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THE IMPACT OF INTEGRATED GUIDANCE AND CONTROL TECHNOLOGY ON WEA--ETC(U)
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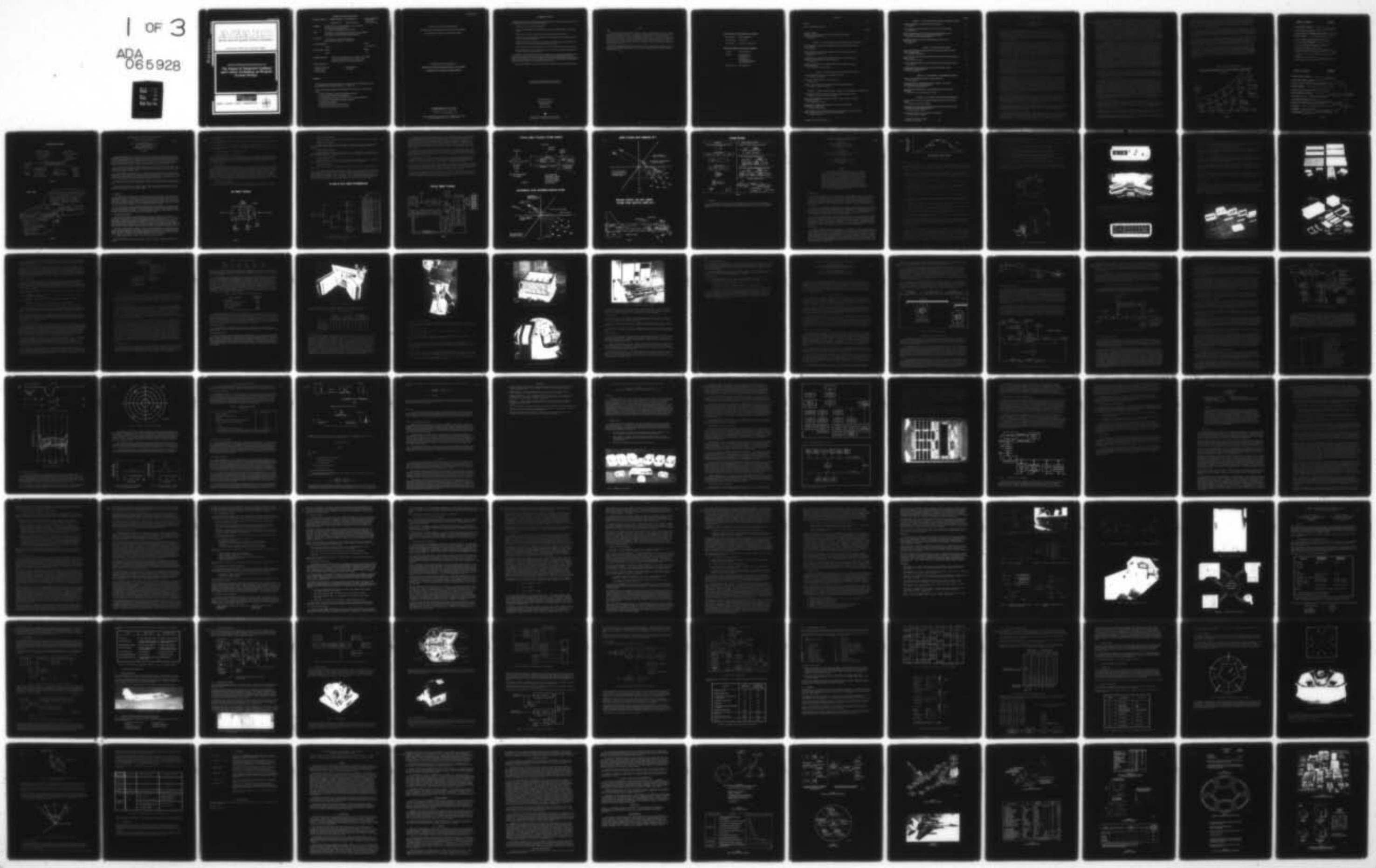

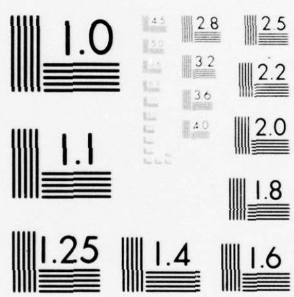

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AGARD CONFERENCE PROCEEDINGS No. 257

The Impact of Integrated Guidance and Control Technology on Weapons Systems Design

REPRODUCED BY
NATIONAL TECHNICAL
INFORMATION SERVICE
U.S. DEPARTMENT OF COMMERCE
SPRINGFIELD, VA. 22161

NORTH ATLANTIC TREATY ORGANIZATION

REPORT DOCUMENTATION PAGE

1. Recipient's Reference	2. Originator's Reference	3. Further Reference	4. Security Classification of Document
	AGARD-CP-257	ISBN 92-835-1303-7	UNCLASSIFIED

5. Originator Advisory Group for Aerospace Research and Development
North Atlantic Treaty Organization
7 rue Ancelle, 92200 Neuilly sur Seine, France

6. Title THE IMPACT OF INTEGRATED GUIDANCE AND CONTROL TECHNOLOGY ON WEAPONS SYSTEMS DESIGN

7. Presented at the Guidance and Control Panel Symposium held in Sandefjord, Norway, 9–12 May 1978.

8. Author(s)/Editor(s)	9. Date
Various	December 1978

10. Author's/Editor's Address	11. Pages
Various	248

12. Distribution Statement This document is distributed in accordance with AGARD policies and regulations, which are outlined on the Outside Back Covers of all AGARD publications.

13. Keywords/Descriptors

Integrated technology
Data processing
Weapon delivery
Terrain following
Crew station

14. Abstract

The Proceedings include papers presented at a symposium of the AGARD Guidance and Control Panel, held in Sandefjord, Norway, 9–12 May 1978.

Twenty seven papers (not all cleared for publication) were presented on the following topics:

- Functional Design, Concepts, Requirements and Trends
- Weapon Delivery/Flight Control Integration
- Communications, Command, Control (C^3) and Sensor Data Integration
- Crew Station Configurations and Display Concepts
- Pilot/System Interaction
- Data Processing and Distribution Systems
- Development and System Test Experiences

AGARD-CP-257

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.257

THE IMPACT OF INTEGRATED GUIDANCE AND CONTROL TECHNOLOGY ON WEAPONS SYSTEMS DESIGN

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Papers presented at the Guidance and Control Panel Symposium held in Sandefjord, Norway, 9–12 May 1978.

