AP card. AP software is stored on a floppy disk and loaded into volatile memory on the AP card at PACMON initialization.

The IOP Control program handles the interface between the PACMON and the PACQ/PRUN subsystems.

The IOP Debug utility provides monitor and debug functions used in maintaining and developing PACMON hardware and software.

Nucleus performs functions to support the Executive and Application processes such as memory management, interrupt handling and inter-process communications.

The Executive performs the functions of CRT/keyboard handling, job control, data base/disk management, measurement processing and other miscellaneous functions.

AP bootstrap begins execution when power is applied to the system or when the AP is reset. It initializes the Nucleus and begins execution of the Executive.

AP Debug provides monitor and debug functions for use in maintaining and developing application programs.

Application Programs

PACMON application programs support data maintenance, setup, tape control quicklook display and strip chart display. They consist of a "command" task that is executed whenever a command for that program is entered on the command line. If the command requires data processing, then the command task will cause one or more data tasks to be executed. The command task and all data tasks execute in parallel until a command task deletes the data tasks.

The Edit program enables an operator to modify, delete, add and display entries in the Measurement Information Table (MIT), LIST, and Format data bases.

The MIT and LIST programs enable an operator to display a catalog of entries in the respective data base.

The Copy program allows an operator to copy a PACMON diskette and to move application programs or data bases from one diskette to another.

Format enables an operator to initiate the setup of the PACQ such that PACQ will acquire and output the measurements specified in the Format data base. It also displays a list of the measurements contained in the setup specified by the Format data base.

The Time program allows the operator to set the PACQ time-of-day clock.

Tape Control program enables an operator to control the operation of the PRUN tape drive.

The Quicklook program displays engineering units values of selected ARINC 429 measurements on the PACMON CRT display.

The Analog program outputs scaled analog voltages representing engineering units values of selected ARINC 429 measurements to the PADDS Analog Display Assembly (PANDA). It also outputs time-of-day and control signals to PANDA.

Analog Display Unit

The PADDS Analog Display Assembly (PANDA) is a Watanabe Linearcorder Mark VII WR3101 strip chart recorder. It is used to record up to eight measurements concurrently output from the PACMON. The measurements to be output to the PANDA are selected through the LIST data base. In addition to data, the recorder writes the title of the LIST data base being used, date, and time on the recorder paper margin.

PANDA records on heat-sensitive paper with individual pen heat control. Paper speed may be controlled via recorder selector switches or via PACMON in external control. A timing tic mark may also be controlled either via recorder selector or via PACMON.

Individual pen position and sensitivity controls are provided through signal preamplifiers with range selection switches, fine gain adjustment pots and a zero volt position pot.

Field Setup Terminal

The Field Setup Terminal (FIST) is a hand-held terminal, Model 42A, made by G. R. Electronics Ltd. The unit is used in place of the PACMON to set up or monitor the PACQ operation when it is not convenient to transport or install the larger PACMON.

FIST is a full 128-character ASCII terminal connected to the utility port on the PACQ. It functions at a baud rate of 9600 using RS232 protocol. It displays 40 characters, in two lines of 20 characters, on a liquid crystal display screen. The unit "remembers" 80 lines of conservation. The keyboard contains 53 keys with some serving double or triple functions through the use of three "shift" keys.

FIST enables the operator to display, search, add or delete a "record" which is the information needed to select a label on a bus and pass the information to the PRUN. After modification of the records, the PACQ front-end cards may be setup with a command from FIST.

The PACQ time-of-day clock may be displayed or changed by FIST.

In the monitor mode the operator may select a bus number and label to be displayed. If the selected bus and label are being updated, the data in hexadecimal format is displayed on FIST. If it is not being updated, the message "NOT FOUND" is displayed.

Data Bases

PADDS utilizes three data bases to define, monitor and display ARINC 429 data. These are the Measurement Information Table (MIT), the LIST, and the FORMAT.

The MIT data base contains all the information necessary to acquire the desired data, reduce the information to obtain engineering units data, and display the information in an intelligible format. It is indexed by a one to seven digit measurement number.

The LIST data base contains groups of up to 20 measurements each that are to be displayed or output to PANDA as a unit. The LIST data base is identified by a one to four alphanumeric character string.

The FORMAT data base contains the airplane model number, airplane serial number, test number, ARINC bus versus input port assignments, and the list entry identifier which identifies up to 28 lists from the LIST data base, which in turn defines the set of active measurements.

System Operation

One of the first and most important steps in operation of PADDS is data base maintenance. The MIT data base is updated when there is a change to one of the ARINC 429 specifications or when a new ARINC 429 measurement is to be added to the MIT. The last thing to be updated is the FORMAT data base which is entirely dependent upon the airplane and testing to be accomplished. Updates to the data bases are accomplished by using a ground-based PADDS or the one being installed on the test airplane.

If the airplane to be tested is in revenue service, the PACQ and the PRUN equipment is mounted in the airplane's electronics bay with the PROP located in the cockpit. Power and ARINC bus connections are then made. FIST is connected to the system and used to make any PACQ "front-end" modifications and to preflight and verify the desired measurements. A flight tape is loaded and checked, FIST disconnected, and the system ready to go.

During flight, the system may be operated by the normal flight crew, left on during the entire flight, or operated by Boeing personnel occupying the cockpit observer's seat.

Following the flight, the tape is removed and taken to a location where the PACMON, PRUN, and PANDA are set up. The tape is played back and the data observed on both the PACMON CRT display and the PANDA strip charts.

For long flights, up to four recorder units may be installed. The system automatically starts recording on the next unit when tape runs out on the one currently being used. Tape capacity is approximately 67 megabytes. Record time is dependent upon the number of measurements and sample rate. At maximum recording rate of 17,500 bytes/second, one tape will give approximately one hour of recording.

For an airplane being operated and tested by Boeing, the PACQ and PRUN equipment may either be mounted in the electronics bay or in the passenger cabin. The PROP, PACMON and PANDA may also be mounted in the cabin if real-time data monitoring is desired. When PACMON is installed, it is used for system setup and preflight rather than FIST.

Future Plans

Several items are being considered to improve PADDS. One of these is the addition of a non-ARINC analog input capability. This would allow us to add some information that is not on the ARINC buses. Careful consideration is being given to this in order not to turn PADDS into the large system that we were trying to get away from when PADDS was developed.

Another improvement is to allow the maintenance and creation of the PADDS data bases via our large-scale IBM computer. The data bases could then be written "off-line" on various instrumentation subsystems throughout the Boeing Company areas. With this improvement, one master MIT for each model airplane could be more easily maintained. The LIST and FORMAT data bases could be created from any of the 74 large-scale computer terminals available throughout Flight Test without having to have a PADDS system available.

Conclusion

PADDS has accomplished the job it was designed for. It has shown itself to be the quick-response, easily installed and maintained data acquisition, monitor and reduction system that was needed by Flight Test. The future of PADDS looks promising with more and more digital avionics being installed aboard airplanes. We only have to keep reminding ourselves what PADDS purpose is, and resist the temptation to let it grow into an all-encompassing large system.

References

1. ARINC Specification 429
   Revised August 1, 1980

2. Boeing PADDS Documentation

   D6-49935-1 Requirements
   -2 System Specifications
   -3 Subsystem Specifications
   -4 Detail Specifications
   -5 Users Manual
   -6 System Test

767/757 INSTRUMENTATION SYSTEM

by
Thomas E. Smidt
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Abstract

In 1978, the Boeing Company took on the monumental task of building, developing, and certifying two new airplanes. As a result, the Flight Test Instrumentation Projects Group faced defining and addressing the requirements for certifying those new airplanes.

With program requirements defined, surveys revealed that the existing data system was unable to handle the expected data workload. Areas with the greatest need for improvement were data acquisition, real-time monitoring, and data support.

Improvements made in the areas of data acquisition, real-time monitoring and data support saved many hours and allowed on time cerification of two airplane types plus an alternate engine.

Introduction

Two concurrent airplane certifications with entirely new digital avionics systems presented new challenges for acquiring flight test data onboard 767 and 757 airplanes.

The 767 and 757 are twin engine commercial airplanes in the 215 and 186 seating capacity respectively. The 767 first flight was in August, 1981. The 757 flew six months later in Feburary, 1982.

Each program lasted nine months and logged over 2000 hours. To add to the challenge, the 767 had a second engine to certify at the same time.

Preparation for the 757/767 testing began in 1977 with a survey to see if the existing data system could handle the expected data collection, reduction and turn-around times.

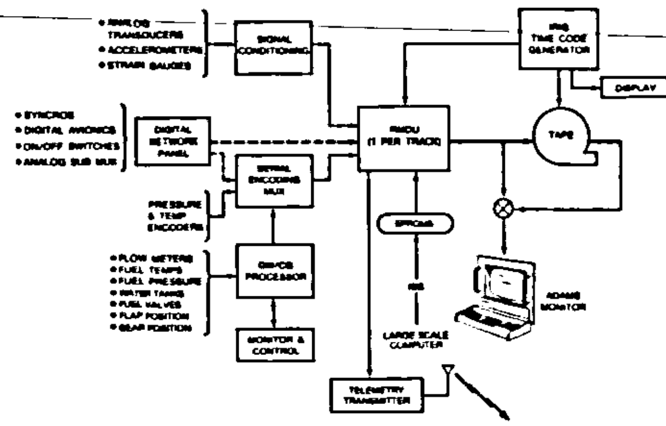
The result of the survey was that there was no possibility of being able to handle the expected data with the existing system. There were three major areas of concern to do the data collection:

1. Data acquisition system
2. Real time data monitoring
3. Data support system

Data Acquisition

The basic data acquisition system was developed for the 747SP testing: a high-speed pulse code modulation (HSPCM) system. With added capabilities and improvements, this system would handle the task.

Since the system was basically the same as had been used, it enabled approximately 80% of the hardware to be defined. This in turn allowed an early start on the engineering design associated with the equipment console installation, as well as being able to optimize the procurement of equipment and the identification of long lead items.



Airborne Systems

The basic data acquisition system consists of analog transducers, such as differential pressure, acceleration and strain gages. These are connected to the system through an analog signal conditioning network panel which provides excitation voltage to the transducers as well as performing such functions as filtering, isolating, presample amplification, etc.

The digital side of the house is handled in much the same way, except that the signal conditioning consists mainly of "bit-shuffling" and timing circuits to transform the digital signals to our standard PCM signals.

One thing not foreseen during the very early stages of planning was the impact of the new digital avionics systems on the data acquisition requirements. Digital avionics measurement requirements jumped from around 450 measurements to about 3500 measurements. After much discussion over various methods of handling this requirement, it was decided not to attempt to develop anything new. The same system would be used but in much greater quantities of hardware. This method is to take the ARINC 429 digital data and convert it to our standard offset binary PCM system. This approach allows ARINC 429 data to be merged with all the other data being collected from other sources in real time. This in turn eliminates the problem of having to merge and time correlate data from two separate sources during the post test data reduction phase.

Unlike the missile side of the business, the transducers cannot be fired away, never to be seen again. Instead, they are used over and over again on different projects, and used for a long time on the same project. As a result, transducers must be periodically recalibrated in the calibration lab. The equipment used to calibrate the transducers are secondary standards that are directly traceable to the National Bureau of Standards through the Boeing Metrology Lab. At present there are around 30,000 active calibrations on file.

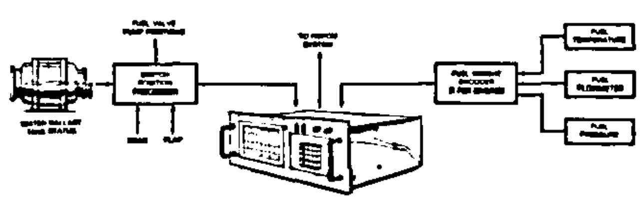
For many stress/strain loads type measurements, strain gages must be installed by the "strain gage lab" and calibrated by the cal lab. These parts range in size from small inch-long rod ends to airplane landing gear parts weighing tons. In many cases, parts come off the airplane and into the lab. For example, the main structural members of the airplane obviously cannot just be taken off the airplane and taken into the lab. In cases like this, two possible things can be done. One is to calculate what the output of the gage will be knowing the characteristics of the gage and the part the gage is placed on. The other method is to do a "static" calibration, where the airplane itself is loaded at various points and the output of the gages calibrated.

Calibration data is transformed into curves ranging from a simple straight line y = mx + b slope intercept type to a ninth order polynomial fit. The curve fitting is done semi-automatically in that the accuracy wanted is specified for the the curve we want to conform to the data. The computer finds the lowest order fit that is within that limit and displays the results. The problem with this is that every so often the polynomial fit looks very good at the points, but is way out of tolerance at other points along the curve. Therefore, each curve fit must be approved before it is placed on the active file. This calibration data is then converted from the calibration of millivolts out to the units of "counts" as used on the airplane. This curve transform equation is then used by the Final Data System to reduce data and is also sent to the airplane monitor system for each test for real time data reduction. Using a common data base for the reduction of data assures the results will be the same whether the data is reduced aboard the airplane or in the final data computer. Because of this common data base, the airplane data can sometimes be used for final data, resulting in a cost savings.

The heart of the system is the remote multiplexer-demultiplexer unit (RMDU). This device contains an analog processing module with a gain programmable amplifier that can apply any one of ten preprogrammed gains to any analog input at any time, gains from 1 to 500 are available. The output of the RMDU is a ten-bit binary word. Zero volts into the system will give a binary output of 512 counts. A positive five volts into the system (after amplification) will increase the counts to 1012, while a negative five volts will result in a count of 12. This output is called offset binary.

A typical output format is a 256-word mainframe with a data cycle of 20 frames. Typical data sample rates are from 5 to 400 samples per second. Higher rates are possible but become less practical. The system runs at 100 frames per second. This calculates to 256,000 bits per second of data being recorded per RMDU, which is a tremendous amount of data. To show how much data this is, the average 767 airplane had six RMDU's, each recording data 256,000 bits per second. The 767 test program consisted of 2136 hours of testing.

The proliferation of microcomputers into the marketplace facilitated designing and building some "smart transducers" in-house. One problem was in quickly determining the gross weight and center of gravity (GWCG) of the airplane. So much of the aerodynamic data depended on the GWCG that a special run of the flight tape had to be made through the ground station before any other data could be reduced. Needless to say, this was a source of much of the delay associated with turn-around times. A real time GWCG system was designed, developed, and built in-house which performs all the calculations necessary to determine the GWCG and outputs the information in real time to the normal HSPCM system. This is one of the few times that calculated rather than raw data is recorded on the airplane.



Gross Weight/Center of Gravity System

Other smart transducers included rate-of-sink, remote analog transducer, thermocouple systems, and tail height system.

The rate-of-sink system employs a police radar unit to measure aircraft vertical velocity during landing approach and touchdown. Strain gauges on the landing gear serve as event signals to lock the touchdown value.

The Remote Analog Transducer System (RATS) box contained 16 pressure transducers held at a constant temperature. Locating the box near the pressure ports improved frequency response. The output of the RATS box to the Transducer Encoder Digitizer (TED) card is a multiplex analog signal; this card feeds the HSPCM system.

The Chromel-Alumel thermocouple system (CATS) handles 20 thermocouples. Each channel is amplified and converted into a calibrated value. The digital output to the HSPCM system is a multiplexing of the 20 channels.

The Tail Height Ultrasonic Measuring System (THUMS) employs the transducer found in the Polaroid camera distance sensor. The transducer is mounted near the tail to give the pilot and the HSPCM the distance the tail is to the ground. This is important when trying to avoid dragging the airplane's tail during test conditions.

The information gathered is recorded onboard the airplane on instrumentation quality magnetic tape. This program required higher data rates than previous programs so the higher tape speeds decreased time per tape reel with the existing 14-track deck. Converting the decks to a 2x7 configuration - seven tracks down and seven tracks back - restored a four-hour tape time.

Telemetry is used for minimum crew flights, mainly flutter testing, where oscillations are introduced into the airplane which are expected to dampen out and not go divergent. Normally, the test engineers are aboard during the test flights. Monitoring the results in a telemetry room is limited by the telemetry range in Seattle, which is ringed by mountains. By being aboard, the airplane may go anywhere needed to find the proper test conditions. The conditions usually are

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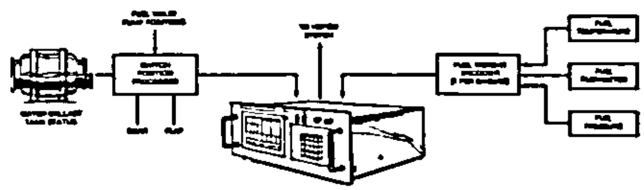

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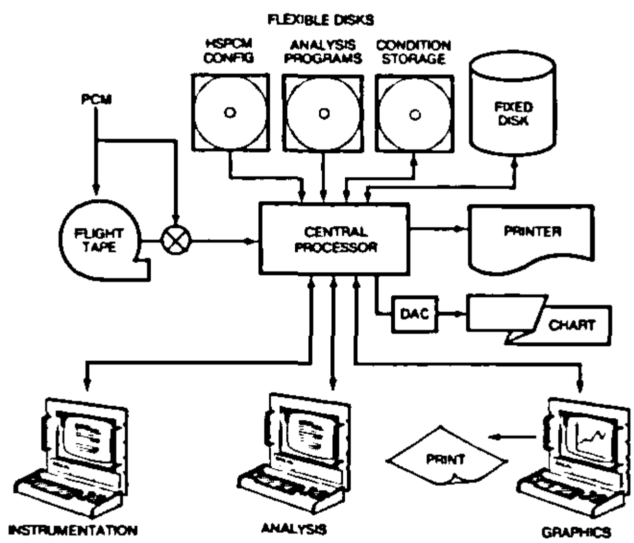
Real Time Monitor

The onboard system has been called the Airborne Data Analysis and Monitor System or ADAMS. The improved monitor software is the second generation and is called ADAMS II.

The first computer was put aboard an airplane, not as an analysis tool, but as a "window" into the data acquisition system for daily preflighting the airplane's data system. Prior to ADAMS, only one measurement at a time could be looked at and then only in "raw" counts. To get engineering units data, the values had to be looked up on a plot of the data or calculate the results manually. When preflight started to get between 300 and 500 measurements for a test, the job of properly checking each one became a rather difficult and time-consuming task. Once the computer was aboard the airplane, the analysis engineers started seeing how much use it could be in calculating some of the parameters that they needed to determine the quality of the test condition. The first application programs were written in BASIC and were very restricted in what they could do in real time. The system grew with an operating system being written in-house and the application programs rewritten into machine language to allow more multi-functions to be run concurrently. The system that resulted, was known as ADAMS I. For the 757/767 testing it was determined that changes would have to be made in what was by now a very complex and extensive real time data processing system.



Airborne Data Analysis
and Monitor System (ADAMS)

There were two main areas of concern with the hardware - the central processing unit (or CPU) and the cartridge tape system. The CPU was a Rolm 1602, which is a militarized version of the Data General Nova/Eclipse line. The maximum memory that could be used with the 1602 was 32K, which made the job of a multi-job processing challenging to say the least. In addition, newer versions of the CPU were on the market which were faster in execution time. The transportable media to get programs and calibrations loaded on the airplane system was a cartridge tape system (endless loop). The other major area of concern was the in-house written operating system which could no longer handle all the simultaneous processing requests and was not as efficient as it should have been.