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Published December 1978

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ISBN 92-835-1303-7

Printed by Technical Editing and Reproduction Ltd
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PREFACE

The rapidly developing technologies in navigation, sensors, target identification sensors, command and control and computation capability are structuring a command network that demands increased functional integration of crew station and control configuration to permit effective use of that technology. This technology, when combined with advancing technologies in guidance and control, the driving forces of acquisition and life cycle costs, needs for operational tactical flexibility, survivability, vulnerability, and critical volume and weight constraints, dictates the need for integrated guidance and control at a higher functional level than heretofore considered. This higher functional level involves an effective blend of the sensor, vehicle and kill-mechanism that can provide a multi-role capability for advanced and present operational vehicles.

The papers presented herein definitely indicate that when one considers the large array of sensors available and the fundamental commonality of functions and control algorithms for different missions, it appears logical that these capabilities should be utilized to augment each other to achieve flexibility and growth capability.

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† Published in CP-257 (Supplement) Classified.

Reference

† Published in CP-257 (Supplement) Classified.
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CMOS

The Impact of Integrated Guidance and Control Technology on Weapons System Design

by

Colonel Eugene L. DeNezza
Commander
European Office of Aerospace Research & Development
London, England

K-1

It is a pleasure for me to be with you at this 26th meeting of the Guidance and Control Panel. I know the subject of this Panel's meeting has a widespread interest and concern to all member nations represented here. It is of vital importance and also a very meaningful subject to me as a result of experience in both the development and operational sides of the question being addressed here today. So, I am very delighted to be with you and be part of your deliberations for the next few days. For my part, I would like to attempt to set the stage for the next four days by describing to you in somewhat general terms where I feel some of the functional concepts and trends you will be discussing here will be leading us in the next decade or so.

I was very interested in the choice of topic for this meeting where we are stressing integration as a functional process and at a lower level than is normally considered integration. For the most part, integration has been thought of as a physical process which is generally left to the vehicle designer wherein all the other elements are considered either components or subsystems of the total system. This process is currently in a state of change in that integration is being considered as one of primary tools to affect some level of standardization or interoperability, increased performance and effectiveness, and a mechanism to achieve cost reduction. This function, or process, is rapidly changing to a methodology because of the need to consider the entire mission in the air warfare context. The entire system and its dynamics must be functionally structured to insure compatibility with sensor, vehicle, and weapon dynamics. This now demands that the functional integration be performed in the very preliminary design stages of a system to dynamically and functionally architect the overall system similar to civil engineering practices wherein the architect performs the integration, not the builders.

The first slide illustrates both history and trends in guidance and control technology. Briefly, it summarizes some of the guidance and control technology transitions that have occurred over the past two and one-half decades beginning with the 1950s; but more significantly, it illustrates two basic trends. The first is the increase in guidance and control functions and requirements and the second is the progression from the vehicle to the weapon systems into what we now call air warfare systems. During the early 1950s, the trend was predominantly vehicle oriented in terms of stability augmentation systems to improve the response and control characteristics, although more external loop closures were being made at that time in missile systems. This was followed by closed-loop terrain-following control systems to improve adverse weather, low altitude performance and the safety aspects of the system. Adaptive system concepts became operational in the F-111. These concepts which have been discussed extensively in other publications contributed substantially to the current modern control theory and technology. As a side-line, a seldom recognized fact is that the Kalman Filtering Theory was a direct outgrowth of the theoretical work undertaken to describe, analyze, and design adaptive systems.

Emergence of fly-by-wire technology in various forms became evident during the early 1960s with fly-by-wire's Spoiler Controls on the F-111 and electronic flight control in the Mirage. Increased use of inertial sensors and navigation systems to augment vehicle or flight path control for reducing landing minimums and improve interaction with area navigation and advanced air traffic control concepts during the early 1960s represent initial steps in functional integration for increased performance. During the mid-60s, the benefits that feedback control could offer toward alleviating structure fatigue problems and turbulence sensitivity were recognized. This work culminated in application of maneuver load control, structural mode control to alleviate turbulence induced fatigue problems and improved crew ride quality in the B-1 and other systems. Progressing into the early 1970s, the MRCA, Viggen, Concorde, F-16, and YC-14 designs witnessed incorporation of sidestick controllers, electronic displays, increased emphasis on digital techniques, control and propulsion system dynamic integration, direct lift control, ride control, and relaxed static stability, all of which were made possible through the implementation of emerging technologies such as fly-by-wire systems.

In looking toward the future of air warfare systems, the impact that command control and communications will have on the vehicle guidance and control in terms of tactical control is significant. This need generates another guidance and control dynamics loop which dictates substantial dynamic and functional integration to provide the desired operational capability. The time-space positioning capability permitted through accurate position-fixing employing advanced navigation systems and on-board vehicle trajectory control permits the implementation of the command and control function to marshal forces for tactical deployment and also offers a means for a redirect capability.

The six-degree-of-control-freedom projected through active control technology permits design freedom and tactical capability unavailable heretofore other than in specialized rotary wing configurations. These capabilities have stimulated application of modern control theories involving differential gaming, to determine strategies that can provide optimum trajectories, tactical options to increase weapon delivery accuracies and minimize pilot workload.

With that brief trend projection, and knowing that the guidance and control discipline is probably more sensitive to technology advances in other fields than other disciplines to satisfy operational needs, it appears appropriate to examine a few of these to determine potential capabilities. This next chart is a brief listing of a few of those technologies that appear significant, particularly if appropriate combinations are made. If one examines computation capability, it is well recognized that we will be seeing several orders of magnitude increases within the next two decades. The anticipated computation

thickness by increments of 0.300 in. (7.62mm) (see Figure 3). Figure 2 also identifies the features of a typical SEM Program module:

speeds of up to 5×10^8 bits per second with a memory capability of about 10^9 bit storage appears readily achievable. If this is combined with some of the advanced control theories like state space control theory and optimal control information and decision theories, the potential power of these theories now will become practical in implementation for design and real time operation. For example, state space control principles which have been mathematically possible for over a decade can now become a reality. This permits application of multi-variable control principle theory into the multiple dimensional domain wherein complex trajectories and multiple vehicle mission states can be dynamically optimized to achieve the desired mission capability. It also permits evaluation of maneuver tactics, configuration impact on new technology for design purposes and, also, through on-board computation, display complex optimum trajectories to counter threats, increase maneuvering performance and weapon delivery accuracy. Similarly, with this increase in computation capability the related modern control theories can be exploited in real-time to provide capability for handling increased information and making logical decisions, optimizing design methods of active control functions, and permit design of higher order state filters which can reduce dependence upon accurate measurement of specific states. One example being reduction in complexity of inertial systems through better modeling and state estimation. Increased use of decision theories will permit reduction of the amount of information required by the pilot or crew to execute functions or monitor system performance, thereby significantly reduce crew workload or possibly crew complement.

K-2

Considerable performance capability and simplification of system design can be achieved by blending outputs or augmenting outputs rather than having dedicated sensors for each function. This, of course, can have a significant impact on cost and design methods to achieve system integrity and safety by permitting continued capability without employing dedicated reversion systems or back-up capability. For example, with the increase in computation capability and theories, there are many surface motions and response capabilities implicit in the vehicle which, if tracked and integrated, could result in using the vehicle itself as a source of state information. Another example is, the capability to integrate command and control concepts into a vehicle in a manner that updates the on-board systems to permit autonomous operation with minimum change in crew operation or mission accomplishment. Similarly, active control technology presently is addressing examination of specific mode capabilities and control capabilities of these modes for specific application, has capabilities that have not been addressed to date such as, providing increased resistance to vehicle disturbances, increased accuracy of weapons line control, and introducing dominant changes in design methodology to reduce vehicle size and weight. Operationally, a significant potential exists for the capability to provide a multi-role function for vehicles at very minimal compromises such as, using air-to-air vehicles for air-to-ground operations. Some aspects of this will be addressed in more detail. Similarly, task or mission oriented control capabilities are receiving increased attention and provide distinct potential for defining control characteristics and dynamics for the total mission guidance and control as opposed to the classical vehicle handling qualities. These principles will also provide the methodology for definition of degrees of automation considering a total pilot capability such as information requirements, display techniques, and control characteristics for specific mission functions.

Recent research in the bio-medical field has indicated that through appropriate sensors, human thought processes and decisions can be used for control inputs without relying upon limb operated manipulation devices. Considerable amounts of pilot workload consist of converting information available from visual, audio, and motion cues into limb motion and associated integration required to manipulate control devices. One example could be the entry of new coordinates on mapping display by tracking eye motions and entering data without any hand motion.

The advent of holography and related techniques in terms of multi-function displays can now provide the capability to generate situation and command information in three-dimensions thereby reducing a number of display components required by the pilot. This will significantly reduce the amount of information integration the pilot has to perform, and it can be implemented in either heads-down or heads-up configuration. The new data of graphics and the capability to generate digital graphic projection images is promising the capability to generate trajectory, ground or terrain profiles in true time perspective thereby functionally providing a means of generating synthetic visibility for low altitude and adverse weather operation. The area of mapping, though not directly related to control technology, now provides ground positioning information that will permit a common grid system with sufficient accuracy for terrain mapping which, when combined with graphics, has the potential of providing the terrain-following and avoidance with reduced dependency upon terrain-following radars. The increased capabilities for accurately securing position and velocity information on-board the vehicle through inertial sensor technology and updating with satellite positioning information is providing a capability to achieve an on-board time-space positioning capability which, when integrated with other sensor information within the vehicle, can minimize the criticality of continuous communication capability and substantially minimize total system dependence on this capability.

This next chart illustrates that appropriate combination of these technologies offer potential capabilities that are not recognized when examining individual technologies. For example, the combination of high computation capability with multi-dimensional state space control theory provides a gaming strategy for tactical operations plus increased maneuvering capability. Further combinations with optimal control information and decision theory can yield a capability for maneuvering dynamics with reduced workload and substantial reduction in cost.

Another thrust could be the combination of sensor augmentation blending with active control technology and task oriented controls. This could achieve multi-state control with increased positioning accuracy and increased performance using full capability of the six-degree-of-freedom control.

Combining some of the work in bio-medical research such as, thought control implementation with multi-function displays using holography and the graphics capability, yields a potential for generating terrain projections and the medium for a synthetic visibility capability.

Figure 4 illustrates a double-span, single-thickness (referred to as a 2A size) module of 80 pins, and which represents an 8-Bit Central Processing Unit constructed from two 2901 microprocessor devices.

Appropriate combinations of the advanced mapping techniques and the accuracy available from these techniques with on-board positioning capability postulates the capability of creating a grid positioning system which has significantly increased accuracy of fixed target location. It further provides an autonomous precision time-space positioning capability. By re-combining all these elements, a potential exists for achieving a true night, all-weather capability which is compatible with the command control environment and has complete autonomous capability at considerably reduced cost from attempting to do these elements in the non-integrated manner.

The next chart illustrates the interrelationship between guidance and control, advanced technologies, and the forcing functions that exist in the design and development of a total system. The conclusion that can be drawn is that guidance and control and associated dynamics is the dominant factor in attempting to functionally integrate these elements into a manner where we can utilize the technology capability, satisfy operational needs as threat changes within the physical limitations, and provide a mechanism for reducing system cost.

I visualize future systems being conceptually structured as shown in this next chart which indicates that guidance and control will have a dominant role in the functional structuring of advanced system because of the closed-loop dynamics, control theory and the design methods required to blend the sensor-vehicle-kill mechanism in a manner that can effectively use multi-sensor inputs and the vehicle's dynamics capability to achieve the precision night, all-weather delivery capability. This, then, can be physically implemented using digital information processed data systems technology discussed earlier.

By way of conclusion, I would like to say again that there are many other considerations in the field of guidance and control than I have eluded to here. I was very impressed in looking over your program for the next four days in that it addresses many of the areas discussed and projected future technology trends. I realize that I have taken you through this rather quickly and with very few words. Nevertheless, I feel that the steps we are taking are in the proper direction. I also think how well we do our jobs in designing and structuring these systems can have a very profound effect on any major air/land battle of the future. I wish you a very enjoyable and very productive symposium this week. Thank you very much.