The CPU was replaced with the Rolm 1666 with 64K of core memory and expansion capabilities up to 256K. The cartridge tape system was replaced with a combination fixed head/floppy disk system. The fixed head disk (two megabytes) held all the programs and operating system while the floppy disk was used to update the fixed head disk with the latest measurement configuration and test requirements. The operating system was completely rewritten and the programs that were written in machine language were rewritten in FORTRAN. With ADAMS I each airplane had to have a unique software system configured just for that airplane, which meant that each time an application program was changed or a new airplane added to the test program, a great deal of work had to be done in updating all the different systems or in generating new systems. ADAMS II made the software programs independent by removing all the internal program variables that differed from airplane to airplane or model to model. These were such things as wing area, number of engines, type of engines, etc. These variables were put into a table called the airplane parameter table which was loaded into the system onboard the airplane via the floppy disk. The same technique of using various data base tables resulted in 11 separate data base files which are loaded into the system prior to each test.

Basically ADAMS II works by looking at the PCM bit stream, either directly from the RMDU or from the playback heads of the tape recorder, as data is being collected. Being able to monitor from the tape assures us that data is being recorded properly and will be recoverable during the post test data reduction. The cost of monitoring from the playback heads is a slight time delay and the susceptibility of the system to noise that is generated in the recording process.

The PCM decommutators accept the serial PCM bit stream, synchronize on bit, frame, and subframe, and convert the data from serial to parallel. It then passes the data to the word identifier.

The purpose of the word identifier is to assign each data word coming from the decom a unique computer memory location and place the word in that location using direct memory access (DMA) techniques. This hardware decommutation system has two big advantages over software decommutation. First, since it works on a cycle-stealing basis, it requires much less CPU time which can be used for data processing. The other advantage is that application programs do not have to wait for the required data to be sampled and passed to the program. The latest data is always in a known memory location and can be accessed at any time. This same technique is used to pass information between application programs. For example, an application program called "Basic Airplane" calculates most of the basic data needed by many other programs, such as mach number, pressure altitude, equivalent airspeed, indicated airspeed, calibrated airspeed, true airspeed, etc. and makes the calculated results available to any other program that requires the information, rather than recalculating it in each application program.

Two CRT/keyboard test stations are the main interface between man and machine. From these stations the engineer can monitor any measurement (either real, such as rudder position, or calculated, such as mach number) in real time. The measurements are presented in groups of up to 20 measurements at a time. These groups may be predefined and loaded onto the system through the floppy disk prior to flight or may be created aboard the airplane. The readout gives data title, measurement number, current data value (in engineering units, counts or binary), and measurement limits, and will flag any measurement that is outside those limits by placing an asterisk beside the measurement number. The displayed data is updated once per second.

The fixed head disk is the device that contains the operating system and data bases. It is a ruggedized disk that can withstand the severe conditions of test flying. The only problem with the disk is that it is a pressurized unit. Problems can arise when depressurization testing and the internal pressure gets too low (air too thin) and the heads stop flying.

The floppy disk is used for updating the fixed head disk with the latest information about the measurements and for loading new software systems as necessary. It is also used to record application program summary data that can be re-entered into the system on subsequent tests to compare and summarize data.

The line printer has two uses. It can print a hardcopy of the CRT on command from the keyboard, or it can be used as a time history printout of measurements selected by the operator. The operator may select any number of measurements to be printed at any sample rate, with the limit measurement/rate being dependent upon the print speed (1500 lines per minute). The more measurements you want printed, the greater the time increment between printed samples must be.

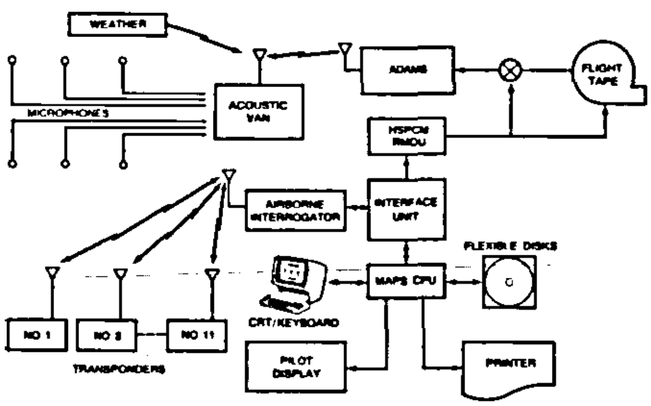
The digital-to-analog converter and strip chart recorder go together to provide the engineer with visual representation of any measurement, again either real or calculated, in order to see such things as phase relationships, sequence eventing, relative amplitude, etc. which cannot as readily be seen with digital readouts.

The remote digital displays are two-channel, switch-selectable devices that can be placed anywhere in the airplane and will give a decimal display in engineering units of any measurement being recorded or calculated. The measurements to be displayed on any readout are selected by program control, not at the readout.

Late in the test program, an onboard color graphics system was added to give the capability of obtaining real time crossplots, time history plots, and the ability to plot deviations from standard curves. A graphics printer was also added to give a hardcopy.

Over 20 application programs have been developed for ADAMS which range in complexity from simple information display of instrumentation configuration to very complex real time data reduction.

As an example, one of the big concerns is the effect of aircraft noise on the surrounding community. A great deal of time, money and effort goes into the measurement of sound levels in order to find techniques to minimize the levels. An application program called Acoustics was developed which uses the data from the onboard Microwave Airplane Positioning System (MAPS) to determine the airplane position and attitude with respect to a ground-based microphone field. It compares the airplane position with that of up to four microphones on the ground and, when a microphone crossing occurs, stores selected data for future summarization. When all the data has been colleced, the summarized data is transmitted to a ground-based acoustics van. The acoustics van receives data from both the airplane and the microphone field.



Microwave Aircraft
Positioning System (MAPS)

For each microphone, the airborne system transmits:

Crossover Time	
Range Position	(meters)
Centerline Displacement	(meters)
Altitude	(meters)
Pressure Altitude	(feet)
Dew Point Temperature	(deg. C)
Ground Speed	(knots)
Pitch Angle	(degrees)
Roll Angle	(degrees)
Flap Position	(degrees)
Flight Path Angle	(degrees)
Rate of Climb	(feet/second)
Gross Weight	(E3 lbs)
Center of Gravity	(% MAC)
Ambient Air Temperature	(deg. C)
Calibrated Airspeed	(knots)
True Airspeed	(knots)
Relative Humidity	(%)
Wind Speed	(knots)
Wind Direction	(degrees)

The program also develops range and distance-to-go information for display to the pilot.

Other application programs compute information for stall performance, takeoff and landing performance, engine performance, cruise performance, stability and controls, and others.

speed, true airspeed, etc. and makes the calculated results available to any other program that requires the information, rather than recalculating it in each application program.

Two CRT/keyboard test stations are the main interface between man and machine. From these stations the engineer can monitor any measurement (either real, such as rudder position, or calculated, such as mach number) in real time. The measurements are presented in groups of up to 20 measurements at a time. These groups may be predefined and loaded onto the system through the floppy disk prior to flight or may be created aboard the airplane. The readout gives data title, measurement number, current data value (in engineering units, counts or binary), and measurement limits, and will flag any measurement that is outside those limits by placing an asterisk beside the measurement number. The displayed data is updated once per second.

The fixed head disk is the device that contains the operating system and data bases. It is a ruggedized disk that can withstand the severe conditions of test flying. The only problem with the disk is that it is a pressurized unit. Problems can arise when depressurization testing and the internal pressure gets too low (air too thin) and the heads stop flying.

The floppy disk is used for updating the fixed head disk with the latest information about the measurements and for loading new software systems as necessary. It is also used to record application program summary data that can be re-entered into the system on subsequent tests to compare and summarize data.

The line printer has two uses. It can print a hardcopy of the CRT on command from the keyboard, or it can be used as a time history printout of measurements selected by the operator. The operator may select any number of measurements to be printed at any sample rate, with the limit measurement/rate being dependent upon the print speed (1500 lines per minute). The more measurements you want printed, the greater the time increment between printed samples must be.

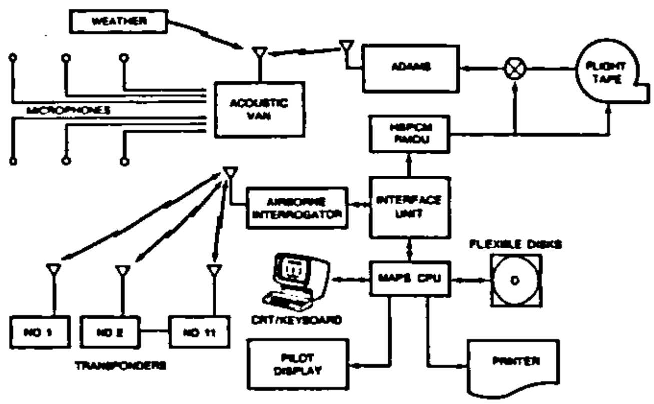
The digital-to-analog converter and strip chart recorder go together to provide the engineer with visual representation of any measurement, again either real or calculated, in order to see such things as phase relationships, sequence eventing, relative amplitude, etc. which cannot as readily be seen with digital readouts.

The remote digital displays are two-channel, switch-selectable devices that can be placed anywhere in the airplane and will give a decimal display in engineering units of any measurement being recorded or calculated. The measurements to be displayed on any readout are selected by program control, not at the readout.

Late in the test program, an onboard color graphics system was added to give the capability of obtaining real time crossplots, time history plots, and the ability to plot deviations from standard curves. A graphics printer was also added to give a hardcopy.

Over 20 application programs have been developed for ADAMS which range in complexity from simple information display of instrumentation configuration to very complex real time data reduction.

As an example, one of the big concerns is the effect of aircraft noise on the surrounding community. A great deal of time, money and effort goes into the measurement of sound levels in order to find techniques to minimize the levels. An application program called Acoustics was developed which uses the data from the onboard Microwave Airplane Positioning System (MAPS) to determine the airplane position and attitude with respect to a ground-based microphone field. It compares the airplane position with that of up to four microphones on the ground and, when a microphone crossing occurs, stores selected data for future summarization. When all the data has been colleced, the summarized data is transmitted to a ground-based acoustics van. The acoustics van receives data from both the airplane and the microphone field.



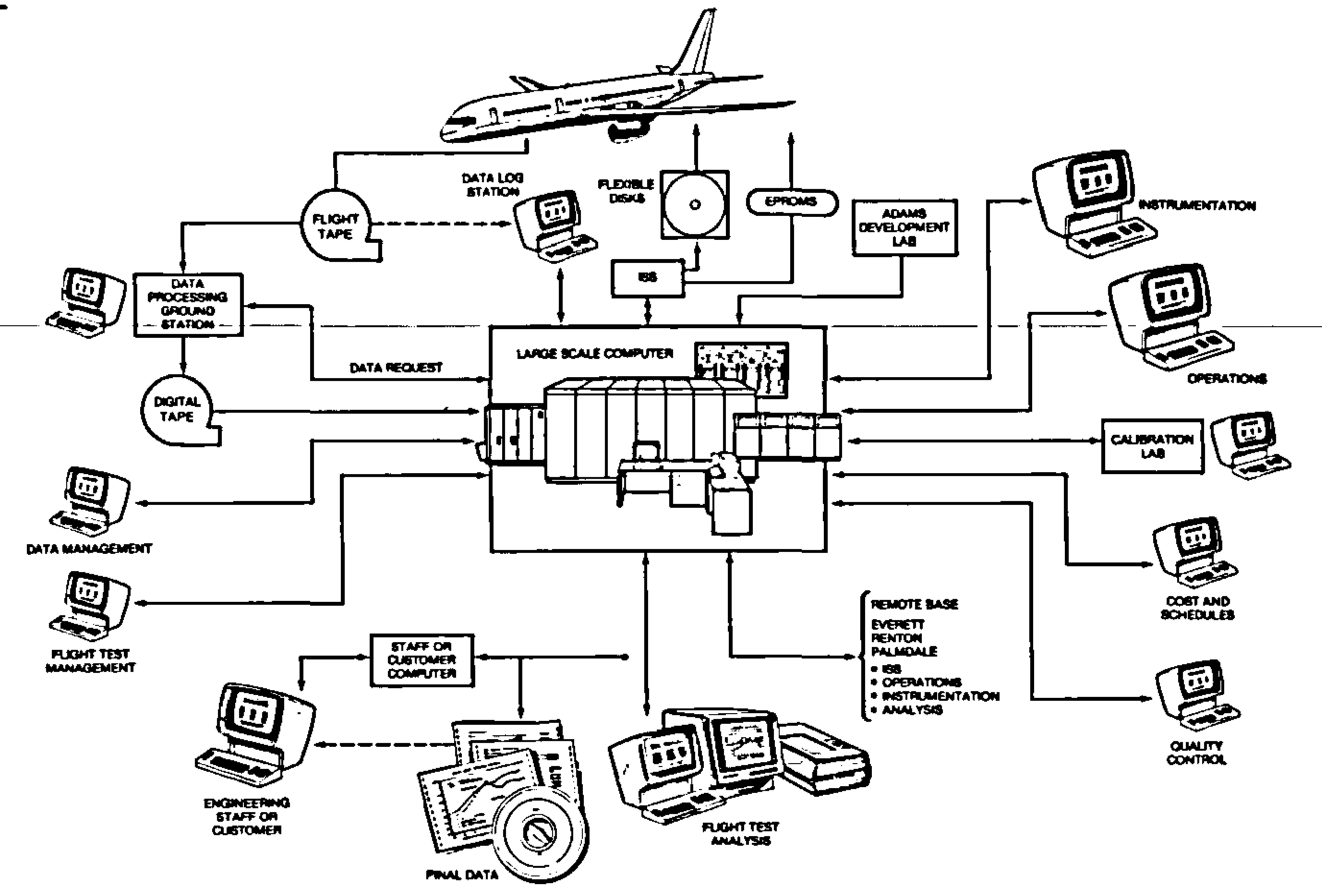
Microwave Aircraft
Positioning System (MAPS)

For each microphone, the airborne system transmits:

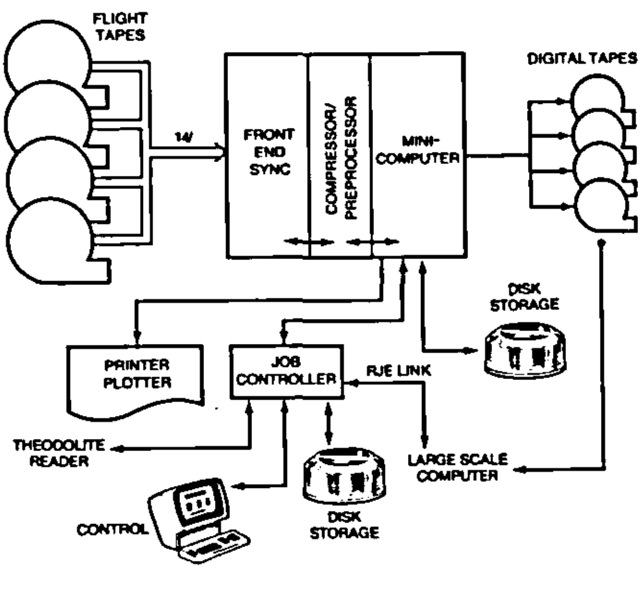
Crossover Time	
Range Position	(meters)
Centerline Displacement	(meters)
Altitude	(meters)
Pressure Altitude	(feet)
Dew Point Temperature	(deg. C)
Ground Speed	(knots)
Pitch Angle	(degrees)
Roll Angle	(degrees)
Flap Position	(degrees)
Flight Path Angle	(degrees)
Rate of Climb	(feet/second)
Gross Weight	(E3 lbs)
Center of Gravity	(% MAC)
Ambient Air Temperature	(deg. C)
Calibrated Airspeed	(knots)
True Airspeed	(knots)
Relative Humidity	(%)
Wind Speed	(knots)
Wind Direction	(degrees)

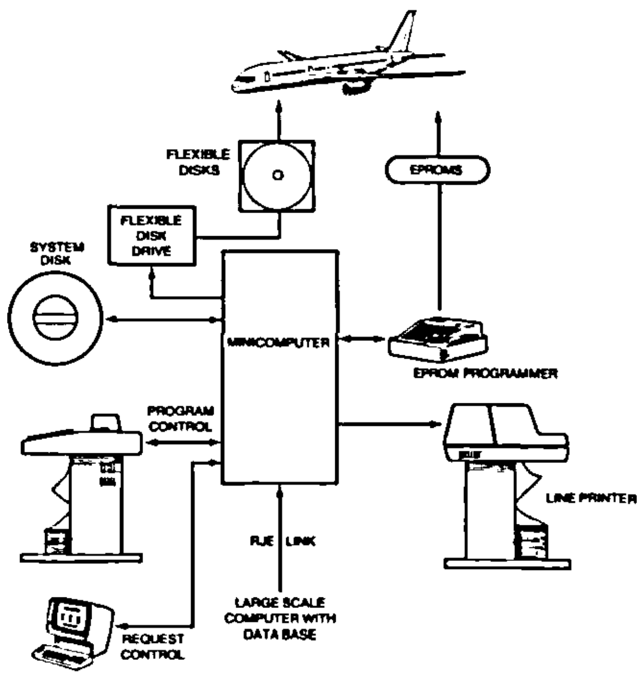
The program also develops range and distance-to-go information for display to the pilot.

Other application programs compute information for stall performance, takeoff and landing performance, engine performance, cruise performance, stability and controls, and others.



Flight Test Computing System (FTCS)





Data Processing Ground Station (DPGS)

Instrumentation Subsystem (ISS)

Data Support System

The data support system was the area of the biggest change; not only in the new hardware that was available, but much of the operating concept and procedures were changed when going from a card input, batch job-oriented system to an interactive system. Some of the engineers that thought sitting at a keyboard was a job for a typist or keypuncher soon found that they were almost enjoying being able to get the information they wanted so quickly by sitting at a keyboard.

The data support system, which is now part of an overall mainframe computer system called Flight Test Computing System (FTCS), consisted at the peak usage time of two IBM 3033 computers running in parallel with 125 alpha-numeric terminals and 18 graphics terminals connected to it. A special FTCS operating system was written which is either menu driven or command line driven.

The file most commonly accessed by Instrumentation Projects is the airplane configuration document (CONFIG). This file maintains the relationship between the measurements and which transducer is installed with what calibration. This file is in constant change, so a snapshot is taken and stored for each each test by test number.

The data processing ground station went from a system that handled a single-tape, single-track to a multiple-tape, multiple-track system.

In order to process data on the large scale computer the flight tape must be converted to an acceptable (IBM) format. In addition to extracting only requested times from the tape, the data processing ground station (DPGS) handles this convention task.

After a test flight the data tape is delivered to the DPGS. The front end hardware is set up with the unique airplane configuration.

Analysis engineers returning from the flight use the computer terminals at their desks to request data from the flight. The computer does a check to see that data request times are available; if not, the engineer is warned.