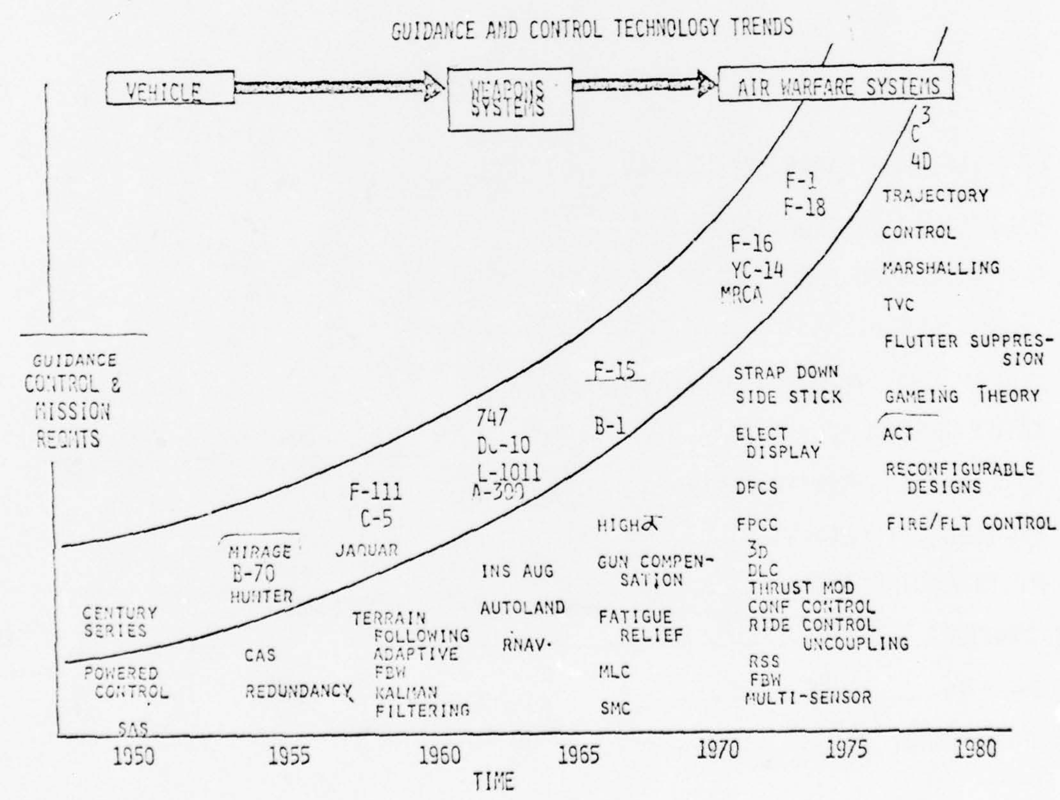

FIGURE 1

ELECTRICAL REQUIREMENTS

By adopting certain widely employed industrial standards and design practices, specific design guidance was established for SEM developers, thereby concentrating design

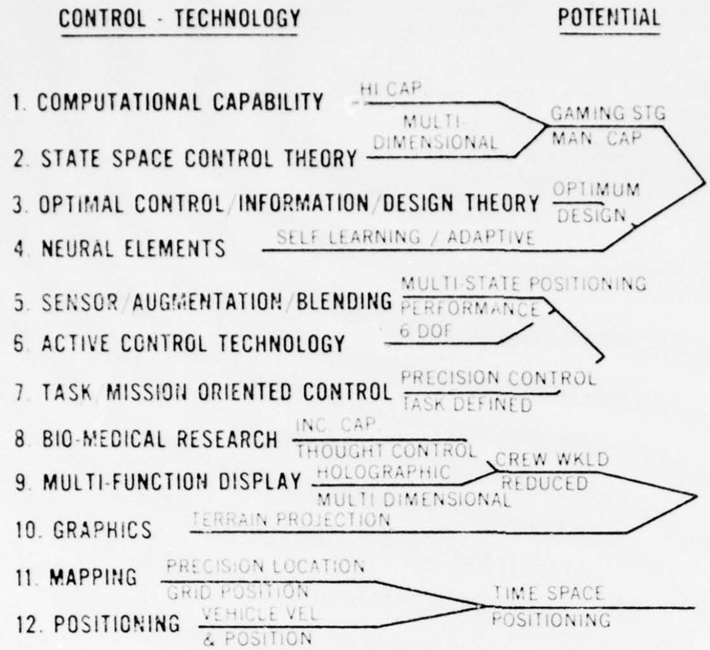

FIGURE 2.

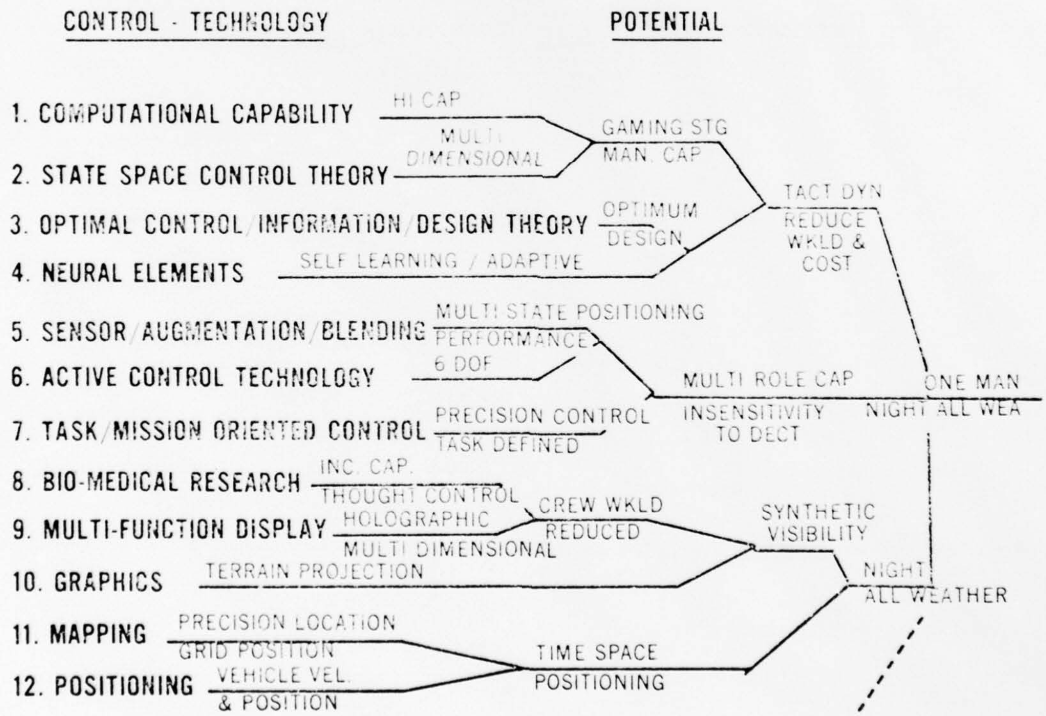

FIGURE 3.

FORCING FUNCTIONS

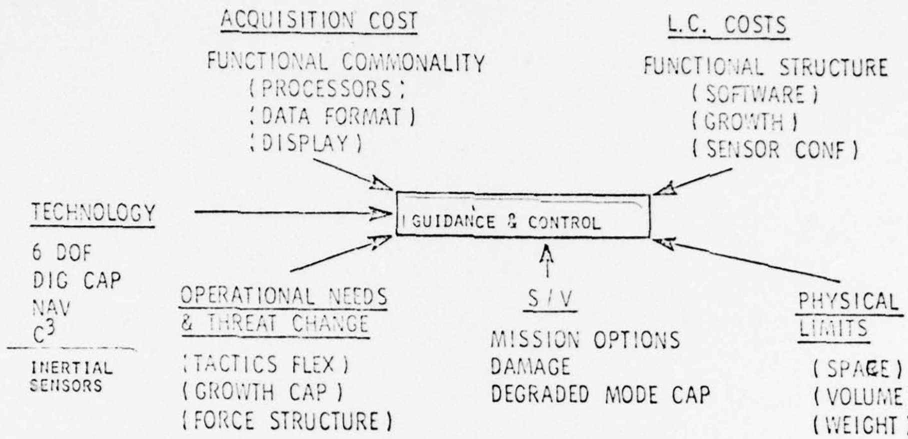

FIGURE 4.

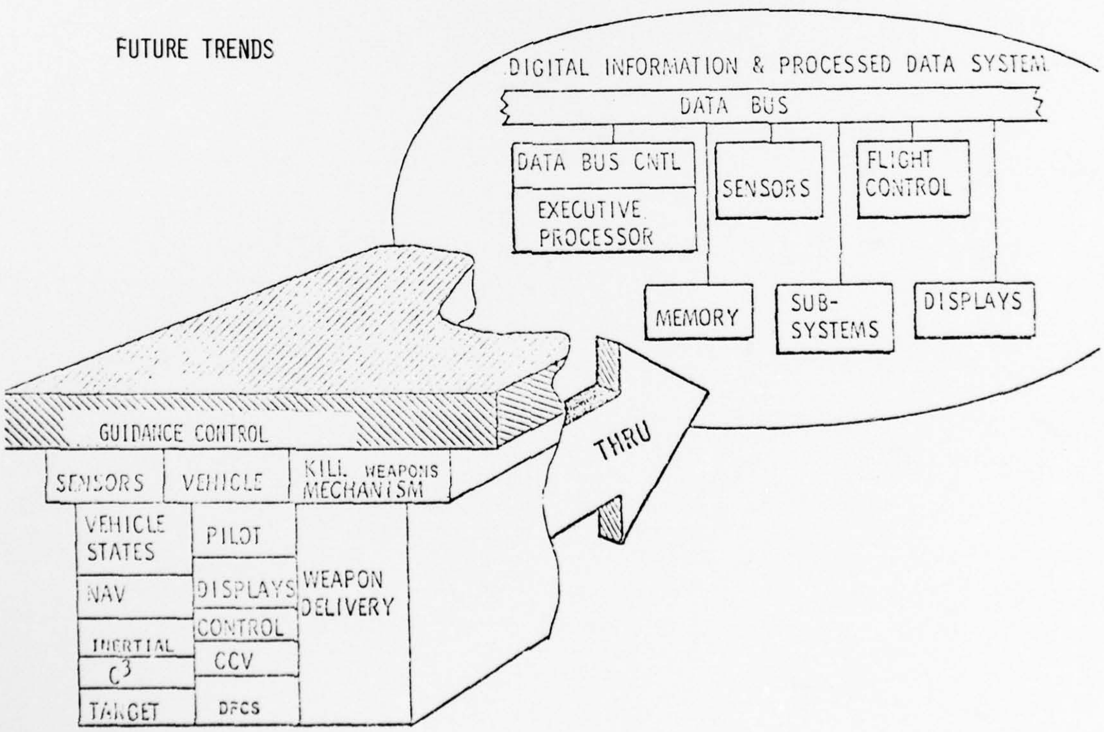

FIGURE 5.

2-1

NEW WEAPON CONCEPTS DEVELOPED FROM ADVANCED NAVIGATION GUIDANCE AND TARGETING TECHNOLOGY

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ABSTRACT

Technology advancements over the past few years now make it possible to develop improved air-launched guided weapon systems where the man, aircraft, and weapon can be integrated to provide highly responsive and effective capabilities against a large portion of the tactical target spectrum from standoff positions and under adverse weather conditions. These capabilities can greatly improve our aircraft and weapon survivability while increasing sortie effectiveness. This paper is mostly concerned with advanced navigation, guidance and targeting technologies with emphasis on those systems that can be integrated at a higher functional level to improve tactical strike capability and hopefully reduce life cycle costs.

The main threat that the concepts in this paper address is the massed attack of mobile targets. The concept of using single launched unitary weapons against single selected targets in a direct attack scenario is believed to be highly ineffective. Ideally, a weapon system should be capable of multiple target kills per weapon launched from a standoff range.

A highly integrated "Smart-Package" concept is presented in this paper showing how advanced imaging systems and data processing techniques could be used to develop an automatic system for searching, detecting, identifying and selecting targets, and for controlling the simultaneous attack of a group of weapons. Each individual weapon of the group launched would be directed to a specific target. This approach could lead to multiple kills per weapon system launched and greatly reduce the pilot work load.

Several weaponization concepts are presented to illustrate optional applications of the smart-package. In each case the smart-package is selecting targets in real time as the weapon system flies over the target area.