A network of "satellite" computer stations called the Instrumentation Subsystems (ISS) was created. These stations were connected to the IBM mainframe through dedicated 9600 baud telephone lines and ran on an RJE/HASP system. Five stations were built. Two of these were installed in Seattle at Boeing Field where most of the testing was to be done; one in Everett for the 767 program; one installed on the Renton flight line for the 757; and the fifth designated the mobile system. The mobile system was briefly located in Glasgow, Montana, but was in Palmdale, California, the majority of the time.

The ISS accomplished several jobs. It eliminated delays associated with hand carrying data bases since the communication was electronic. It eliminated the data base tape having to be processed by the ground station by writing the information to floppy disks which were directly compatible with the airborne analysis system. It eliminated processing of mylar tapes by interfacing directly with a PROM programmer. It provided Instrumentation with an interactive method for programming the EPROM's used in various acquisition system components.

By linking the systems development lab computers into the same network as the ISS computers, another network was created that allowed communications between those two areas. New ADAMS software or fixes to bugs could be transmitted electronically to any or all of the ISS stations. Where previously new software had to be flown to a remote base or delivered to Everett, it could now transmit the information and in a matter of minutes the software could be installed on an airplane.

Request for Instrumentation Preflight (RIP) is a list of measurements required for the next day's test. The average time for the process (which was seven hours) was reduced to about 1.5 hours, and on rare occasions, can take as little as 45 minutes.

Improvements came in many other areas. The calibration lab became much more automated with the addition of computers that controlled the calibration process from start to finish.

Conclusion

Finally, to show the success of the system improvements, compare the testing accomplished on the 757/767 with the last major test, the 747SP.

Model	No. of Flights	Flight Hours	Data Turnaround	Data Requests Per Day
767	1828	2136	10 Hours	51
757	1338	1827	8 Hours	48
747	413	776	100 Hours	10

The three main objectives were accomplished:

1) Increasing data recording capacity to handle the tremendous volume.
2) Enhancing the on-board monitor system which saved time and money.
3) Enabling the support system to respond quickly to daily needs and to handle the sheer volume of data to process.

In conclusion, never before has an aircraft company undertaken the job of testing and certifying two new model airplanes at the same time. It was a huge job, but proper preparation in both equipment and trained personnel, the job was accomplished more easily than any comparable certification task.

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HADS DIGITAL PRESSURE TRANSDUCER

BY: Andrew Dimitriou
David Hopmann
Rosemount Inc.

ABSTRACT

A new, high accuracy digital pressure transducer developed by Rosemount is discussed.

The objective was to develop a digital pressure transducer that would provide a corrected output proportional to or some air data function of the measured pressure.

The Rosemount Model 1501AT digital pressure transducer utilizes Rosemount's patented High Accuracy Digital Sensor (HADS) pressure sensing module. The module consists of a vibration/shock mounted case which houses a vibrating element sensor and buffer electronics. The HADS module is an interchangeable subassembly of the Model 1501AT digital transducer which can be changed without effecting calibration or performance on the transducer.

The 1501AT transducer performs a polynomial correction for the HADS systematic temperature and linearity effects utilizing the module's frequency, temperature and calibration coefficient signals. The result of this computation is a corrected output proportional to or some air data function of the measured pressure.

Several output formats are available, including both high- and low-speed ARINC 429 and parallel binary, with other serial formats and MIL-STD-1553B to be announced.

The incorporation of both continuous and initiated BIT functions offers capability to detect a high percentage of failure modes and denoting the resultant data as invalid.

This product provides the technical performance required for today's advanced flight testing of new generation aircraft.

INTRODUCTION

Performance requirements of the new generation aircraft have placed increased importance on accurate pressure based measurement for a variety of flight test applications. With advances in instrumentation data acquisition, usage of high accuracy digital pressure transduction has grown. There are many sensing techniques utilized by these digital pressure transducers to provide a digitally corrected output proportional to or some air data function of the applied pressure. Due to the inherent stability, the more accurate of these devices will utilize some form of vibrating pressure sensor element providing a pressure-to-variable frequency output. This output, when converted into a digital format, is then digitally processed for correction of characteristic systematic errors (temperature and non-linearity). Correction of these systematic errors is normally accomplished by a programmed set of equations, or algorithm, which will have frequency and temperature as independent variables. Coefficients or calibration constants for these equations are determined during calibration. These scaling and temperature coefficients are accessed by a digital processor from a memory device which is optimally resident with the sensor. Once the corrected signal is computed, it is sent to an I/O buffer where several different digital output formats may be utilized. This is a proven technique and is being used in various digital transducers for many different flight test applications.

It is the purpose of this paper to describe Rosemount's 1501AT digital air data pressure transducer for flight test applications. This transducer utilizes the High Accuracy Digital Sensor (HADS) recently developed by Rosemount for reliable measurement of air data.

The 1501AT was developed using several application requirements as guidelines. Highest priority among the list of flight test application requirements is maintainability and reliable performance throughout the air data transducer's life. Efficient use of microprocessor space, software, BIT (Built-In Test) and mechanical interface and installation must also be considered for high accuracy digital pressure measurement and transduction in today's flight testing of high performance aircraft.

SENSOR OPERATION

The performance of any air data transducer used for pressure measurement of high performance aircraft is highly dependent upon the sensing technology utilized to convert measured pressure into a usable output while experiencing the environmental conditions imposed by the operation of the aircraft.

Problems associated with high accuracy digital pressure transducers, which have been identified, result in reduced accuracy and performance (Ref 1). The source of the majority of these problems is the sensing technique used. Gas density, humidity, condensation and severe environmental stress (vibration, acceleration, etc.) encountered during flight test applications will all contribute to reduced sensor performance. A number of sensing techniques either ignore or unsuccessfully attempt to compensate for these errors.

To achieve the performance required for high accuracy flight test measurement, the 1501AT utilizes Rosemount's High Accuracy Digital Sensor (HADS). The HADS sensor is a vibrating element, digital pressure sensor of simple design.

The method relies on pressure to load a media-isolated vibrating metal beam, changing its natural frequency with applied pressure. The beam is loaded through a pivotally mounted lever and excited by a coil driven by a loosely coupled, closed loop, electronic oscillator, utilizing a capacitive detection scheme (see Figure 1).

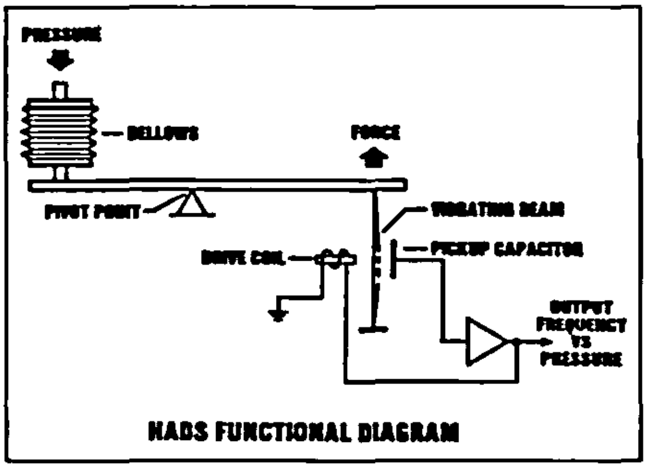


Figure 1

The sensor is mounted with its temperature resistor in a modular casting. A flight test qualified shock and vibration isolation system is provided to eliminate errors which can result from aircraft imposed vibration environments (see Figure 2). Rosemount developed and patented this system to effectively isolate from the module case the pressure sensing element,

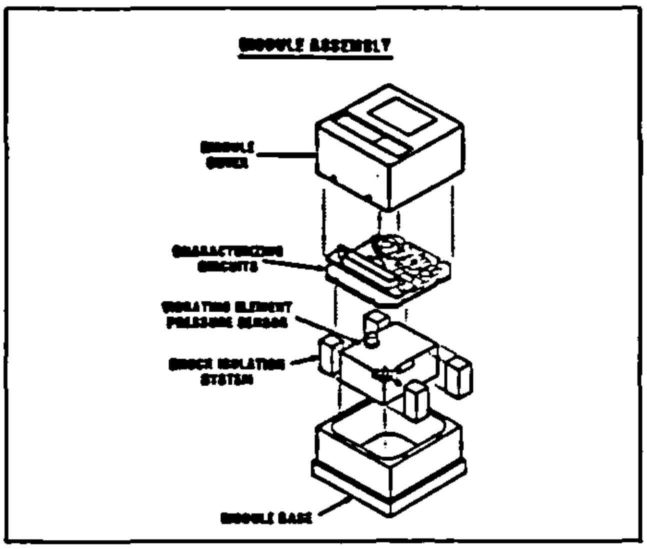


Figure 2

allowing guaranteed performance up to 15 g's of shock and 10 g's RMS of vibration. Standard isolation methods, such as foam or hard rubber mounts, will decrease accuracy and introduce vibration noise through transmissability.

The module contains interface circuitry, including a ROM (Read Only Memory) storing scaling temperature and pressure calibration coefficients (see Figure 3). External digital processors use this information to correct for systematic errors.

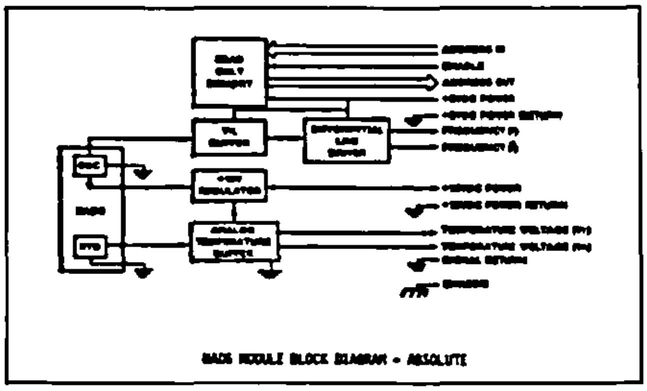


Figure 3

Calibration is accomplished through the use of a computer controlled automated test facility using a dedicated Schwien manometer as the pressure source. Pressure versus frequency data is automatically taken at several temperatures and numerous pressures. This data is then curve fitted using a Rosemount derived pressure algorithm. The calibration coefficients of the algorithm are automatically computed and stored in ROM, resident in the sensor module.

All HADS modules are calibrated so the same set of equations is used to compute measured pressure. It is only the coefficients of these equations which vary from module to module. These equations are referred to as the HADS Algorithm, and consist of a set of polynomials that utilize the HADS module frequency, temperature and calibration coefficient signals. This results in complete interchangeability of the HADS pressure sensing module.

For a more detailed description of the HADS sensing module and definition of error sources, their effects and HADS compensation for them, see Ref. 1.

TRANSDUCER OPERATION

The external digital processing necessary for removal of repeatable errors and generation of a signal proportional to or some air data function of the measured pressure is accomplished by the HADS transducer. The transducer operations can be split into primary functions and secondary functions representing a modular design approach, complimented by the interchangeability of the HADS module.

The primary functions of the HADS transducer are measurement of HADS module signals, computation of pressure, and formatting and transmission of the output. The secondary functions of the transducer are computation of pressure-based parameters, BIT (Built-In-Test) capability and data manipulation (filtering, control functions and scaling).

HADS TRANSDUCER FUNCTIONAL DESCRIPTION

The 1501AT electronics receive a TTL compatible frequency signal (f), analog temperature (V_T) and reference voltage (V_R)

The circuit card assemblies positioned in the housing plug into individually keyed connectors interfacing with the motherboard assembly's connector. The motherboard assembly further interfaces with a front mounted electrical connector and the pressure sensing module.

The module assembly is mounted to a cast baseplate with an O-ring seal providing the pneumatic link with the transducer front plate.

This modular design approach provides minimum size and weight and offers easy fabrication and testing which results in simplified maintenance and reduced life cycle costs.

The internal construction is a simple, cost effective design. The basic structural members are a machined-cast aluminum baseplate, a structural stiffner and an aluminum welded sheet metal housing. This structure provides a lightweight, easily fabricated assembly which is sufficiently sturdy to withstand high performance aircraft operational shock and vibration specifications.

The mounting surface of the baseplate is treated with chemical film per MIL-C-5541, Class 1A, for electrical bonding in accordance with MIL-B-5087 requirements. This ensures a bonding resistance of less than 25 milliohms between the transducer and vehicle structure. The same treatment is also applied to each structural member within the transducer. All remaining exposed aluminum surfaces are coated with inert, fungus and corrosion resistant military qualified finishes.

<u>1501AT ENVELOPE</u>
The 1501AT is effectively smaller than a 3/8 short Air Transport Rack LRU (3/8 ATR LRU), per ARINC 404A. The LRU measures approximately 9 inches in length, 3.6 inches in width, and 4.75 inches in height, yielding a 154 cubic inch volume. Maximum weight of the 1501AT is 5.75 pounds (see Figure 6).

<u>MECHANICAL INTERFACE/INSTALLATION</u>
Transducer mounting is accomplished using captive socket screws as indicated in Figure 6.

The pneumatic pressure port is per MS33649-06 and is an integral part of the baseplate/front surface. Appropriate adaptors with O-ring seals can be threaded into the front surface making the required connections.

<u>OUTPUT FUNCTIONS</u>
The 1501AT is currently capable of two output configurations. The first is the ARINC 429 Digital Information Transfer System (DITS) defined for unidirectional information flow and currently in use in new generation commercial aircraft. This output has 32 BITS of information including parity, identification label, data, and status flags. The entire word is transmitted over a differential pair of lines, lending a high degree of noise immunity. The output is capable of driving up to six standard ARINC loads.

The second output format is a simple parallel bus containing 17 BITS of binary data, a sign bit, 2 identifier BITS, and a failure indication for the transmission of BIT results. Syncronization of the output is accomplished by a "Data Ready" line, which is asserted only when the data present on the bus is stable. This line is usable as a trigger for reading the most recent output data.

In both cases the output data is refreshed a minimum of once every 50 milliseconds.

<u>MAINTAINABILITY</u>
Each of the major electronic subassemblies are independent, interchangeable modules to the extent that replacement has no impact on performance or need for adjustment resulting in minimum effort required. The primary module is the pressure sensing module which, as described, is completely interchangeable. A common defined interface arrangement between the pressure sensing module and the transducer assures that one can change without impacting the other.

In a similar manner, the electronics are packaged so each circuit card assembly contains a defined set of transducer functions. Each of the circuit card assemblies are then considered modules which are defined levels for fault isolation. Each of these modules can be replaced by a like module without any need for transducer or system level calibration.

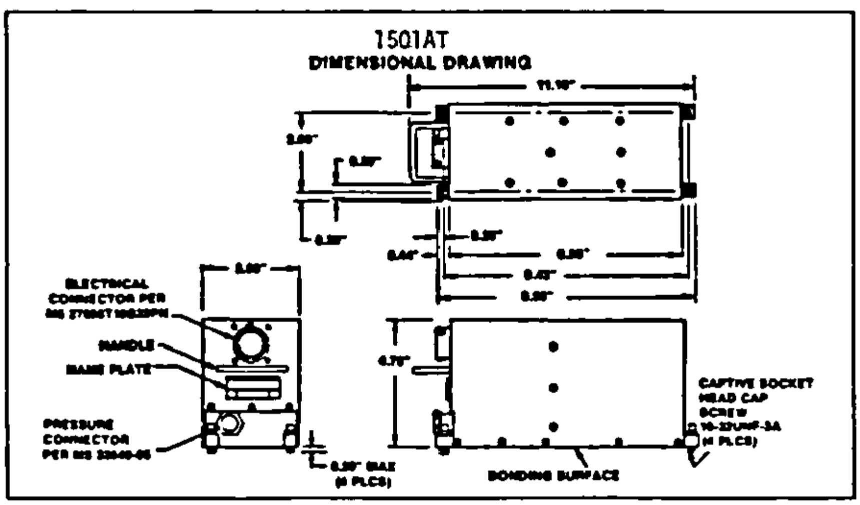
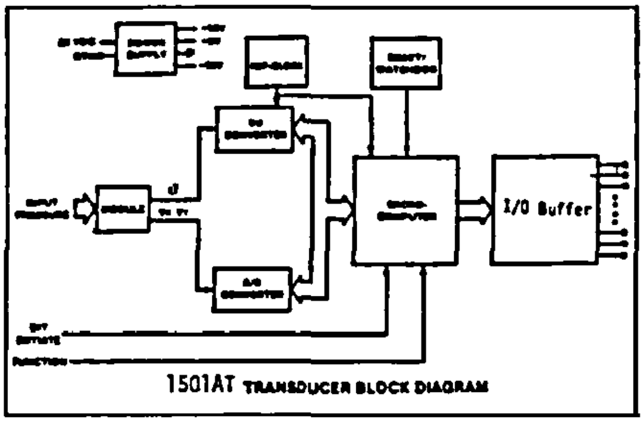


Figure 6

signals from the HADS module (see Figure 4). The analog signals are fed to a 12-BIT A/D converter. The TTL frequency signal is fed to a frequency to digital converter-period measurement circuit where a high frequency reference clock is then used to count a predetermined number of periods. The number of periods measured is optimized to provide 17 BITS of resolution within the update time of 50 msec. The module's characteristic calibration coefficients, determined for each module during calibration and stored in the ROM located within the module, are accessed by the transducer's 8-BIT Motorola 6809 microprocessor which performs the pressure computations utilizing the HADS Algorithm. Watchdog timer/reset circuitry for reset of program operation in the event of software upset is included. After required computations are performed, the resultant information is fed to the I/O circuitry which may be configured to output data in either a 17-BIT, parallel binary or ARINC 429 (high or low speed) format.



1501AT TRANSDUCER BLOCK DIAGRAM

Figure 4

ELECTRICAL CHARACTERISTICS

The 1501AT operates from an 18-32 VDC supply at a maximum power of 11 watts. Grounding is accomplished with a single point ground (power return connected to chassis) for high electromagnetic interference immunity.

SOFTWARE

The 1501AT employs software to control the electronic hardware used to acquire the frequency and temperature data from the HADS module, to compute the corresponding pressure, and to format the output and resulting data. This software is also capable of computing barometric altitude as a function of pressure. The software used is a set of program modules, each of which is written in assembly language. The use of assembly language allows a level of control in the software process which enables the device to operate with top efficiency.

Eighteen program modules have been included in the 1501AT. These include the basic mathematical functions (add, subtract, multiply and divide) as well as routines to perform power-on initiation, pressure computation, built-in test, numerical filtering, polynomial computation, output formatting, altitude computation, and others. These routines are controlled by and at the disposal of the executive routine, which interfaces with the hardware features of the device and defines the operation of the transducer.

The level of modularity incorporated in the device software allows flexibility in the transducer for the incorporation of special purpose functions and has allowed a high degree of commonality between the software packages used in various HADS products.

BUILT-IN TEST (BIT)

The 1501AT has both continuous and initiated BIT functions. Continuous BIT consists of continual functional monitoring by both hardware and software. Hardware monitoring includes circuitry to measure internal supply voltages for comparison with operating limits, and watchdog timer/reset circuitry (see Figure 4) for reset of program operation in the event of software upset. Software monitoring includes a continual comparison of computed pressure values to known operating limits.

During initiated BIT, comprehensive software tests are done. Checksums of all program and calibration (module) ROM's are compared to stored values, and repetitive read/write RAM tests are conducted. The initiated BIT mode is entered by inputting a logic "0" on the 1501AT transducer initiated BIT line. The transducer will respond to this input within 100 msec of initiation and return to the normal operating mode within 250 msec of its removal.

MECHANICAL

The mechanical assembly of the 1501AT is shown in Figure 5. The transducer consists of the following functionally designed modular assemblies:

1. Pressure Sensing Module
2. Pressure Transducer Circuit Card Assemblies (Shown)
3. Socket Mounted Software
4. Common Motherboard Assembly Which Includes Electromagnetic Interference (EMI) Circuitry

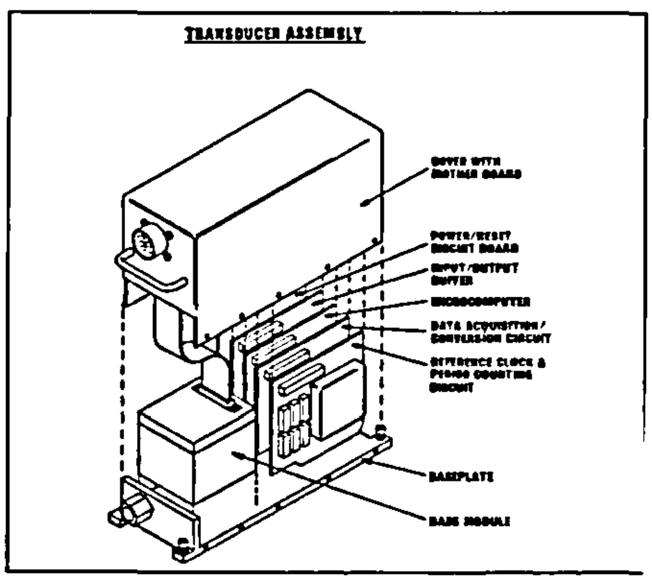


Figure 5

RELIABILITY

Rosemount has designed the 1501AT using high reliability parts to the maximum extent practical. The 1501AT reliability is expected to exceed 11,000 hours MTBF based on operational conditions in accordance with the Airborne Uninhabited environment defined in MIL-HDBK-217.

PERFORMANCE

Performance is broken into three categories: static performance, dynamic performance and long-term stability. Static performance is defined as the performance the pressure transducer will exhibit when not exposed to environmental stresses. Its component error sources are temperature, linearity, hysteresis, repeatability and calibration. Dynamic performance is defined as the performance the pressure transducer will exhibit in a highly stressed in-flight environment. Component dynamic error sources are defined as vibration, acceleration, power supply variations, overpressure and miscellaneous effects such as shock and case pressure.

Long-term stability is defined as the ability to reproduce output readings obtained during original calibration, for a specified period of time. A brief explanation follows with a summary of error tolerances presented in Table 1. For a detailed explanation of these error sources see Ref. 1.

STATIC ERROR SOURCES

The static accuracy reflects the combined effects of temperature, linearity, hysteresis, repeatability and Rosemount calibration. The component errors of temperature, linearity, hysteresis, repeatability and supplier calibration are grouped collectively as an error source rather than delineated seperately because of their interrelationships, which are typically ignored in sensor performance. Rosemount has determined that when dealing with accuracy tolerances of this magnitude, these interrelationships are no longer insignificant.

In order to effectively correct such errors in a manner simple enough to be efficient at both the calibration and user system level, a calibration technique has been designed in which all repeatable errors are corrected concurrently. This is performed by taking a significant number of calibration points, changing all variables (e.g.: pressure and temperature) and developing a multi-dimensional polynomial in these variables to solve for measured pressure. Thus, the previously discussed HADS algorithm relates and incorporates the combined effects of temperature, linearity, hysteresis, repeatability and calibration at Rosemount.

The maximum 1501AT static error is .021% of full scale pressure. This includes transduction errors resulting from transducer circuitry which measures module outputs and from finite computation resolution. Altitude conversion error is an additional ±3 feet.

DYNAMIC ERROR SOURCES

In pressure based air data systems, bench top measurements of accuracy on the ground are not necessarily indicative of in-flight performance. Typical in-flight environmental characteristics of vibration, acceleration, power supply variations, and other miscellaneous effects are difficult to simulate simultaneously on the ground and hence are typically ignored. These error sources, however, can be significant in overall operational accuracy of an air data transducer and need to be considered carefully for their individual and collective effects. The 1501AT has been designed to provide the performance stated in Table 1 while experiencing the dynamic conditions listed. This is important because these dynamic conditions will always be present in one form or another during operational flight testing. The HADS shock and vibration protection system previously discussed was developed in light of these imposed dynamic conditions to minimize the transmissibility normally associated with conventional methods (i.e.: foam or rubber mounts). See Table 1 for dynamic error tolerances of the 1501AT.

TABLE 1

1501AT PERFORMANCE SUMMARY

	3σ Corrected Error (% of Full Scale)*
Static Error	
Linearity	
Hysteresis	
Temperature (-55°C to +71°C)	±.021%
Repeatability	
Transduction	
Total Static Error	±.021%
Dynamic Error	
Vibration (10 g's RMS)	±.010%
Acceleration (6 g's Max)	±.003% per g
*Overpressure (150% FSP Max)	±.003%
Miscellaneous (Shock, Case Pressure)	±.003%
Long-Term Stability (12 Months)	±.025%

* Full scale pressure (FSP) is defined as the calibrated pressure range.

* Defined as maximum calibration shift as a result of overpressure within specified limits.

LONG-TERM STABILITY ERRORS

Stability is additive with static error and implies that the calibration will not change more than .025% of full scale pressure in a one-year period. Stability error will be a unidirectional shift in calibration.

PERFORMANCE VERIFICATION

The HADS device has existed in product form since 1979. During this time the product has been tested against performance claims in several different manners.

First, all performance parameters and characteristics are based on performance testing begun on the initial HADS R and D devices and re-tested in ensuing stages of product development.

In 1979, HADS was chosen as the Primary Air Data source for the Grumman Aerospace Corporation (GAC) X-29A Forward Swept Wing technology demonstrator. As a team member on the X-29A program, Rosemount has developed two unique configurations of the 1501AT, which have successfully completed flight worthiness testing. This consisted of comprehensive static and dynamic performance tests including temperature (-55°C to +71°C and 95°C intermittent), vibration (10g's RMS), shock (15g's) and acceleration (6g's). Further testing was conducted which verified performance in severe electromagnetic interference (EMI) environments (MIL-STD-461A, Notice 3; MIL-STD-462, Notice 2 in addition to Transient Radiated Interference Susceptibility).

Also as a part of the 1501AT flightworthiness effort, Rosemount was required to provide equipment to be flight tested by Grumman in order to develop confidence on the products viability under operational flight conditions. In 1981, two flight test transducers were produced and tested under laboratory conditions by Grumman. These units were installed in an F104G test aircraft. (See Figure 7.) Several flight tests were flown. During these flights the HADS outputs were recorded and compared with reference measurements. Results of this testing have been documented and indicate HADS is equal to or better than the reference in all aspects tested.

1581A DUAL CHANNEL TRANSDUCER

A prototype dual channel 1581A is now undergoing testing. The dual channel 1581A contains two absolute pressure sensing modules and incorporates the same modular design features and benefits as the 1501AT. The added benefit is the addition of a Qc output. (Altitude and IAS will also be available.) Qc is determined by computing the difference between measured Pt and Ps. The initial output format provided is ARINC 429, with 1553B to follow. The 1581A will use a captive mounting scheme with the same cross-sectional dimensions as the 1501AT, increasing in length by only three inches. This envelope maintains a size capable of mounting in a 3/8 ATR short LRU.

OUTPUT FORMATS

The 1501AT transducer is currently capable of 17 BIT parallel binary or ARINC 429 outputs. The modularity of the 1501AT transducer electronics allows for easy implementation of alternative output formats. The 1501AT can be equipped with defined, user unique formats in addition to more standard output formats such as BCD, CMOS and 1553B. With this ability, the 1501AT can interface with already defined flight test data acquisition system formats providing product level interchangeability.

CONCLUSION

When in-flight performance is critical to the success of a program, the requirement for accurate, reliable flight test data is increased. Similarly, as the cost per flight test hour increases, effective, efficient and timely use of available aircraft necessitates use of state-of-the-art flight test equipment that is both maintainable and reliable.

It has been the purpose of this paper to technically describe Rosemount's digital pressure transducers for today's flight test needs. By compensating for errors normally neglected, the 1501AT provides reliable, accurate performance under operational dynamic environments present during flight. The modular design approach ensures simplified maintainability. Proven flight test performance indicates the 1501AT digital pressure transducer has the technical edge required for today's advanced flight testing of new generation aircraft.

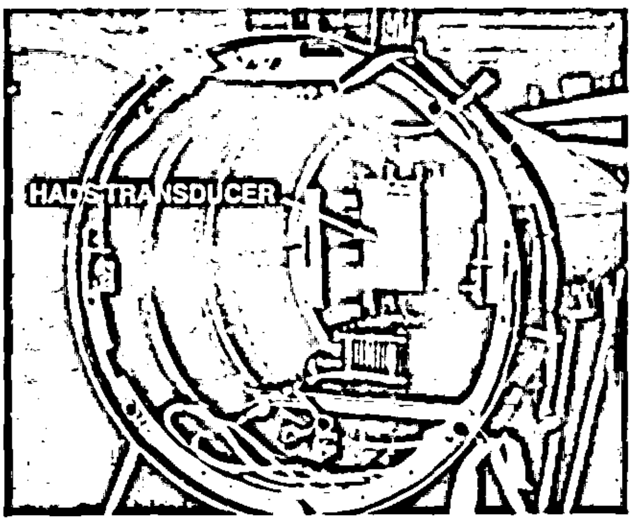


Figure 7

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4. Rosemount Product Data Sheet 2422.

USES OF A DIGITAL ELECTRONIC THEODOLITE SYSTEM IN A WEAPON SEPARATION PROGRAM

by
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Abstract

A digital electronic theodolite system has been incorporated to measure target locations used in the weapon separation program. What once required several work shifts to accomplish can now be done in one 8-hour shift. In addition, significant improvements in accuracy and reduction in cost occurred, not only related to measuring, but also to the actual computations involved in the weapon separation.

Introduction

An important aspect of any attack aircraft flight test program is the weapon separation program. This involves the development of a safe weapon deployment envelope. Our standard procedure for developing this envelope consists of filming the release of weapons during a number of flight configurations and maneuvers. The film is then processed and digitized. The digitized values are input to a weapon separation software program to obtain the required weapon displacement. Using the displacement results of a number of weapon separation flights, we finally develop the deployment envelope.

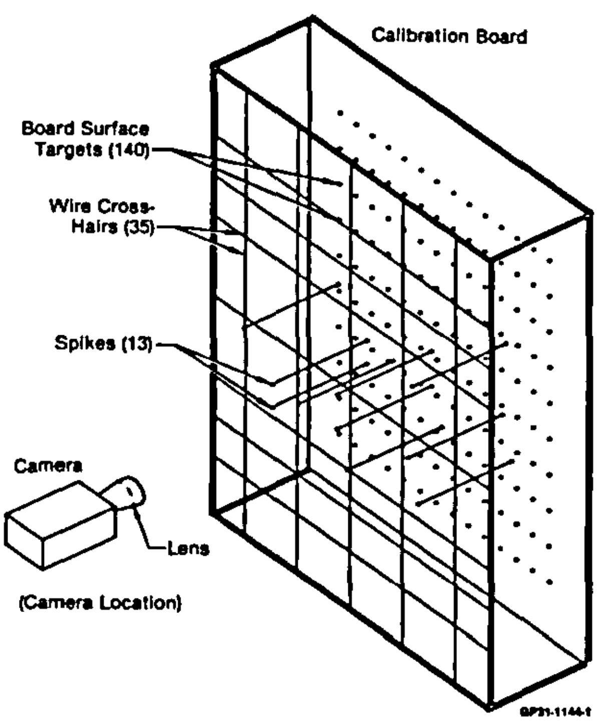

FIGURE 1. VIEW OF CALIBRATION BOARD

Although the actual software logic involved in obtaining the weapon displacement is not a critical issue, a basic understanding of what is involved is essential to grasp the value of the digital electronic theodolite system. Obtaining weapon separation data consists of two major parts:

(1) Calibrating the camera-lens system, and

(2) Processing the weapon separation film data.

To calibrate the camera-lens system, a calibration board is used, as shown in Figure 1. Using spike projections from the board and a frame of film of the camera viewing the board, the camera location (center of the lens) can be calculated. Once the camera is located, the lens is calibrated using a curve fit routine and the targets at the surface of the board. Accurate target and spike tip locations are essential for a good camera lens calibration.

To process the weapon separation film data, weapon separation software is used. This software, which yields the final weapon displacement information in aircraft coordinates, must compute two locations per frame of film data:

(1) Location of the camera with respect to the aircraft coordinate system, and

(2) Location of the weapon with respect to the camera coordinate system, (Figure 2).

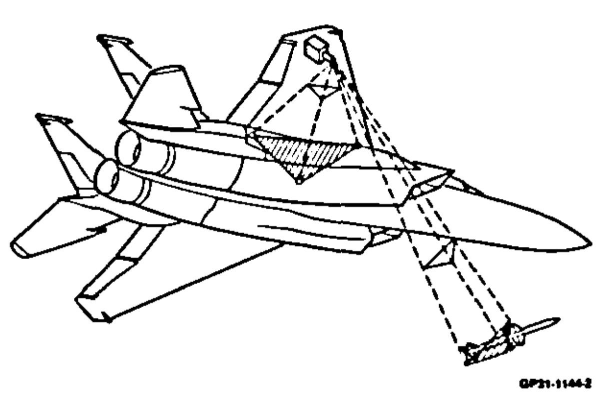

FIGURE 2. VIEW OF BORESIGHT TARGETS AND WEAPON TARGETS (ILLUSTRATES IDEA USED TO LOCATE CAMERA AND WEAPON)

Once these locations are computed, the weapon location is simply transformed and translated to aircraft coordinates. The actual technique used to calculate these various locations incorporates the use of aircraft target points and the previously described lens calibration.

The approach uses three aircraft targets (boresight targets) to compute the camera location in aircraft coordinates and three weapon targets to compute the weapon location in camera coordinates (Figure 2). The accuracy of these locations, and thus the accuracy of the weapon displacement calculations, is influenced by three parameters:

(1) The digitized (x,y) values of the above six targets from each frame of the weapon separation film,

(2) The (x,y,z) values in aircraft coordinates of the boresight targets and the (x,y,z) values in weapon coordinates of weapon targets, and

(3) The lens calibration.

Because of the sensitivity of the software to small measurement errors, the accuracy of these parameters is crucial. Sufficient error will cause calculated locations to converge to incorrect solutions. (Illustrated in the Results section under Weapon Displacement.)

Since the digitized values are obtained from a Benson-Lehner Teleradex film reader, any related error is attributed to the operator. This error is assumed a constant and defined as the thickness of the wire cross hairs on the digitizer, about 11 counts out of 6500 counts or approximately .16%. Since this error is extremely small, no effort was made to improve the digitizer.

Therefore, improving weapon separation results was achieved by focusing attention on the other two parameters. These two parameters both involved the use of target coordinates; therefore, the objective was to more accurately locate the targets.

Prior Methods

Prior methods used for measuring target locations consisted of blueprints, loft information, and transits (surveying telescopes). Before any measurements could be made on an aircraft, it had to be leveled. Bolts located on the landing gear were used to accomplish this tedious task. Large 15-foot tooling bar segments were then placed around the aircraft. The bars were connected together and were used to mount the transits. Each bar had holes at increments of 5 inches into which the transits could lock. By locating the bar position in the aircraft coordinate system using three reference targets, each hole was located, and thus so was each transit.

Next, a target was viewed at various predetermined locations along the bar. The vertical and horizontal angles from the transit were recorded at each location. By using these angles, the theodolite locations and the methods of triangulation, the x,y,z coordinates of a single target point were computed. All additional targets were located in a similar manner. The actual computations were not performed until later but required hours of hand calculations and rechecking to avoid calculation errors.

New Methods

The new method used for improving the target location measurements for both the calibration board and aircraft incorporates a Hewlett-Packard (HP) digital electronic theodolite system. This system uses two theodolites (surveying telescopes) and an HP minicomputer (Figure 3). The theodolites are linked to the minicomputer and can measure the vertical and horizontal angles of a viewed target point. The four angles (two from each scope) are stored and later used to compute the target's (x,y,z) location. The theory behind this will be discussed shortly. Though the system is similar to the old method, the capabilities of the digital electronic theodolite system significantly surpass those of prior methods.

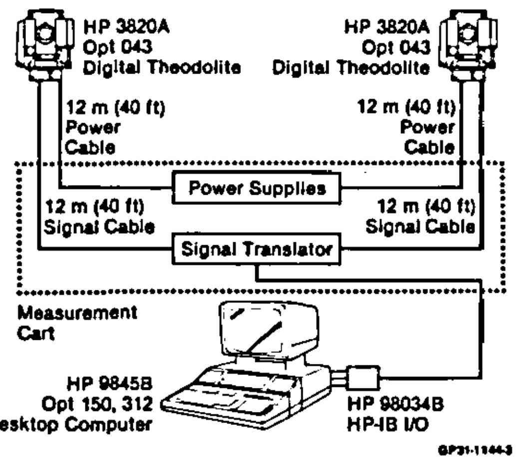

FIGURE 3. THEODOLITE SYSTEM

To use the system, a minimum of two operators is required, one for each theodolite. The computer can be operated by one of these two people or a third person. When measuring positions on an object, for instance the calibration board or aircraft, the theodolites are positioned to maximize the following goals:

(1) Create a large triangle defined by the two theodolites and a measured target, and

(2) Locate theodolites to incorporate all or most of the targets into the field-of-view of both theodolites.

For the case of an aircraft, theodolites are positioned on one side of the aircraft, and then the other side.

The first requirement for locating targets is to compute the line of reference from one scope (1) to the other scope (2). This involves determining the horizontal and vertical angles from scope (1) and scope (2) and is accomplished with back sighting. Back sighting is the simple process of a scope viewing itself with the use of the other scope. A small mirror inside the second scope reflects the image of the viewing scope back to itself, and the viewing scope aligns its cross hairs with the reflected image cross hairs. Back sighting is done concurrently with both scopes. The resulting angles from the scope yield the reference line.

With the reference line defined, the distance between the theodolite scopes is measured. A bar of known length is used. By viewing two points of known distance on the bar with the two scopes, the distance between the scopes is calculated.

With these two steps completed, targets points are ready to be located. To locate a target, the target is viewed through both scopes and the resulting four angles are recorded. As mentioned earlier, triangulation is then used to compute the position of the target. Target locations are defined in a theodolite coordinate system where scope (1) is the origin and the reference line is the X-axis. When all targets have been measured, the values are transformed and translated to the desired target board or aircraft origin using the known locations of three reference targets on the calibration board or aircraft.