There are some technology voids in these concepts; namely, the data processing algorithms for false target discrimination are not fully developed. Hopefully, these problems can be solved in the near future and the concepts presented here can be further pursued.

I. INTRODUCTION:

The current threat indicates that an air-launched guided weapon system capable of defeating a massed attack of mobile targets is highly desirable. The concept of using single launched unitary weapons against single selected targets in a direct attack scenario is believed to be highly ineffective since only a small percentage of the targets could be killed and aircraft attrition might be very high. Ideally, the weapon system should be capable of multiple target kills per weapon launched from standoff range. A good effectiveness objective would be to attack and kill on the order of ten targets per weapon launched; and since some aircraft might carry several weapons, the sortie effectiveness could be very high.

This paper describes a technical approach that could be taken for development of a system for automatically searching, detecting, identifying and selecting targets; and for controlling the simultaneous attack of a group of guided weapons (missiles).[1] The heart of the system would consist of multispectral imaging sensors and real-time imagery processors. Each individual weapon of the group launched would be directed to a specific target which was selected from the real-time imagery processor.

II. DISCUSSION OF THE PROBLEM:

Multiple and simultaneous air-to-surface attack of massed mobile targets using guided weapons is believed to need one of the two following design approaches to maximize the number of kills: (1) Design each weapon of a group to be very smart for target search, detection, identification and acquisition; each being capable of rejecting false targets and communicating with all other weapons during the attack to prevent everal weapons from selecting the same target; or, (2) Design a "Smart Package" for the delivery vehicle which can view the target area and collect high resolution data for automatic detection, identification and acquisition of many targets and can reject false targets. The smart package would then compute the lines-of-sight to selected targets and direct a group of relatively simple weapons to individual targets.

The first of the above design approaches is believed to be a high technical risk and high cost program because of the weapon complexity. Furthermore, it is probably not achievable to a satisfactory level, due to the many problems associated with the individual weapons finding and locking-on different targets after they are launched. Assuming that the individual weapons would have either TV, IR or RF sensors capable of search, detection and acquisition, the main design problems are due to the following:

a. Unpredictable formation of targets - This causes search design problems.

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[1]This paper describes some ideas of the author. It does not represent a development committment of the USAF.

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b. Spacing variation of targets - This causes instantaneous field-of-view problems particularly for the RF sensors.

c. Relative location of targets at time of launch - As a worst case this can increase the aerodynamic and propulsion demands.

d. Wide variation of target signature - This will tend to increase the seeker complexity.

e. Passive countermeasures - Seekers must be capable of recognizing and rejecting and this increases their complexity.

f. Active countermeasures - Same comment as e. above.

g. Natural false targets - Same problem as f.

h. Probability of several weapons selecting the same target and insufficient time to select another target if given the opportunity - This is highly probable if there is no weapon-to-weapon communication. This approach is believed to be very expensive due to the many functions that each missile must perform and it is doubtful that it can be done at all within the volume and weight constraints imposed on air launched weapon systems.

III. ADVANCED CONCEPTS:

The preferred approach, and the subject of this paper, is the single smart-package which could support a group of small and relatively simple guided weapons. To accomplish this the carriage vehicle (aircraft, missile or drone) would be highly instrumented with the smart-package to preselect targets, command line-of-sight information to the small guided weapons and control their launch. Figure 1 illustrates the subsystems of the smart-package. There should be two imaging systems covering three spectral bands of interest. One of the imaging systems, having high resolution would cover two infrared spectral bands (3-5 μ and 8-14 μ) and the other, having much less spatial resolution, would be a scanner working at about 35GHz. Imagery from the sensors is converted to a digital format in real time. A digital processor would analyze the imagery in real time to select targets that the weapons will be directed to attack. A rather complex digital imagery processing program would be developed containing the following as a minimum.

a. Pattern Recognition Algorithms

Detection can be enhanced at long range by programming typical target patterns and correlating the imagery with these patterns. Areas to search for more information as range closes can be selected.

THE SMART PACKAGE

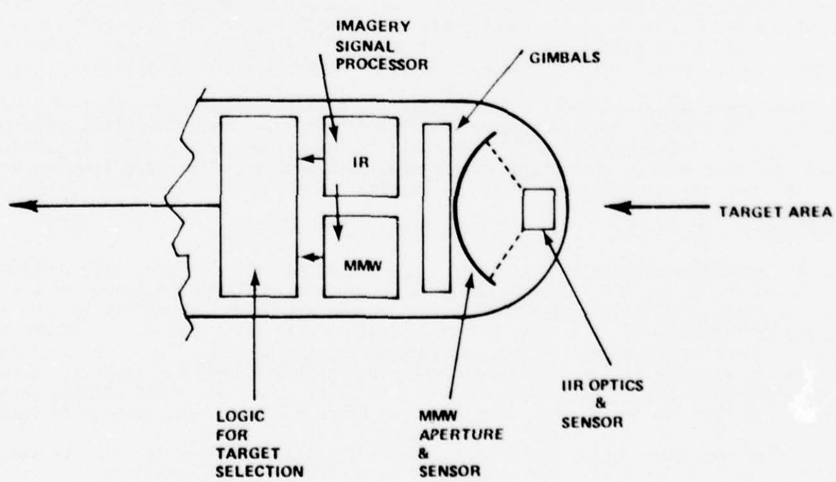


FIGURE 1

b. Spatial Frequency Algorithms:

Recognition of targets can be enhanced by programming target and non-target dimension algorithms.

c. Temporal Frequency Algorithms:

Identification of targets can be enhanced through a knowledge of the temporal frequency content which is related to vehicle shape and edges. Target motion can be detected and evaluated with these algorithms.

d. Spectral Frequency Algorithms:

Additional recognition and identification confidence can be obtained through a knowledge of the spectral frequency signatures of vehicles. Correlations for further confidence in target selection can be obtained from the multi-spectral signatures.

e. Counter-countermeasure Logic:

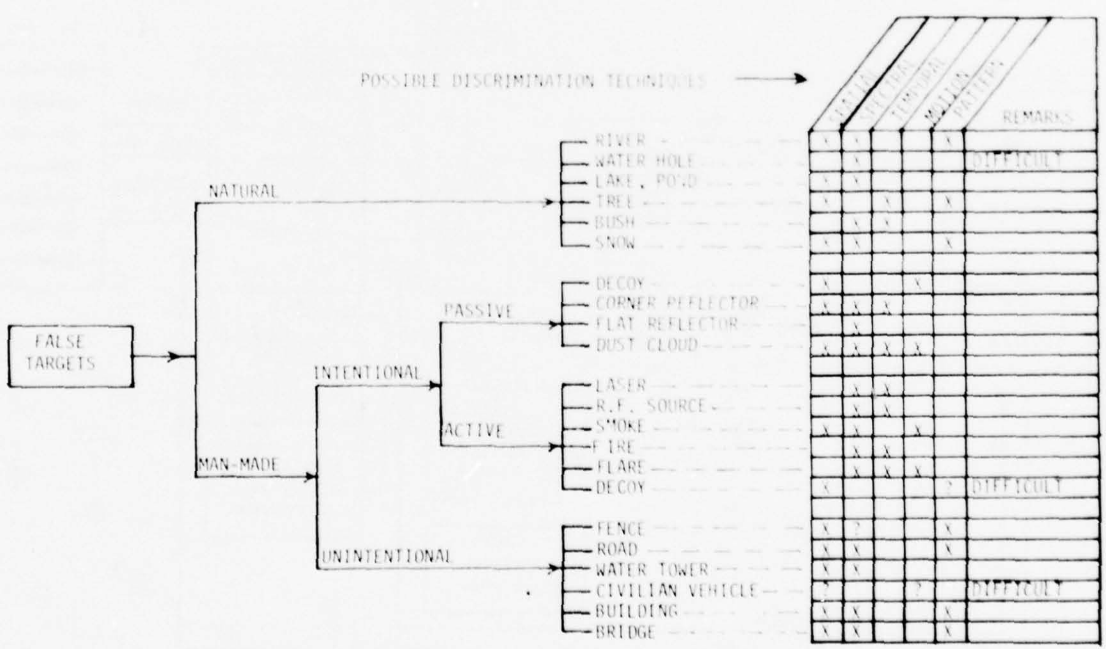
Information from a, b, c, d, above will allow algorithms to be developed to detect and identify any countermeasures and natural false targets to prevent missiles from attacking them.

f. Target Motion Algorithm:

Motion can be detected in each imaging system and this can further assist to identify the true targets.

False target discrimination using the above algorithms is the technical challenge for the smart-package. This will be a difficult task to accomplish in a very short time since the ratio of false targets to true targets may be very high. A false target "tree" showing some typical targets that an RF or IR sensor might acquire is shown in Figure 2. Also shown are the discrimination techniques that can be used with the appropriate algorithms to reject these targets. Spatial frequency discrimination appears to be the most important technique that can be used, followed by spectral and temporal frequency techniques. Pattern recognition may be the most powerful technique but also the most difficult to accomplish.

RF AND IR FALSE TARGET DISCRIMINATION



NOTE: TYPICAL FALSE TARGETS ARE SHOWN.
NO EFFORT HAS BEEN MADE TO SHOW ALL FALSE TARGETS.

FIGURE 2

As targets are selected by the imagery processor, the lines-of-sight (azimuth and elevation angles) to the nearest targets which are within the aerodynamic range of the missiles are sent to missiles in the group to command the seekers to "look" at the selected targets. Seeker operational parameters such as sensitivity, field-of-view and slew rates are programmed in the smart-package to be certain that the seekers can acquire the selected targets when they are commanded. The actual lock-on could be accomplished before launch if a system was designed similar to that shown in Figure 3. In this case, the missiles would be locked-on and launched in small groups following lock-on confirmation as the carriage vehicle passes over the target area. Figure 4 illustrates a typical smart-package system concept for a group of missiles.

In this concept the missiles would not have to be very smart. The seekers would be simple trackers with good tracking logic but no target selection or discrimination logic since the smart-package in the carriage vehicle does the target selecting. Field-of-view can be kept small and high resolution imaging trackers would not be necessary. This concept will greatly reduce the cost of the missiles and it should greatly improve sortie effectiveness.

The main problem with this concept is not hardware technology. We have the imagery capability and the computer technology. The problem is that we do not have sufficient multispectral target and background (false target) signature data. We need this information so that we can develop and program the algorithms in the imagery data processor for target detection, identification and selection.

This paper has been prepared to indicate an approach for simultaneous attack of multiple ground mobile targets. The same concept of using a smart-package to manage group of guided missiles could be used with other munitions and other delivery concepts. Figure 5 illustrates a concept using semi-active laser guided or beam rider weapons where the beams are controlled by the smart-package. This concept could lead to a very low cost missile but it has the disadvantage of having to illuminate the targets.

Figures 6 and 7 illustrate a concept using self-forging slugs (SFS) that are aimed at selected targets by gimbals which are commanded from the smart-package. This could be a returnable drone where the SFSs are reloaded for another mission. Figure 8 illustrates a modular concept where the smart-package can be employed in different ways to control several configurations of weapons.