Theodolite Theory

The theory used to determine target point coordinates incorporates the principle of triangulation. The angles involved are the vertical and horizontal angles (θ_1, θ_2, Φ_1, Φ_2) measured by theodolites 1 and 2.

First, the line of sight from a theodolite to point P (the point being located) is considered a vector. Because of the ease of small operator errors, the two vectors may not intersect at point P, or at any point for that matter. Therefore, the yielded solution will be the mid-point of the shortest distance between the two vectors, which will be very close to P. The end points to this shortest distance are points G and H. See Figure 4.

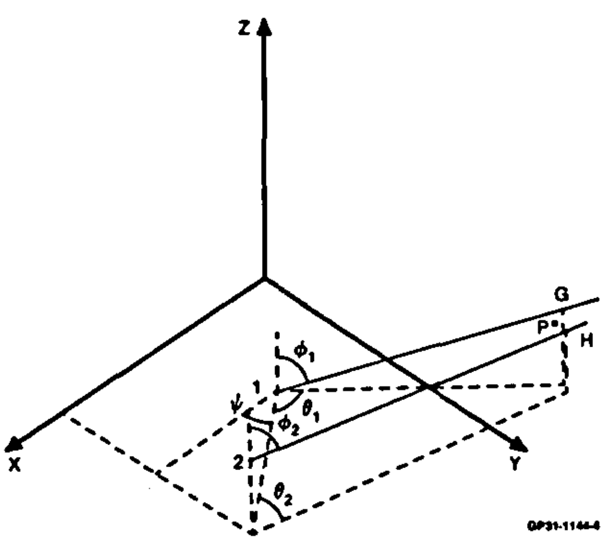

FIGURE 4. APPROXIMATION FOR POINT P

Next, using the angles from the theodolites, the two vectors can be defined by their direction cosines, the angles between the vector and three coordinate axes. This is done by knowing two points on the line. Point (a) can be the theodolite location while point (b) is computed from Θ, Φ, Ψ (known) and (a). The direction cosines are defined as:

$$\cos\alpha = l = \frac{x_a - x_b}{D}$$

$$\cos\beta = m = \frac{y_a - y_b}{D} \qquad (1)$$

$$\cos\gamma = n = \frac{z_a - z_b}{D}$$

Where $D = [(x_2-x_b)^2+(y_a-y_b)^2+(z_a-z_b)^2]^{\frac{1}{2}}$. See Figure 5.

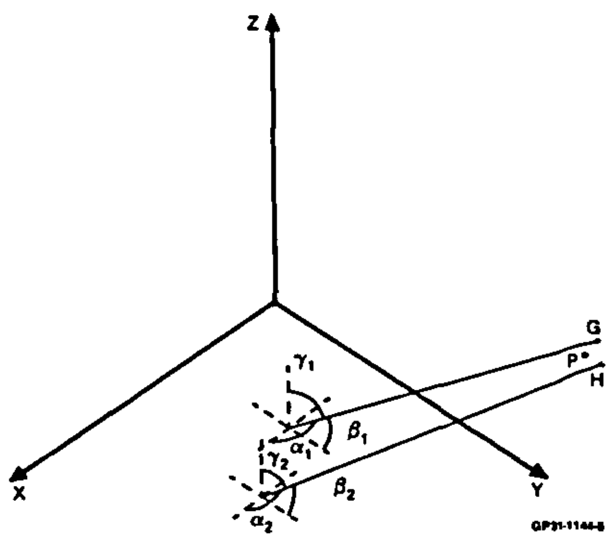

FIGURE 5. DIRECTION COSINES OF LINES

Equations (1) can be combined to yield:

$$\frac{x_a - x_b}{l} = \frac{y_a - y_b}{m} = \frac{z_a - z_b}{n} \qquad (2)$$

Equation (2) can be rewritten to yield the coordinates for points G and H:

$$y_G = \frac{m_1}{l_1} * (x_G - x_1) + y_1$$

$$y_H = \frac{m_2}{l_2} * (x_H - x_2) + y_2$$

$$z_G = \frac{n_1}{l_1} * (x_G - x_1) + z_1 \qquad (3)$$

$$z_H = \frac{n_2}{l_2} * (x_H - x_2) + z_2$$

Where 1 and 2 are theodolites 1 and 2 respectively.

The additional two equations available from equation (2) are linearly dependent to equation (3) and therefore cannot be used. Since six unknowns exist, two more equations are required. The distance between points G and H is defined as:

$$D^2=(x_G-x_H)^2+(y_G-y_H)^2+(z_G-z_H)^2 \qquad (4)$$

Since y_G, y_H, z_G, z_H are functions of x_G and x_H, D is simply a function of x_G and x_H.

$$D^2=(x_G-x_H)^2+\left[\frac{m_1}{l_1}(x_G-x_1)+y_1-\frac{m_2}{l_2}(x_H-x_1)-y_2\right]^2+\left[\frac{n_1}{l_2}(x_G-x_1)+z_1-\frac{n_2}{l_2}(x_H-x_1)-z_2\right]^2 \qquad (5)$$

If the partial derivatives of D with respect to x are set equal to zero, the function will be minimized, resulting in the shortest distance between the two lines. The two additional equations which appear are:

$$\frac{\delta D^2}{\delta x_G}=0=\frac{x_G}{l_1}-\frac{Sx_H}{l_2}+m_1\left[\frac{m_2x_2}{l_2}-\frac{m_1x_1}{l_1}+y_1-y_2\right]+n_1\left[\frac{n_2x_2}{l_2}-\frac{n_1x_1}{l_1}+z_1-z_2\right]$$

$$\frac{\delta D^2}{\delta x_H}=0=\frac{Sx_G}{l_1}-\frac{x_H}{l_2}+m_2\left[\frac{m_2x_2}{l_2}-\frac{m_1x_1}{l_1}+y_1-y_2\right]+n_2\left[\frac{n_2x_2}{l_2}-\frac{n_1x_1}{l_1}+z_1-z_2\right] \qquad (6)$$

Where $S=l_1l_2+m_1m_2+n_1n_2$

Combining equations (3) and (6) gives the required six equations. These equations are then solved to yield the (x,y,z) coordinates of points G and H. By averaging these values, the approximate coordinates for point P are determined.

To improve upon these coordinates, a least squares adjustment from the four original angles (θ_1, θ_2, ϕ_1, ϕ_2) can be determined and applied to the three unknown coordinates. This, however, is beyond the need of this paper and is omitted from this discussion.

Results

The new digital electronic theodolite system was used to measure the targets on the camera-lens calibration board and three F-18 aircraft used for weapon separation (TF1, F4, F7). An average of 38 boresight targets was measured on each aircraft. Table 1 shows the difference between the original and new target coordinates for various aircraft targets. Though the differences are less than an inch in any axis, this error is greatly magnified when incorporated in displacement calculations, as will be seen shortly.

TABLE 1. DIFFERENCE BETWEEN OLD AND NEW TARGET LOCATIONS

Target	ΔX	ΔY	ΔZ
1	−0.4704	−0.2684	0.7446
2	0.0719	−0.1130	0.4657
3	0.4487	−0.0321	−0.2161
4	0.0011	0.0037	−0.0149
5	−0.0350	−0.0645	−0.1602
6	−0.0047	−0.0044	0.0177
7	0.6289	−0.0561	0.3619
8	0.0435	−0.0437	0.5566
9	−0.2311	0.0670	−0.1918
10	0.1247	0.1083	−0.0387

Data from aircraft F18 – TF1. Differences in inches.

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Once the calibration board and aircraft targets were measured, the resulting target coordinates were stored in the corresponding software files. A number of significant improvements resulted.

Improved Lens Calibration

As mentioned earlier, the calibration board is used to correct the distortion in the camera lens and to help develope a curve fit equation relating digitized target values with physical targets. To use the board, the camera is placed in a fixture viewing the board, and a frame of film is shot. The developed frame of film is placed in the digitizer and the 188 targets (140 board targets and 48 spike tips) are read. These values along with their corresponding physical coordinates are used to compute two parameters:

(1) Camera location in board coordinates, and

(2) Curve fit calibration.

The prime importance of this calibration is the calibration equation. The accuracy of this equation is directly related to the accuracy of the camera location, which in turn is directly related to the accuracy of the targets' physical locations. Table 2 gives a brief summary of significant variables from the calibration before and after the theodolite measuring.

TABLE 2. COMPARISON OF LENS CALIBRATION RESULTS

Variable	Results From Old Measurements	Results From Theodolite Measurements
Camera Distance From Board	97.3 Inches	107.1 Inches
Difference Between Measured and Calculated Camera Distance*	9.2%	0.1%
Standard Deviation of Curve Fit Equation	0.020 Deg	0.018 Deg
Image Distance**	14.743 mm	16.235 mm

* Hand measured camera distance was 107.2 Inches.
** A 16.0 mm lens was used.

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As the table shows, the calculated camera location is almost identical to a hand made measurement when the new values are used. The standard deviation of the calibration equation decreased, as was hoped. In addition, the computed film image distance was slightly larger than the focal length of the lens, as it should be. Overall, using the target coordinates obtained from the digital electronic theodolite system greatly improved the lens calibration technique.

One interesting observation from the theodolite measurements was the discovery of a previously undetected warp in the calibration board. This warp, which was approximately .5 inches maximum at the edges, was outward (the edges warped away from the camera). This warp in turn caused the spikes to warp outward. Because of the warp, the camera would see the board closer than it really was, as the table shows. Though the board need not be warp free to yield a good calibration, it is important to know that the board has a warp. The theodolite system yielded that fact.

Reduced Camera Wing Motion Scatter

An important step in computing weapon displacement is the calculation of the camera location on the aircraft during the separation flight*. Any error in defining camera location during the flight is carried through in defining weapon displacement.

TABLE 3. STANDARD DEVIATION OF CALCULATED CAMERA LOCATIONS

Before Theodolite System		After Theodolite System	
Flight*	Deviation (inches)	Flight*	Deviation (inches)
8	2.9	222	2.3
20	2.4	222	1.2
23	1.2	223	0.84
46	0.5	224	0.53
115	0.13	239	3.2
115	1.0	303	1.2
160	7.8	303	1.2
160	7.1	309	5.7
162	1.2		
Average	2.7	Average	2.0

*Repeated flight numbers either used a different camera or recorded a different weapon for that flight

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For wing located cameras, the camera location is computed once for each frame of the weapon drop (approximately 100 frames) using three boresight targets and the lens calibration. Because the wing of the aircraft will flex at the time of release from the reduction in weight, camera movement will occur during separation. This motion should, however, be smooth. Any large scatter (error) in the computed camera location then would have to be related to digitizing and/or the target location coordinates. This scatter is computed using the method of standard deviation. Therefore, to compare the new target locations with the old ones, the standard deviation of the camera location was used. Since the incorporation of the theodolite system took place half way through the F-18 separation program, the comparison was made between data before and after the introduction of the theodolite target values. Table 3 gives the comparison of standard deviations of some typical flight data.

Table 3 indicates that on the average, the standard deviation of the calculated camera location decreased for the new target location values. Since digitizing error remained constant, this decrease in average standard deviation is attributed to improve target definition.

Improved Weapon Displacement Accuracy

Since the camera location has a direct influence on weapon displacement calculations, the improvement in camera definition just shown is partly responsible for the improvement in weapon displacement. The additional improvement, however, is from the lens calibration.

The weapon's location for each frame of film is defined with three weapon targets*, the lens calibration, and the camera location. As shown in the discussion related to the lens calibration, the improved target values significantly improved the distance computed between the camera and the board. This same improvement took place for the distance between the camera and the weapon targets. This improvement is depicted in Figure 6 for

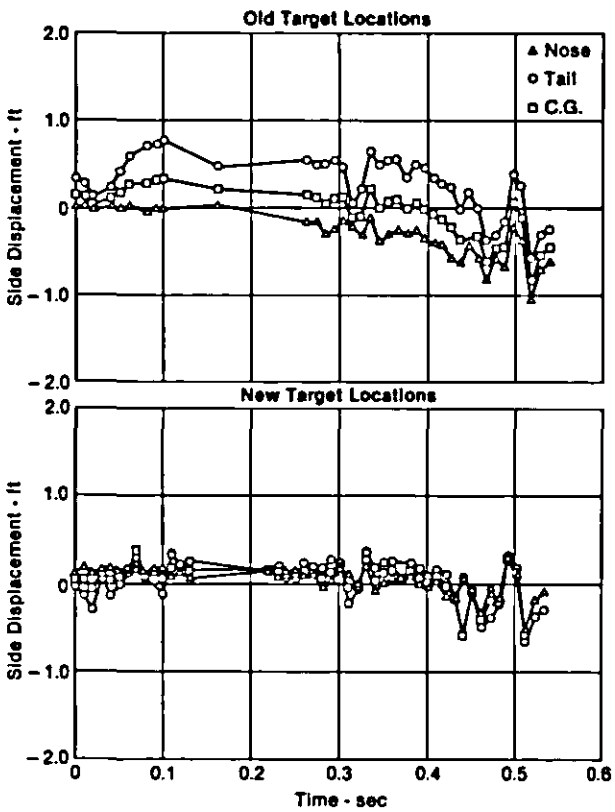

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FIGURE 6. COMPARISON OF WEAPON SIDE DISPLACEMENT

*This camera location is different from the one computed for the lens calibration.

side displacement where a weapon drop was processed using both sets of a target values.

Figure 6 shows the motion of the nose, tail, and C.G. of the weapon with respect to time. The effects of the errors discussed in Tables 1, 2 and 3 are illustrated in Figure 6.

Though not illustrated in this example, the computed location of a target occasionally converges to a point which is obviously not correct. This results from the previously mentioned errors. When such a problem occurs, additional software is used to statistically compute and subtract out some of the error in the target locations. This method requires additional passes through the computer for the same job, a costly and time consuming process. With the new target values and lens calibration, the need for this follow-on software has been reduced, since the convergence to wrong locations is not nearly as prevalent as before.

Reduced Man-Hours

An additional bonus of the digital electronic theodolite system is its ease of use. Because the system can be operated by two persons, is easily transported, and does not require a leveling of the aircraft, a sizable reduction in man-hours occurs. Table 4 compares the man-hours used for measuring the three F-18 aircraft with the theodolite system to the man-hours the old method would have required.

TABLE 4. COMPARISON OF MEASUREMENT METHOD FOR LOCATING BORESIGHT TARGETS

Aircraft	Theodolite Measurement Time (hr)	Equivalent Man-hours	
		Theodolite	Conventional
1	9.00	27.0	125
2	8.75	17.5	122
3	10.75	21.5	150

Theodolite system required approximately 17% of the man-hours of the conventional methods.

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This table reflects only the reduction in man-hours from the measuring of the target points. Though not determined, additional reduction took place in the actual computation of the locations. Where as the old methods required numerous hours of hand calculations, prone to calculation errors, the digital electronic system had three axes coordinates in the aircraft reference system available at the close of the measuring, free from any calculation errors.

*The weapon targets were not measured with the digital electronic theodolite system, since they are installed on each weapon just prior to flight using a weapon target fixture.

Summary

With the incorporation of the digital electronic theodolite system, a number of improvements have resulted.

- Reduced time in measuring targets and computing weapon displacements.
- Reduced efforts and errors in triangulation calculations.
- Increased accuracy in measuring targets and computing weapon displacement.
- Reduced cost for overall process.

Future Uses

Planned future uses or potential uses for this system are:

- Aircraft structural tolerance checks
- Surface calibrations
- Tooling hardware verification
- Standard reference point positioning.

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CONTINUED DEVELOPMENT OF DISTANCE MEASURING EQUIPMENT FOR REAL-TIME SPATIAL POSITIONING IN MILITARY AIRCRAFT TESTING

by
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Abstract

This paper discusses the development of a cost-effective microwave spatial positioning system by Lockheed-Georgia Company. The program is an ongoing effort to develop space positioning techniques and real-time spatial data processing applicable to existing performance, stability and control, and systems flight test programs. A cost-effective research and development program has been implemented by exploiting recent advances in microwave positioning technology and utilizing in-house advances in microprocessor applications. Results from ground calibrations and accuracy testing have been very encouraging, although several major problems were encountered. The feasibility of a portable, ground crew independent, real-time spatial positioning system has been successfully demonstrated. Actual Takeoff and Landing (TOL) testing using a laser tracker as a baseline revealed that utilization of the calibration techniques can significantly improve the accuracy of a microwave positioning system. Current and future development will continue in the areas of transformation error minimization and antenna characteristics analysis. This paper relates the cost-effective techniques used in developing this spatial positioning system, while attempting to achieve the highest possible accuracies.

Introduction

An important part of flight testing includes measuring spatial position during takeoff and landing (TOL) performance testing and airspeed calibrations. Evaluation of the extremes in military aircraft TOL performance requires data from various runway altitudes, pavement conditions (dry, wet, or icy), and soil conditions (off runway testing). This entails the need to conduct many TOL tests at remote runway locations. The development of a TOL system that is portable, not dependent on ground crews, capable of in-flight analysis (real-time analysis), and inexpensive to develop and maintain will greatly facilitate both military and civil test programs.

There are many ways to determine an aircraft's position and velocity. Most of the existing methods can be classified as photographic, electronic, inertial, or laser. The majority of the photographic processes are time-consuming and not cost-effective. Inertial and laser systems are very accurate, but are costly to implement. An inertial platform system capable of measuring TOL performance is priced in excess of a quarter of a million dollars (1982), and the purchase of a precision laser tracking system would approach one and a half million dollars (1982). Laser systems can be leased; however, availability is not always coincidental with a test program's schedule and lease costs can exceed $10,000 (1982) per week. An alternative to obsolete photographic processes and costly inertial or laser systems is the utilization of electronic microwave systems.

Implementing microwave distance measuring equipment for TOL performance and airspeed calibrations is not a new concept. Although there are times when the accuracies of inertial and laser systems cannot be sacrificed, microwave techniques have been proven to be of sufficient accuracy in many instances. In 1976 and 1978, Franklin Schick of the Gates Learjet corporation reported on a microwave system implemented for the FAA certification performance testing of two separate Learjet aircraft. The Gates Learjet microwave system consisted of recording the aircraft's position on magnetic tape as measured from the aircraft antenna to one remote antenna situated at the end of the runway. Post-flight processing prepared data for analysis in a matter of hours. An estimated hardware cost for this TOL system is less than $50,000 (1982), which is

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**Senior Engineer, Aircraft Instrumentation

extremely low when compared with alternate systems. The system as configured does not have real-time analysis or spatial positioning capability.

The techniques were further refined by Grumman Aerospace Corporation for Gulfstream III testing. Joe Kudirka[2] reported these developments at the 1982 SFTE National Symposium. This method involved the transmission of data with Grumman's Automated Telemetry Station as opposed to using magnetic tape and post-flight processing. Portability ground crew independence, and spatial positioning were not achieved, although analysis was.

The unique requirements of military testing dictate additional system enhancements to obtain an optimum system which can be used for both military and civil testing. Lateral runway dispersions are needed when determining short takeoff and landing(STOL) and vertical takeoff and landing(VTOL) runway performance, developing an all-weather landing system, or evaluating the effects of engine loss during takeoff. True portability without dependence on ground support teams would reduce test support requirements. Furthermore, detailed static and dynamic error analyses are warranted to determine optimum system calibration and operating techniques.

System Overview

The Microwave Spatial Positioning System (MSPS) is comprised of three main sections; a main control computer, a Minneapolis Honeywell radar altimeter, and a Del Norte TRISPONDER† system. The MSPS interfaces with an aircraft and the Lockheed Airborne Data System[3] (LADS) as shown in Figure 1.

The first element of the MSPS is the control computer, a microprocessor based computer developed expressly for this function. The LADS remote computer issues a scan command to the control computer which then sends an external update command to the distance measuring unit (DMU) and reads radar altitude at the rate of ten per second. The DMU then measures the distance between the master and remote transponders. The time necessary to acquire each slant range is measured and is used to determine data validity. When both slant ranges and altitude are valid, the control computer performs a real-time calculation to convert these values to true cartesian coordinates using a combination of triangulation and a table of runway correction coordinates.. The control computer then

†TRISPONDER is a registered trademark of Del Norte Technology, Inc.

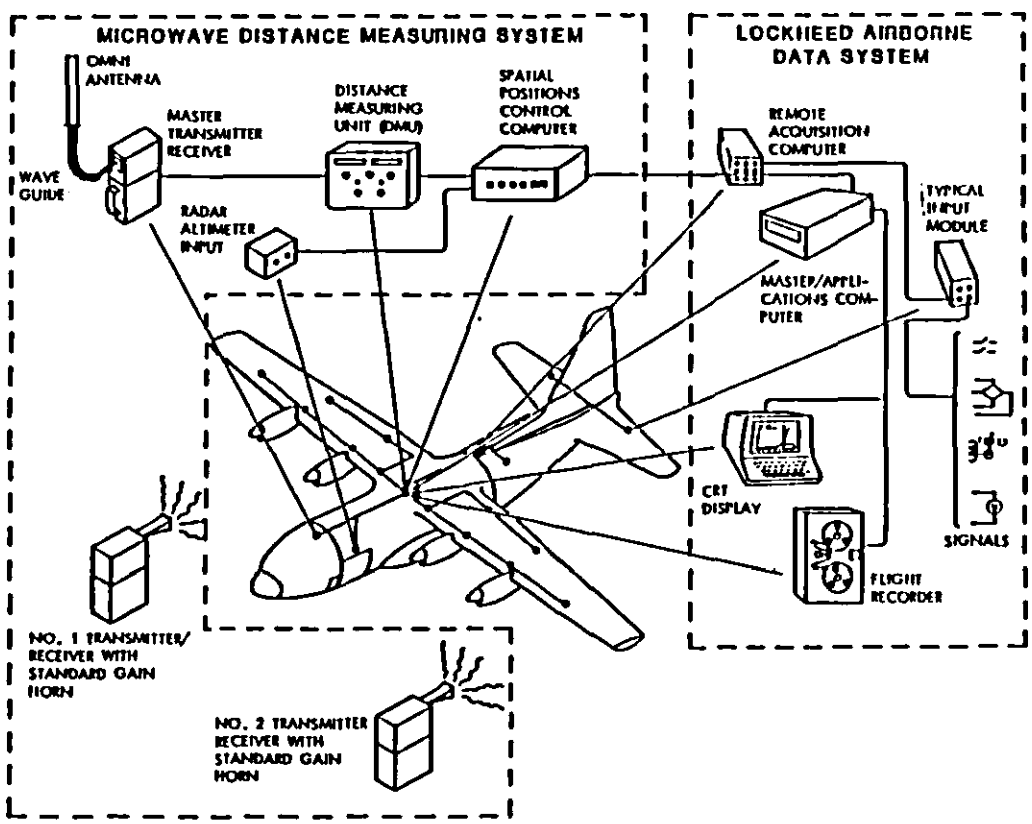


Figure 1 The MSPS/LADS interface

reformats the data for compatibility with the LADS remote controller. These data are output to the remote controller on the next scan command, and the cycle repeats.

The second component of the MSPS is a Minneapolis Honeywell radar altimeter. The output from this device is an analog voltage which is directly proportional to the aircraft's true altitude above ground. The radar altimeter used during the program was valid for an altitude of up to one hundred feet.

The third element is the Del Norte Trisponder system, which is composed of a model R04A Distance Measuring Unit (DMU), a model 261 master transponder and two model 261 remote transponders. The remote and master transponders are similar with the exception of a few control functions. The master transponder is commanded to transmit to one of the two remote units. When the selected remote receives the proper signal, it transmits a return signal to the master transponder. The cycle is then repeated for the second remote.

The DMU controls the operation of the transponders and converts the return times into distance. It receives a command from the control computer and directs the Master transponder to transmit to the first remote unit. Simultaneously, it starts an internal timer which continues to run until the return signal from the first remote is detected by the Master transponder. This process is continued until one hundred good return signals are received from a maximum of up to 200 returns. The elapsed return time is then converted to an average distance and output to the main control computer. The cycle is then repeated for the second remote.

Hardware Design

The main control computer is based on a Texas Instruments TMS 9995 Microprocessor which is a 16-bit internal/ 8-bit external bus machine. There are 8192 words of program memory and 2048 words of data memory. There is one command input channel, one analog input channel, two parallel input channels, and one serial output channel.

The altitude above the runway is produced by the Honeywell radar altimeter as an analog voltage which is measured and converted by a 12-bit analog to digital converter (ADC). The digital altitude data is read by the processor as the least significant bits of a 16-bit memory-mapped word. The upper four bits are used to indicate if the Trisponder has valid range data and on which channel the data is available. The slant range data is read through a memory-mapped word in Binary Coded Decimal (BCD) format. The second parallel port reads a BCD-coded thumbwheel switch which is used to preset the runway offsets and transponder locations.

The data that is read and processed by the control computer must be reformatted and output in a format compatible with the LADS remote acquisition computer. Eight 16-bit data words must be output serially in less than 96 microseconds in synchronization with the scan request from the acquisition computer. This high data rate is too fast to be implemented with the microprocessor serial output channel, so it is handled by hardware external to the processor. The eight words are stored in a first in first out (FIFO) memory after all processing is complete in anticipation of the next scan request. When the scan request is received a six bit counter is cleared. This counter is incremented by a one Megahertz clock and addresses a programmable read only memory (PROM). This prom contains the control sequence necessary to shift data from the parallel FIFO into the parallel-to-serial converter and shift out data in a synchronous serial format. The prom also contains the stop code which disables the counter after all data is output. The counter remains in this suspended state until the next scan request is received, when the cycle is repeated.

Real Time Software Design

The software to determine the aircraft coordinates in real time is written in TMS 9995 assembly language. Assembly language was selected because of the great number of complex calculations that must be performed in the 100 millisecond scan period. The program is an interrupt driven process, consisting of the following routines: Power up and external reset interrupt, valid range data interrupt, timer overflow interrupt, and scan request interrupt.

The power on/external reset is the highest priority interrupt and will take precedent over any other processor operations. This routine presets all the equation variables by reading from the front panel thumbwheel switches and loads dummy data into the output FIFO buffer. It also initializes all external hardware to prepare the system for the first scan request, presets the internal timer, enables all interupts, and then enters a suspended state waiting for the first scan request.

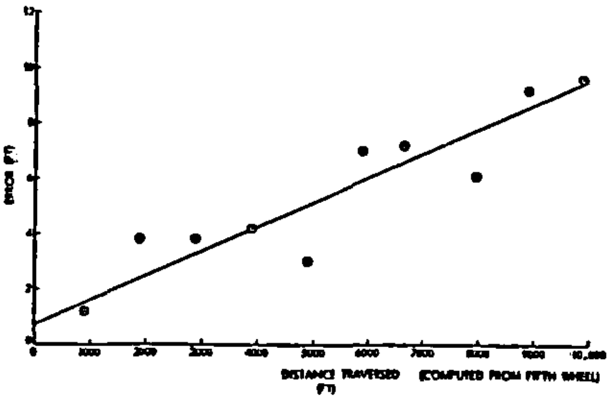


Figure 3 Example of a static calibration test

however, possible effects of acceleration and a higher velocity (i.e., an aircraft's velocity) were not discounted. Some indications of antenna interference were noted at a range below 500 feet, so antenna analyses have been planned to research this problem.

Flight Tests

To correctly evaluate the accuracy of the microwave positioning system, comparison of flight data with a baseline system was necessary. The most efficient system which has the required accuracy and quick processing time to accomplish this is a laser tracker, so flight tests were conducted concurrently with an existing Lockheed flight test program which utilized a laser tracker. Because the inflexibility of the test program did not allow for any time to evaluate the data or implement fixes during the program, validity tests were performed on a C-130 production vehicle at no cost to the MSPS program. Because there was no baseline data to compare the microwave system data with, these tests merely demonstrated the ability of the system to function in an aircraft environment.

The production flights were made after a minor modification to the C-130 aircraft. A crew hatch was modified to accomodate the master transmitter/receiver's omni antenna. Thus, only the crew hatch (located directly behind the pilot's station on the flight deck) had to be replaced, and the airborne components brought aboard (there was no radar altimeter recorded) to complete these tests. One remote transponder was placed at one end of the runway centerline to provide the test data. Several flights were flown without any indication of trouble.

The accuracy flight tests which occurred during the test program were conducted using the configuration in Figure 2. The aircraft was equipped with the components in Figure 1, and integrated with the LADS system on-board to provide real-time processing. The inflexibility of the aircraft test program was the cause of several problems. The antenna analysis tests mentioned in the ground test discussion above were not performed. In addition, software which would allow real-time analysis - aircraft position in three-dimensional rectangular coordinates, landing dispersion programs, and airspeed calibrations - could not be completed in time for the tests. The decision was made to go ahead with the program in this manner because the LADS would permit observation and recording of the range values, and the program would also allow a preliminary evaluation of the system and its current configuration (as shown in Figure 2).

This decision proved to be good because several otherwise undiscovered problems were revealed. The actual data output was limited. Only two flights were recorded with the microwave system. These flights produced viable data from three take-offs and twenty landings. The major problem with the system was attributed to a synchronization error between the DMU and control computer clocks. A modification to the DMU clock was supposed to have been accomplished before the equipment was purchased; however, this modification was not made. As a result, whenever the clocks became unsynchronized, a data dropout occurred, as shown in Figure 4.

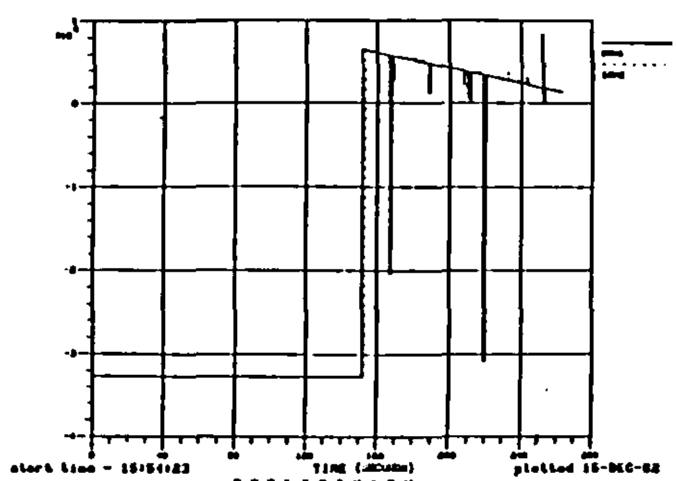
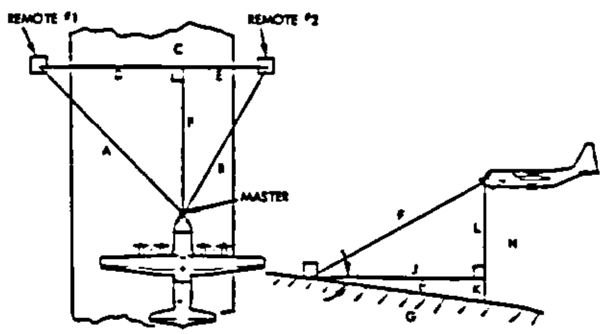


Figure 4 Example of MSPS slant ranges (note asynchronous dropouts) during a landing

Evaluation of the usable data has yielded some promising results. Initial error offsets were corrected by a constant mean error, resulting in data which deviated about the mean by one standard deviation

The valid range data is handled as interrupt level 1, which has priority over everything except power on interrupt. The control computer sends an external update request to the DMU (this is performed after a scan request). After a short delay, the DMU interrupts the control computer to signal that valid data is available, and this data is strobed into memory. The timer is checked to verify that the data was returned within the proper time frame. If not, an error is flagged by returning a special error code. The computer reads the status word to determine which DMU channel is returning data. If channel one is present, the timer is restarted, the radar altitude is read, and the processor waits for the next interrupt. If channel two is present, the timer is reset and the range calculations begin. This calculation involves algorithms that solve the equations in Figure 2. The resultant data is stored in a temporary buffer.



C IS THE KNOWN DISTANCE BETWEEN REMOTE TRANSPONDERS

A AND B ARE MEASURED BY DME EQUIPMENT

H IS MEASURED BY RADAR ALTIMETER

D, J, AND L ARE CALCULATED RANGE

D, J, L ARE CALCULATED AS FOLLOWS:

$D = (C^2 + A^2 - B^2) / 2C$

$F = \sqrt{A^2 - D^2}$

$J = \sqrt{F^2 - H^2}$ (APPROX FOR SMALL θ)

$K = J \tan \theta$

$L = H - K$

$J \approx \sqrt{F^2 - L^2}$

Figure 2 Configuration of MSPS during accuracy tests

The next lower priority interrupt is the timer overflow. The timer is started when the external update request is sent to the DMU. The timer is checked when valid data is returned. If too much time elapses after the external update command is issued before valid range data is received, the timer interrupt is executed. This program loads special data into the output buffer to signify that no valid data was returned, and waits for the next scan request.

The lowest priority interrupt is the scan request. This routine sends the external update request to the DMU. It also moves the data from the temporary calculation buffer into the hardware FIFO buffer, where it can be output on the next scan request. After this process is complete, operation suspends awaiting the next interrupt.

Ground Tests

The ground calibration tests were performed in three parts:

(1) Baseline system calibration

(2) Microwave system static calibration

(3) Microwave system dynamic checkout

The baseline system consisted of an extra "bicycle" wheel (referred to as a fifth wheel) mounted behind an instrumented van. The van acted as the ground counterpart of an aircraft installed with instrumentation comparable to the LADS insrumentation (refer to Figure 1). A remote transponder equipped with an omnidirectional antenna was placed at one end of the runway centerline to act as a signal transmitter/receiver. The fifth-wheel was calibrated by traversing a surveyed distance, Dobbins AFB runway, several times in the van. A mean value was computed from the fifth-wheel for this distance and used as a correction factor during the other ground tests.

The microwave system static calibration was accomplished using the same equipment from the baseline calibration. By using the measured fifth-wheel distance as a standard, the van was stopped at 1000 foot intervals along the runway, and range from the microwave system was noted. Numerous runs were made, after which the results were plotted, as shown in Figure 3. The plots of the results were extremely encouraging, as a definite linear trend with an average slope of about 0.7 feet error per 1000 feet of range was observed. Varying axis crossings (range equal to zero) were observed for the tests, which were done at various times and days; however, it is a simple procedure to zero-out the transponder error during setup using the transponder range calibrate potentiometers.

The dynamic checkout also used the same equipment and similar procedures as the previous calibrations. These tests were made to demonstrate the effects of motion on system accuracy. (Note: motion here refers to a constant speed as indicated on the van speedometer). The van was driven along the runway length at approximately sixty miles per hour while the microwave distance was recorded in PCM format on a standard tape recorder. Plots of these data showed no effect of constant motion on the microwave system accuracy;

of four to six feet. These data were then plotted, and since the results were linear, least squares linear regressions were calculated. These are depicted in Figure 5. The slopes of Transmitter 2 data show promising results. Both flights have error slopes which are approximately equal (1.4 foot error/1000 foot range) and which compare linearly with the static slopes of 0.7 foot error/1000 foot range. These results indicate that a slope correction implemented from the accuracy tests can improve data, corrected by only static test calibrations, by up to fifty percent. These constant slopes are encouraging because software may easily be incorporated to allow a real-time calibration of the data. The data from Transmitter 1, although linear, are not of constant slope. This may be attributed to time and weather effects which may necessitate a recalibration after a specified amount of use. By correcting the data by subtraction of a constant offset, a maximum standard deviation of 5.5 feet was obtained (Transmitter 2), which is an improvement over the 10 foot standard deviation as determined in previous systems. When the linear correction was applied, the standard deviation decreased to 5 and 2.6 feet, a percentage increase of 10 and 14 percent, respectively. Although the Transmitter 1 data was not considered to be valid, this method improved the standard deviation by 48 percent. These results are presented in Figure 6.

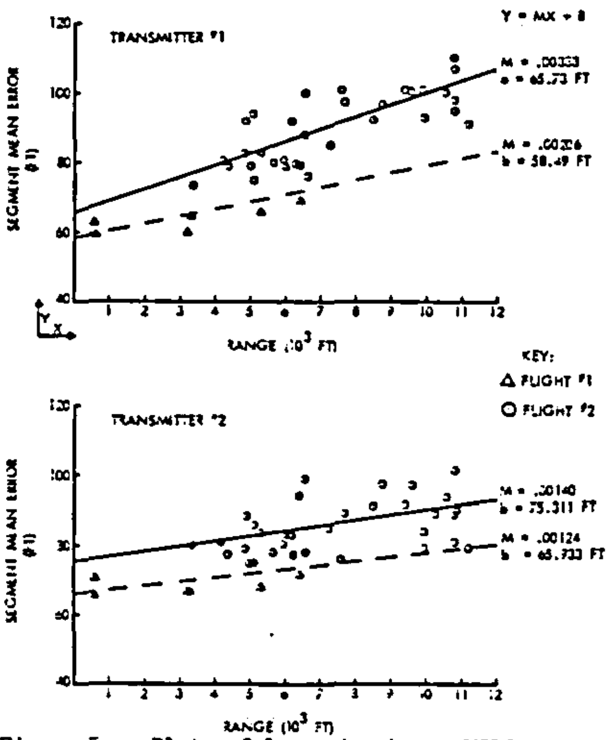


Figure 5 Plots of laser tracker - MSPS range error with linear regression superimposed

Conclusions-Where do we go from here?

Although many problems were encountered during this phase of the MSPS project, these test results signify the desirability of further research in this flight test application of microwave technology.

Antenna analyses are scheduled for 1983 to determine the characteristics of various configurations of the MSPS. These results will be utilized to aid in determining optimum remote transponder locations. Another factor of the system design will be the development of a configuration which will reduce conversion errors during coordinate transformations. The configuration as previously tested (Figure 3), was not adequate because the triangulation techniques necessary to transpose the data do not generate results which meet the criteria of accuracy necessary for TOL positioning (three-dimensional). An example of a preliminary configuration which will be considered is shown in Figure 7. This system will employ one trisponder at one end of the runway centerline to generate a longitudinal coordinate and signal to the control computer to scan the appropriate side remote. This side remote will generate a lateral signal during a specified length of longitudinal coordinate values. The number of side remotes can vary, depending on the length of the test range. The radar altimeter will again provide the vertical coordinate which will also be used to convert the transponder slant ranges to a corresponding two-dimensional cartesian coordinate. This will eliminate the large errors due to the large ratio of longitudinal/lateral lengths as found with the previous configuration.

Static and dynamic ground tests will be reconducted using the proposed remote transponder locations to validate the previous results. These ground test results, along with the necessary real-time analysis programs, will be programmed into the appropriate computer (MSPS and/or LADS). Another Lockheed test program has been tentatively scheduled which will use a laser tracker. The MSPS will again undergo accuracy tests concurrent with this program.

The problem of accurate, cost-effective TOL performance testing and analysis is of major concern to civil and military test organizations alike. The application of microwave technology may well prove to be a breakthrough in minimizing the soaring costs of flight testing. The system is currently capable of real-time, on-board processing at remote locations where test facilities, personnel, and time are a premium. At the conclusion of this program, the system will be enhanced by real-time on-

FLIGHT NO	TRANSMITTER	MEAN (CONSTANT CORRECTION FACTOR) (FT)	RESULTS OF CONSTANT CORRECTION* STD DEV (FT)	RESULTS OF LINEAR CORRECTION ** MEAN (FT)	RESULTS OF LINEAR CORRECTION ** STD DEV (FT)	PERCENTAGE STD DEV IMPROVEMENT WITH LINEAR CORRECTION APPLIED (%)
72	T1	64.81	4.55	-.386	2.38	48
73	T1	91.52	6.31	+.080	5.26	17
72	T2	69.54	3.04	-.398	2.62	14
73	T2	85.76	5.48	-.240	4.95	10

* CONSTANT CORRECTION EQUATION: CORRECTED ERROR = ERROR - MEAN

** LINEAR CORRECTION EQUATION: CORRECTED ERROR = ERROR - (SLOPE (RANGE VALUE) + ERROR INTERCEPT) (SEE FIGURE 5)

Figure 6 Comparison of constant and linearly corrected errors.

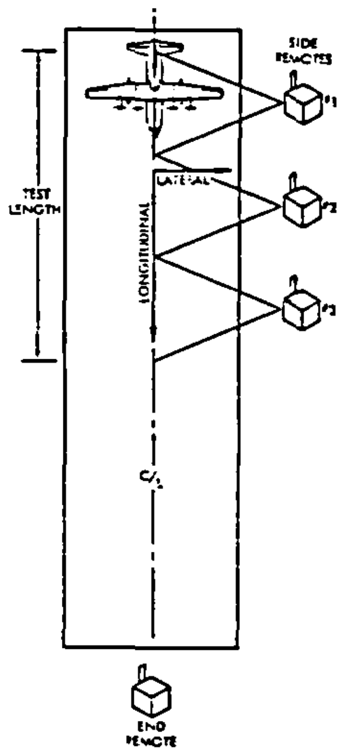


Figure 7 Proposed MSPS configuration to reduce transformation errors

board analysis routines that can be modified for specialized test requirements and analyzed by one flight test/ data analysis engineer (on-board) during the flight.

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TIME-SPACE POSITION INFORMATION AT EDWARDS AIR FORCE BASE, CALIFORNIA

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Abstract

The Edwards Flight Test Range is a typical Department of Defense range which provides support to numerous flight test programs. Increased requirements for more sophisticated and more accurate data has been met through a program of range instrumentation, methodology improvement and by the more sophisticated modeling of errors in the post flight processing software. Upgrades in software have improved technician ability to quality control data and multi-sensor mathematical models have improved the TSPI available to the users.

Introduction

Time-Space Position Information (TSPI) on the Edwards Flight Test Range at Edwards AFB is similar to TSPI at any of the other Department of Defense ranges. The instrumentation and the data processing remains the same as that used at other ranges. The Edwards Flight Test Range has had a regular program of enhancement and has been an active member of the Range Commander's Council over the past three decades.

Two areas of improvement in TSPI which have received special emphasis. The first is the assurance of TSPI quality through analysis of the sources of sensor errors and through new data production software which models these errors and incorporates more precise algorithms in the calculation of TSPI parameters. The second is to extend TSPI to applications which are not local to Edwards AFB, for example space shuttle reentry and landings at Edwards AFB which include radar tracking data from Vandenburg AFB.

Project requirements, not always satisfied have led to recent developments.

a. Accurate position information has required use of a Best Estimate of Trajectory with the radial error calculated at each point.

b. Calibration of instrumentation has been necessary to guarantee that real time tracking errors are less than 0.5 mils.

c. Fast turnaround of post flight data with intermediate range accuracies in position errors less than 50 feet.

d. Combination of tracking data from other ranges with Edwards Flight Test Range data to provide extended range coverage.

e. Aircraft to ground close ranges with 0.1 ft/sec accuracy; currently, the best possible accuracy is 1.0 ft/sec. Although all requirements can't be satisfied outright, they have driven the efforts to the point where TSPI accuracies are asymtotically approaching the practical instrumentation limit.

Arrangements to use the Edwards Flight Range are made through the usual channels by means of the Universal Documentation System. Range scheduling is accomplished according to AFFTC Regulation 55-15. Final details for range use are made through a Range Control Officer assigned to the Range Division and detailed data processing requirements are coordinated with a Data Operations. Analyst in the Computer Sciences Division.

Range and Range Instrumentation

A range is airspace, real estate, and instrumentation with well trained personnel. The Edwards Flight Test Range is 35 X 16 square miles with many additional miles of airspace. The instrumentation of interest is contained within these boundaries along with the special ranges, for example, DAGRAG, the gunnery range, and the Precision Impact Range Area (PIRA). Use of the instrumentation doesn't require direct overflight of the range. Radar coverage extends far beyond the boundaries of the range. The Edwards Flight Test Range is also tied to Vandenburg AFB, California and Hill AFB, Utah by microwave nets to allow extended coverage and the exchange of TSPI in real time. The most accurate information can be obtained only in the area of special ranges at Edwards where cinetheodolite coverage is available.

The instrumentation on the Edwards Flight Test Range is similar to that found on most ranges; consisting of three precision tracking radars, one airspace surveillance radar system, eight Contraves cinetheodolites, and four Askania cinetheodolites used on the main runway for take off and landings. The instrumentation radars consist of two FPS-16's and one TPQ-38, known as the Digital Instrumentation Radar. These radars provide tracking outside the local area since the range of the TPQ-38 is 125 miles and FPS-16 range is 32,000 miles. Actual FPS-16 tracking for aircraft can be provided up to 200 miles. The airspace surveillance radar, the MT-DARC, provides special

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service to the Ridley Mission Control Center for range traffic control, range safety, and also is a primary source of radar information to project where precision radar is not a requirement.

Use of the cinetheodolites is more demanding in the sense that the range of operation is much more restrictive. A Contraves isn't going to do well with a target that is more than ten or fifteen miles away and more than two Contraves observing a target with good triangulation is necessary for good TSPI. Careful planning is required to optimally use any of the instrumentation, but the cinetheodolies are tragetically placed to serve such functions as terrain following tests, bomb scoring, and flight termination.

An analyst familar with the Edwards Fight Test Range is required to select appropriate flight paths and to plan for correct instrumentation.

Real Time Data

Radar data is easiest to acquire and process in real time. Cinetheodolite data is acquired differently from radar data, giving it a decided advantage in accuracy. The difference also makes it more difficult to process cinetheodolite data, in real time or post flight. Unlike radar, cinetheodolites give no positive indication they are on track until film processing is complete.

The information obtained by an instrumentation radar is the spherical coordinates of the target being tracked. Normally the track is assisted by a transponder and detection of a target is positively identified by the radar system. Moreover, an indication of having a good track on a target is sent to any data processing device as part of the data. Cinetheodolites on the otherhand are passive devices and although angles are detected automatically, the range is not available. As a result of the cinetheodolite not sending a signal to the aircraft, no response is detectable in real time. Best cinetheodolite source solution is possible but angle corrections aren't, so the final real time result is only as good as the solution which uses an optimization technique, least squares, to provide position information. A major source of error is in the same data being used for the solution.

Post Flight Data Processing

TSPI data products provided to users are the result of post flight data processing. Real time products lack some of the corrections, for example refraction and cinetheodolite tracking error, and in addition short suts in the equations are required so computer time constraints in processing can be satisfied.

Typical post flight data processing has numerous steps, alternating processing with quality control. For radar data processing there is a method for providing a very accurate product with an accompanying cost in time or there is a faster method with some degradation in the accuracy of the final product. Both methods provide identical menus of data products.

Cinetheodolite data processing by its nature is a time consuming process. Data is recorded on film and at its best, film reading is only partially automated. Numerous data editing steps are required before the final TSPI products are available. The final cinetheodolite data products are identical to those available from the radar processing.

A multisensor solution for example from radars and cinetheodolites or Best Estimate of Trajectory is available. For the best data product of course an additional time penalty is exacted. The data products are the same as the radar and cinetheodolite data products.

The data products consists of the usual parameters which can be derived from the time history of target position plus some attitude measurements of the target in the film plane.

Data Accuracy

The TSPI data from the Edwards Flight Test Range is very good. Some actual values available to describe this accuracy are given as examples. For the FPS-16 radars (1)

Sigma Range < 6.0 feet

Sigma Azimuth, Elevation < .1 mil

Radars and cinetheodolites have numerous sources of errors which contribute to innaccuracies in the TSPI. A large number of these errors are systematic and are minimized through a constant program of instrumentation calibration and maintenance. The systematic errors are also modeled in the software. A continuing program of calibration aircraft flight, calibration satellite tracking, and star trail calibration is vigorously pursued at the Edwards Flight Test Range. Typical systematic errors are refraction, encoder zero-set, mis-level, skew, and droop. Random errors for example, receiver voice and glint are modeled in the usual manner with least squares or Kalman filters.

Post flight data processing uses all of the listed error modeling to correct data. The care taken in this calibration program assures the user of always receiving a high quality data product.

The Future

Improvements are always a goal, and our current plans are taking positive steps in several areas. The most provocative advances are in the cinetheodolite areas, where a program has been initiated to upgrade the cinetheodolites to include video recording and the film reading equipment to include automatic video tape reading capability. Areas of upgrade include potential useful real time cinethedolite TSPI. Additionally, Range Measurement System type sensors, if added to existing instrumentation, could improve the effectiveness of the cinetheodolites.

Another upgrade currently being installed is the Position Information Processing System (PIPS) which is designed to acquire and process TSPI from various range instruments and communicate this information to other instruments and range data processing equipment.

Outlined in this paper are the directions of improvement in range instrumentations and Time-Space Position Information processing at the Edwards Flight Test Range. Examples have been given which indicate but do not completely cover the full scope of Time-Space Position Information improvements.

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IMPACT OF CAD/CAM ON MODIFICATION OF FLIGHT TEST VEHICLES

by

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Abstract

A survey and analysis were made of an aircraft modification/prototype fabrication and installation organization to see if available Computer Integrated Engineering and Manufacturing (CIEM) techniques could reduce aircraft modification costs. The result of the study indicated the Computer Aided Design (CAD) and Computer Aided Manufacturing (CAM) technologies were available to decrease these costs. A 7 million-dollar modernization program was entered into and resulted in a reduction in overall modification cost of approximately 20 percent. Productivity of between 1.5:1 and 1.7:1 for the CAD element of the CIEM system and approximately 6.5:1 for the CAM element of the CIEM system was demonstrated. Reductions in modification times, as well as such ancillary benefits such as higher quality documentation packages and easier transitions from prototype engineering and manufacturing to production engineering and manufacturing documentation packages, can be obtained through use of this technology. System payback was found to occur between thirty-nine and forty-two months.

Background

Some things are common between industry and government flight test programs. There is never enough money in the budget and there is never enough time to do the job that the flight test engineer wants to accomplish. Of course, there are many reasons for this but they mostly have to do with unrealistic project time lines, mostly caused by excessive procrastination in the first place. Sure, budgetary cycles, funding cycles, market opportunities, and forecast cycles also enter into the equation, but the universal truth remains that flight test is last in the development chain and the thing that is last gets squeezed the most. This paper has to do with a way which has been found to shorten and decrease the cost of that phase immediately prior to the flight; that is the modification of the vehicle for performance of the test program.

The 4950th Test Wing is part of Aeronautical Systems Division and is located at Wright-Patterson Air Force Base, Ohio. It has as its prime mission the development of avionic subsystems. It also is responsible for initial tests of refueling, flight control, instrumentation and landing gear subsystems. In order to accomplish this task, it has an Aircraft Modification Center which accomplishes the modification of its fleet of approximately fifty test aircraft. The Aircraft Modification Center is a multi-functional unit which accomplishes design, fabrication and installation of aircraft structural and avionic Class II (test, temporary) modifications to aircraft.

Over the past several years, test costs have risen from approximately 1,800 dollars per flight test hour to the present 4,400 dollars per flight test hour (Figure 1). While there were many reasons for this cost increase (POL [fuel] costs, depot maintenance [DMIF] costs, etc), the basic fact remained that our customers' average flight test costs more than doubled in less than three years.

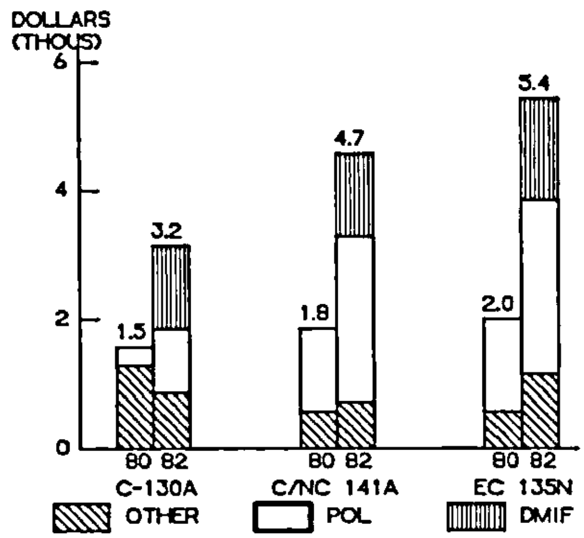

FIG.1. FLYING HOUR CUSTOMER CHARGE (DOD) DOLLARS BY AIRCRAFT AND FISCAL YEAR

Analysis of flight test program costs (Figures 2 and 3) revealed that while the modification cost remained fairly static, the actual percentage of program cost, while artificially reduced by higher flight hours, was still significant, accounting for some 15 to 20 percent of total program cost. While these costs were not the major drivers in the increased cost per flying hour to the customers, they were significant enough to demand management attention.

One of the reasons for the modification costs remaining at significant levels was due to the fact that the design and fabrication processes used to produce the modification had not changed significantly since the late 1940s. Engineers performed manual drafting and design tasks, using the same techniques which had been in existence since the inception of the aircraft industry with occasional diversions in available update technology, such as when they used off-line finite element structural analysis programs such as NASTRAN. The fabrication shops also used 1950's technology methods for producing manufacturing products. In most cases,

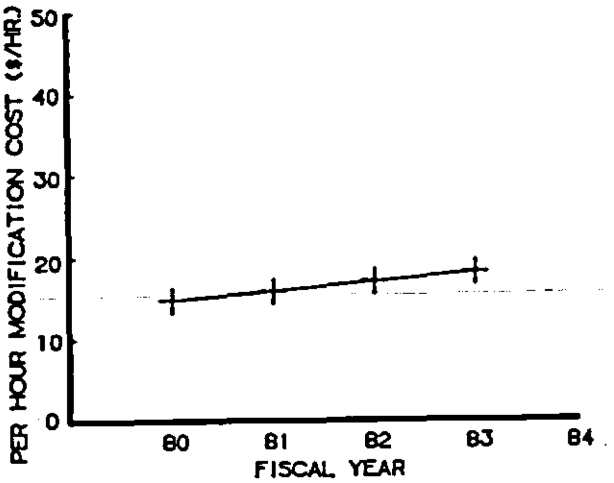

FIG.2 COST PER MODIFICATION HOUR

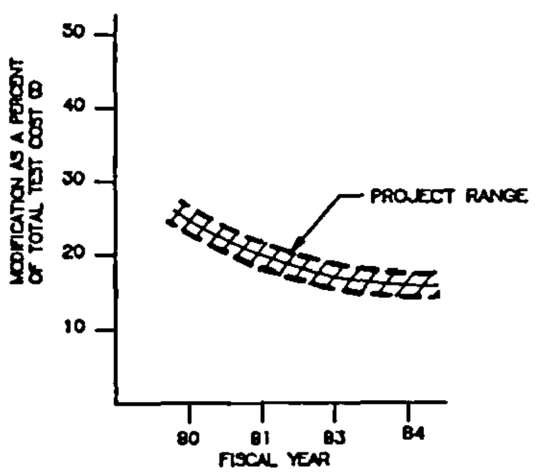

FIG 3. FLIGHT TEST MODIFICATION COST

they were also using machine tools which were themselves 1940's and 1950's vintage.

Against this scenario was constantly being played a virtual symphony of customer complaints about the high cost of flight tests. Since many parts of the flight test cost formula were fixed (fuel, utility cost, instrumentation, aircraft maintenance, replacement parts), the attention turned to the cost of modification of vehicles for flight test. The approach taken at the time (1979) was to define all the various elements which made up the modification process and then look to the existing and emerging technologies to see if any savings could result from the use of their application.

The results of this study indicated that application of CIEM technology and techniques could result in significant decreases in the modification costs of aerospace vehicles. Of the 1979 available technologies, it appeared that utilization of the productivity features of CAD/CAM techniques was the most promising. With this in mind, and after much activity to justify the program through use of economic analysis (net present value, payback, return on investment), the Test Wing set upon the road of productivity enhancement.

System Description

Prior to showing how CAD/CAM was used, some definitions may be helpful. Computer Aided Design is just that; a computer, used as a tool, to assist a designer in some of the more mundane tasks associated with design, such as layout, dimensioning, etc. However, it is much more than a drafting tool, since it can store enough information to allow multiple views, isometrics, and with the newer systems, solids modeling depiction and automatic finite element mesh generation. A typical example can be seen in Figure 4.

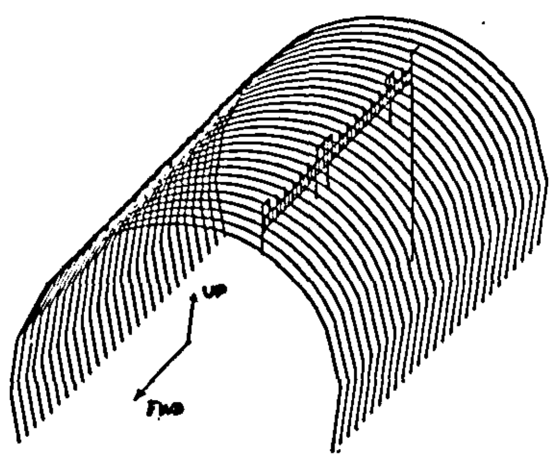

FIG.4 FINITE ELEMENT MESH

The system philosophy which was developed was to pursue a multi-phase approach, essentially to make sure that the modification design engineers could use the tool and reduce cost prior to spending the entire amount of budgeted capital on something that would not reduce modification costs. This go-slow approach is not uncommon and in our case appeared to be rational.

The multi-phase system consisted of a PDP-11/34 based, white-on-black display tube, four workstation system with associated peripherals such as quick-look hard-copy device, a digitizer board (to convert existing drawings into a magnetic tape data base) and a keyboard/electronic pen system for communication purposes. The system immediately found application in modification layout work in restricted areas, such as cockpits and behind instrument panels. The designers found that they could lay out the stringer/frame structure in three dimensions and then insert the modified element and check for static clearances, as well as dynamic interferences, at the CAD system instead of on the hangar floor.

Modification cost did, in fact, come down and the system expanded to its second phase which tripled the number of work stations, added instrumentation design and expanded to include color displays. The third phase of the system upgraded the hardware to a 32-bit VAX central processing unit and upgraded software to allow for true solids modeling through use of Boolean algebraic operators. This final CAD system is shown schematically in Figure 5.

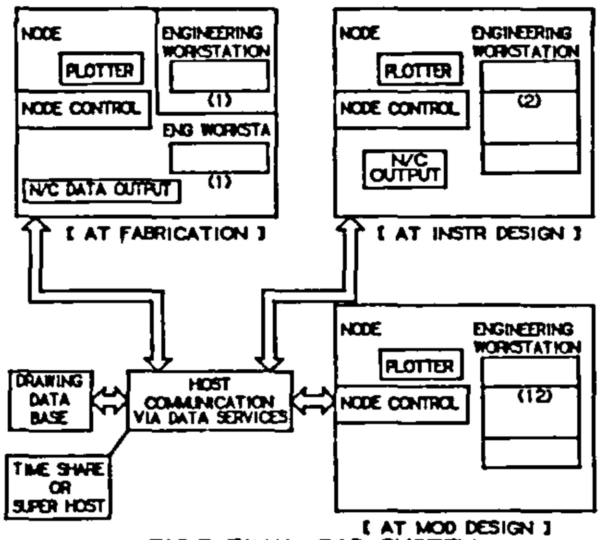

FIG.5 FINAL CAD SYSTEM
(PHASE III)

At this point CAM procurement actions, started in parallel to the CAD effort, contributed to productivity in production of aircraft modifications. While the shop up to this time did have some small numerical control (NC) milling machines, it did not have any computer numerical controlled (CNC) or direct numeric controlled (DNC) equipment. The CAM machinery which was purchased was for the support of the mostly sheet metal and machining operations required for aircraft modification. These machines differed from their predecessors in that they were either micro-processor controlled as in the case of CNC or capable of being directly controlled from a central computer facility as in the case of DNC. Incorporation of these machines then yielded a complete CAD/CAM system.

Integration of the system was assured by specifying that all the CAM machinery purchased has post processors which were compatible with the CAD system. This latter point was made part of the benchmark test for the CAM machinery. In order to prioritize the system to the flight test role, investments in a sheet metal cell (shears, brakes, punches) followed by investments in a machining cell (turning centers, vertical machining centers, jigs, borers and mills) were made somewhat in sequence. Collaterally, investments in a printed circuit board (PCB) facility and a PCB plating facility were accomplished.

Program Cost and Schedule

Having outlined the general problem and philosophy behind the CIEM system composed of both CAD and CAM elements, we'll now look at the cost and schedule for setting up the system, as well as its predicted payback. This will be followed with examples of cost and time savings obtained on the system, their effect of modification of the aerospace vehicle for flight test, the overall reduction in test cost and the actual payback performance to date. Ancillary benefits which are not readily quantifiable in terms of dollars or time saved in the short term will also be discussed.

Computer Aided Design (CAD) System

The system costs for the CAD system consisted of nonrecurring as well as recurring costs. Just as in any procurement of a capital investment, these costs were summed to obtain total system costs. Some of the nonrecurring costs were implementation (solicitation, procurement, benchmark tests), equipment (hardware), software, training and communication installation costs. Recurring costs were for technical support, maintenance, communication line leases and supplies (consumables). System costs are shown on an actual basis in Figure 6.

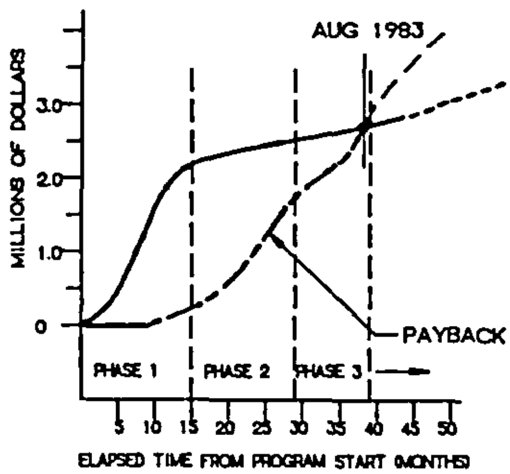

FIG 6. CUMULATIVE CAD SYSTEM COST/PAYBACK

The costs have been smoothed for simplification of presentation. It took us approximately four months to get on contract, five months to obtain hardware and software and three to four months to train operators to a minimally acceptable level. It took an additional three months until the operators' productivity was to a level which showed a positive net gain. However, from that time on the savings generated through productivity improvement have been substantial, resulting in an average 20 percent cost reduction to our customers over the past two years. As can be seen in Figure 6, the system cost to date of 2.75 million dollars has, thirty-eight months into the program, almost been paid back with an offset savings of 2.65 million dollars. Our initial calculations showed payback of the system somewhere between the twenty-sixth and thirty-seventh month point. Our results to date indicate payback at about the thirty-eighth month point. The reason for the discrepancy lies in two central facts. First, in 1979, industry predictions for CAD productivity were running at about 2:1 to 4:1. In our economic analysis, we used the lower value of two hours of manual output for every hour of system input (2:1) due to the prototype nature of our work. It appears that in the design end of the aircraft modification business, we will reach overall productivity values of 1.7:1 to 1.8:1 from our present position of 1.5:1. The second fact which entered into the longer payback period was the longer than anticipated time it took to convert standard aircraft drafting and design personnel from conventional methods over to computer-assisted methods. We forecasted three to six months for training; it took five to eight months for the conversion. However, even with these deficiencies, our design costs for prototype modifications decreased 20 percent over a two to two-and-one-half year period (clock starts when equipment is in-house versus when decision is made to purchase the equipment) and an approximately 2.8 million-dollar capital investment program has been fully paid off. Productivity claims, in our experience, were found to be slightly overexaggerated for CAD, but the technology was found to be applicable to the design phase of modification of aircraft for flight test. For matters of reference the above values were obtained using an Applicon/Schlumberger system. This system actually consists of three generations of equipment in six subsystems tied together by a DEC-net communications network. The system components are listed below in Table 1. Peripheral support for the system is supplied by three E-size pen plotters, one Xynetics J-size pen plotter, two 300 megabyte (Mb) drives, five 200 Mb drives, two line printers, one digitizer tablet and five VT-100 terminals.

It should be noted that the above productivity numbers were obtained from a system which did not have available to it an existing digitized drawing data base. Higher values of productivity, possibly as high as 4:1, may be expected if there is already existing in the company a digitized drawing reference file or if such a file could be obtained which is either compatible with CAD system operating language or with come neutral language, such as the Initial Graphic Exchange Standard (IGES).

Computer Aided Manufacturing (CAM) System

The CAM system costs just as the CAD costs consisted of both recurring and nonrecurring elements. The costs for the system are shown in Figure 7.

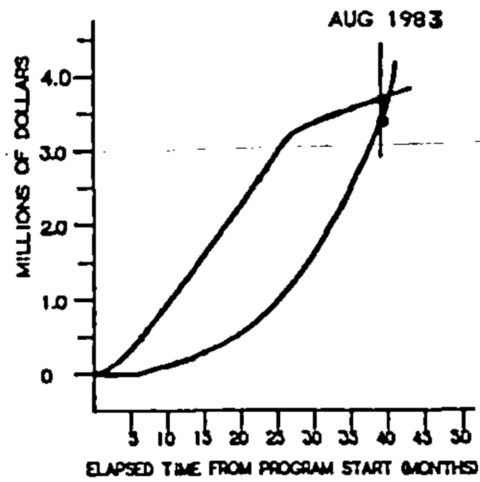

FIG 7. CUMULATIVE CAM SYSTEM COST/PAYBACK

Again the costs have been smoothed for presentation purposes but actually are incurred as discreet disbursements as individual machines are placed on contract.

As was alluded to earlier, the implementation of the CAM system initially centered around primarily the sheet metal function of the shop. This was due to the fact that approximately 75 to 80 percent of all flight test modification work passed through sheet metal on its way to the modified aircraft. The machining center was next funded due to the fact that 40 to 60 percent of modification aircraft parts passed through this function on the way to the flight test vehicle. These two functions have accounted for 86 percent of our CAM investment to date. The remainder of the program is taken up with printed circuit board fabrication machinery and precision coupon/specimen preparation machines which are used in zone shops which support the research laboratories located at the Air Force Wright Aeronautical Laboratories (AFWAL),

Table 1 CAD System Components

Subsystem	Applicon Designator	CPU	Operating Software	Number of Terminals
1	895	DEC PDP 11-34	RSX-11	4
2	895	DEC PDP 11-34	RSX-11	4
3	895	DEC PDP 11-34	RSX-11	2
4	895	DEC PDP 11-34	RSX-11	3
5	4245	Applicon/DEC VAX Emulator*	GPF	4
6	4275	DEC VAX 11-751**	GPF	4

*Completed Beta test
**In Beta test

Wright-Patterson Air Force Base, Ohio. These zone shops are also operated by the Directorate of Aircraft Modification of the 4950th Test Wing.

The payback curve (Figure 7) shows initially a somewhat quicker return on investment than CAD, mostly due to the fact that the training time required for conversion from standard and numerically controlled machine tools to CNC was much less than for the CAD system. This was also true due to the standardization in programming language available with CAM equipment which is not as yet available for CAD systems on a universal basis. The overall slope of the CAM payback curve is shallower than the CAD payback. Some of the reasons for this have to do with machine use, eleven hours per day on average, which is essentially limited by the assigned fabrication manpower and also due to setup and programming downtimes. In spite of these factors, the predicted payback, 95 percent into the present investment, is approximately forty-one months. Additional lag in the rapidity at which full payback occurs is caused by the fact that many of the US made machine tools purchased are not in inventory as yet due to their expense. Since not in inventory, they have to be built. It is not uncommon in this situation that these types of machine tools are not delivered until eight to ten months after the ordering date. This delivery time lag is the main reason for the long payback period, and a prime reason for the increased demand for Japanese machine tools in the US.

Productivity on the CAM machinery has been found to be much higher than on the CAD equipment, averages about 6.5:1 and was, to say at least, a very pleasant surprise. After investigation, several causes were isolated. The first was identified rather early in the system's implementation and goes squarely against the fairly widely held belief that CAM machinery should only be used in production and not prototype shops. It seems, when you think of the process, that the cause for the productivity should have been self-evident. It is simply that in the test business the prototype installation changes are not uncommon due to new requirements, low dimensional tolerance adherence on pre-production prototype vehicles, poor configuration control over the prototype electronic equipment to be installed in the vehicle causing last-minute requests for changes in rack/bay volume, access, mounting, etc. This requires many design changes to the prototype installation which in the past required scrapping hardware and starting the fabrication process over again. With CAM equipment, a minor modification of the numerical control part code is all that is required in many cases to take care of the change. A large percentage of these changes can be made right at the machine, further reducing the turn-around time. Since you don't have to start from scratch, the task is accomplished much more rapidly. Some other features of CAD/CAM which affect fabrication efficiency are visualization of tool paths (saves setting up again after you've machined through a tie-down clamp, avoids machining through boundaries, tolerance violation, etc) and automatic flat pattern conversion from final dimension drawings. Additionally, very low scrap rates avoid long delays for material reordering, as well as decrease energy requirements caused by the rework or remanufacture of scrap parts. While the average productivity factor has been found to date to be approximately 6.5:1, individual examples of as low as 2:1 to as high as 1000:1 have been encountered. In our prototype operation, it appears as if CAM productivity claims by industry have been understated. Again, for matters of reference, our CAM system components on hand, as well as on order, are listed in Table 2 for convenience. The effect of this system on our overall manufacturing cost for aircraft modification has been estimated to be about 11 to 14 percent. This number was obtained by determining the proportionate share of cost of the fabrication to the support and installation costs of the modification.

<u>Applications</u>

Prior to discussing the actual cost reduction resulting from the application of CIEM techniques to the modification process, some examples of how it physically was used may be helpful. Illustrations of CAD and CAM will be given separately and one example showing how both of these elements were used in conjunction will be discussed.

The first example of how we have used CAD has to do with something which was discussed earlier; that is modification of a system into a very constrained and restricted area. During the modification of a T-39B to support the B-52 companion trainer aircraft (CTA) program, whose object was to see if business jet-class aircraft could provide a low-cost vehicle to provide realistic B-52 subsystems training, we were faced with the task of extensively modifying the cockpit area of two T-39B aircraft. Due to the restricted area, the designers were, in fact, having a difficult time in the initial layups; that is, in getting the required project instrumentation to fit in the co-pilot's instrument panel without interfering with existing instruments or wave-guide runs which were routed through the area. Several layups were tried with little success because of the three-dimensional nature of the problem. Just prior to going the wooden or cardboard mockup route, one of our designers decided to lay in the frame, stringers and skin lines, as well as the existing instrument geometries, three-dimensionally on the CAD system. This was accomplished in short order and an alternate route for the wave-guide mounting and placement of the required experiment components were determined. The drawings were released to the floor and the installation was accomplished without any difficulty. The problem was solved in the engineering room without use of several fabrication people, mockups and half a dozen engineering change orders—all because of the three-dimensional representational capabilities of the CAD system. Another example had to do with the addition of a conformal strap on a structural casting on an EC-135N aircraft bulkhead for stress relief

Table 2 CAM System Components

Nomenclature (Descriptive)	Year of Purchase (Fiscal Year)
Cincinnati Fast Setup Power Shear (Aluminum)	1980
Battelle Memorial/PROMECAM Adaptive Control CNC Press Brake	1980
Weidematic Turret Punching Center	1981
Excellon XL-5 Driller/Router	1981
Mazak (Bubble Memory) Slant 30 Turning Center	1981
Mazak (Bubble Memory) Slant 40 Turning Center	1981
Bridgeport Series I Milling Machine	1981
Pratt and Whitney Triax Machining Center (Bubble Memory)	1982
Ekstrom-Carlson Wood Router	1982
Tree Turning Center	1982
Bridgeport Series II Milling Machines	1982
Wood Panel Saw	1983
Wire CNC Electrical Discharge Machine	1983
Vertical Machining Center	1983
Hydraulic Press Brake	1983
Copying/Tilt Table Milling Machine	1983

purposes. In this case, the engineering drawing was digitized into the CAD system using the digitizer tablet feature of the system. The dimensions taken from the aircraft as built, entered into the system, and comparisons and corrections made to the previously entered data base. The corrected non-constant radius of curvature contour with manufacturing imperfections was matched perfectly and the 7075T73 strap fabricated once and fitted perfectly.

On the CAM side, difficulty was experienced in producing a streamline extrusion die, primarily due to the precise requirement for the placement of cubic splines between the cylindrical entrance and the T-section exit elements of the die. Smoothness was required with precise boundary control tolerances to allow proper flow of the matrix aluminum material without fracturing the matrix during the extrusion process. The solution was to convert a mathematical model to three-dimensional surface coordinates, enter this into the CAD system where use of specialized NC routines converted the surface data into an offset tool path program for one of the CNC mills and machine a carbon male electrode from graphite. The contoured graphite electrode was used in an electrical discharge machine and the extrusion die, which more closely resembled a converging nozzle than an impact die, was produced in three days as opposed to three to four weeks if things went absolutely perfect.

A combined use of CAD/CAM was prompted by a request to design, fabricate and install a night refueling floodlight into the vertical fin tip of a KC-135A refueling tanker. The device was also to be flight tested. The period of performance from design through fabrication and installation and completion of the flight test program, including issuance of the flight test report, was thirty days. The system is shown in Figure 8. The design, fabrication, installation and checkout took all of ten days. Local checkout and final flight test, including the report, took an additional twenty days. Obviously, without the detailed integration capabilities of a CIEM system, such a program could not be accomplished within the established time constraints. Our estimate was that this modification portion of this program was reduced by a factor of five through use of computer assisted technology.

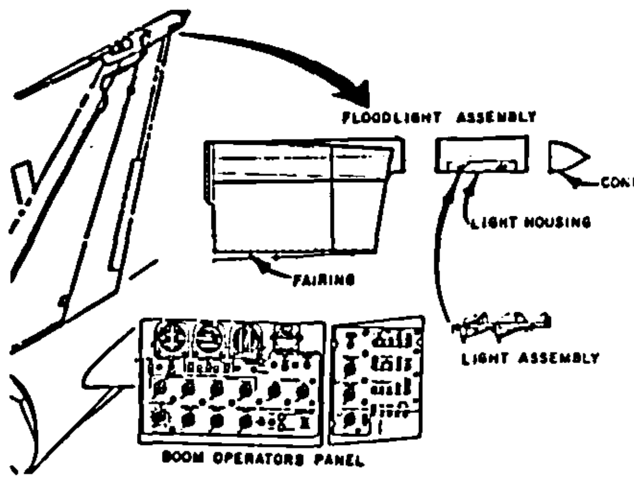

FIG.8 KC-135A FLOODLIGHT

Results

Our use of CAD/CAM technology and CIEM techniques has resulted to date in a 20 percent decrease in the modification costs of flight test vehicles and has been instrumental in shortening modification time. The cost decrease has occurred mostly due to the effect of productivity in the engineering area, with engineering accounting for approximately two-thirds of cost decrease. This has contributed in decreasing the percent of modification cost as a function of total program cost from approximately 25 percent in 1980 to its present level of approximately 15 percent.

Productivity of CAD equipment applied to the modification of flight test vehicles appears to be in the range of 1.5:1 to 1.7:1. Productivity of CAM equipment applied to the fabrication of components for modification of flight test vehicles was found to be approximately 6.5:1. The reason that CAD appears to contribute more to the reduction of modification cost than CAM is due to the fact that the design engineering modification process is weighted more heavily since it is not diluted by noncomputerized processes of plating, material handling and preparation, transportation, and physical installation and custom fitting of the parts on the test aircraft.

Use of CIEM techniques has resulted in retarding the increased cost in flight testing during a period of rapidly rising fuel and aircraft support costs.

Additional benefits of this technology have been higher quality engineering and fabrication documentation on modifications, easier transition from prototype engineering to production engineering packages, more rapid incorporation of experimental changes, easily recoverable family of parts files and higher quality graphics for incorporation into technical orders, as well as higher quality illustrated parts breakdown lists.

In conclusion, CIEM techniques and, more specifically, use of CAD/CAM technology can reduce overall flight test program costs and can reduce the time required for physical modification of the vehicle. The reduction in modification costs and decrease in modification time can return some level of flexibility back to the flight test program, the most important part of any development program, the portion where you obtain the data to prove the efficiency and define the limitations of the system or subsystem under test.