TYPICAL SMART PACKAGE

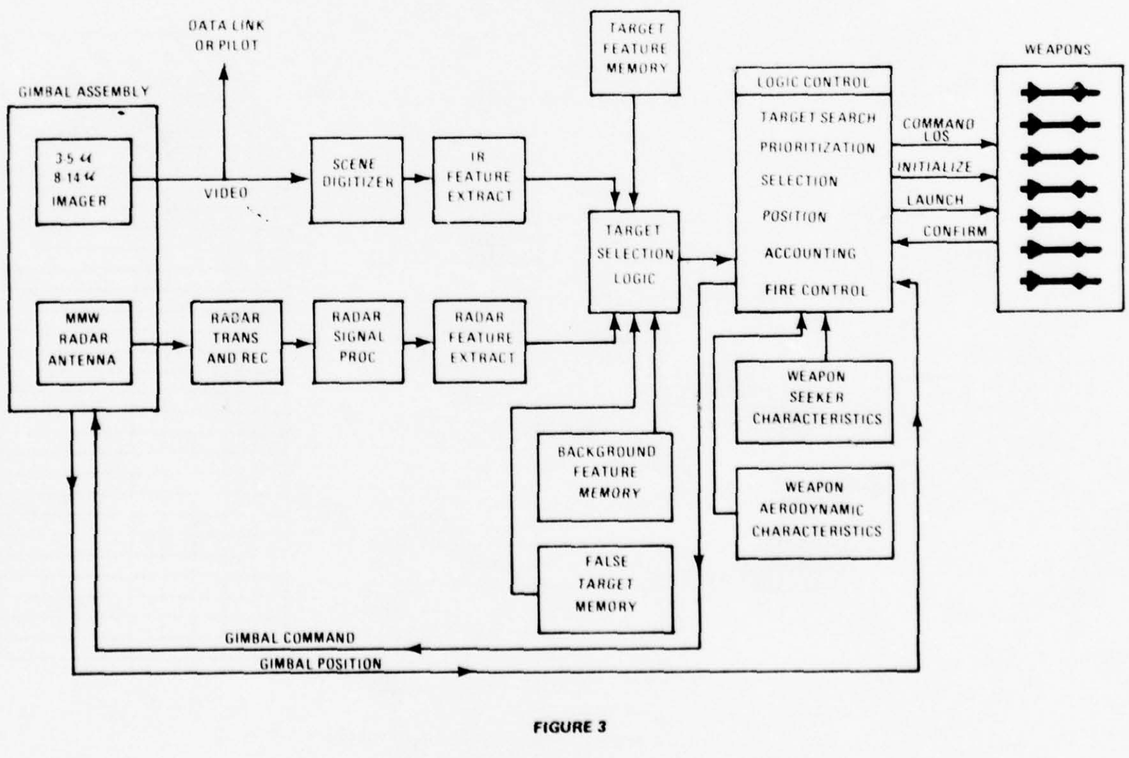


FIGURE 3

TYPICAL SMART PACKAGE SYSTEM CONCEPT

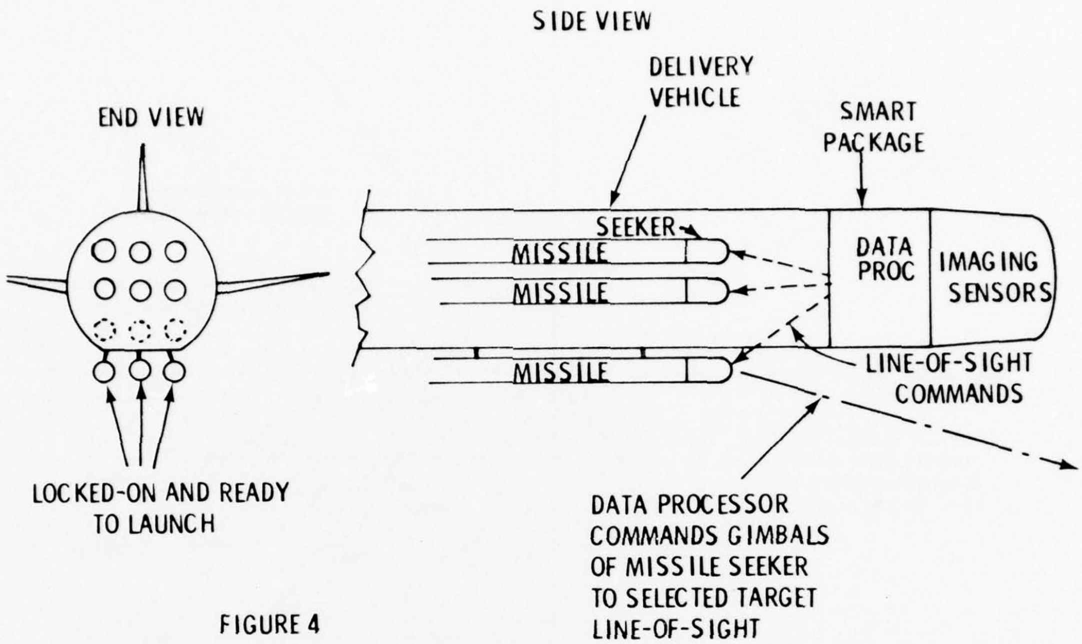


FIGURE 4

AUTONOMOUS LASER ANTIARMOR WEAPON SYSTEM

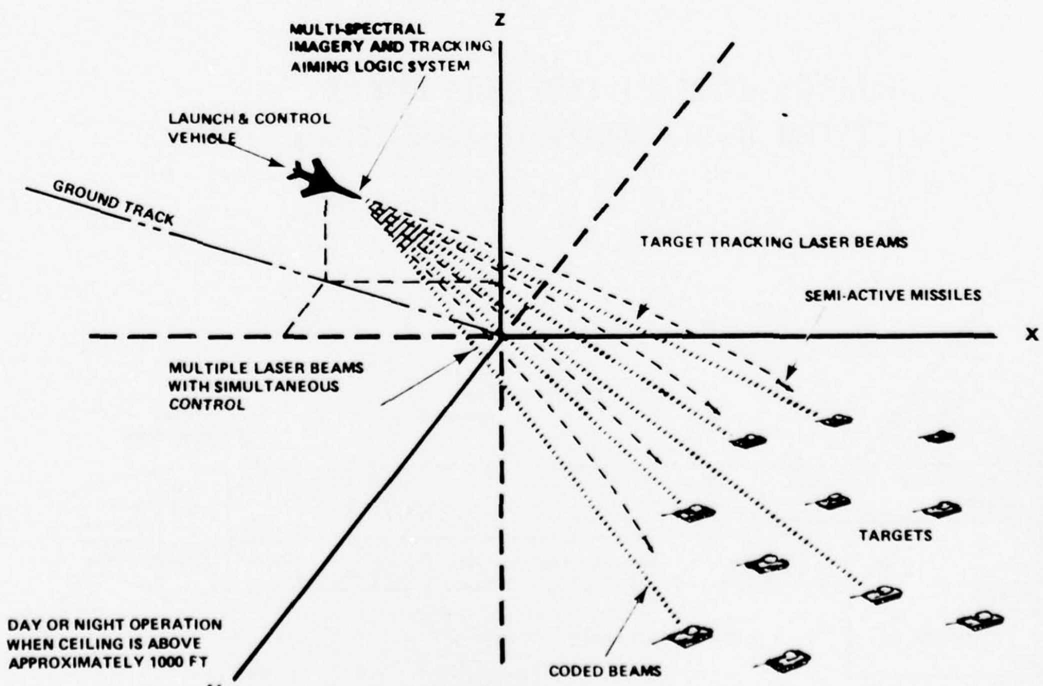


FIGURE 5

2-6

SMART PACKAGE WITH GIMBALED SFS's

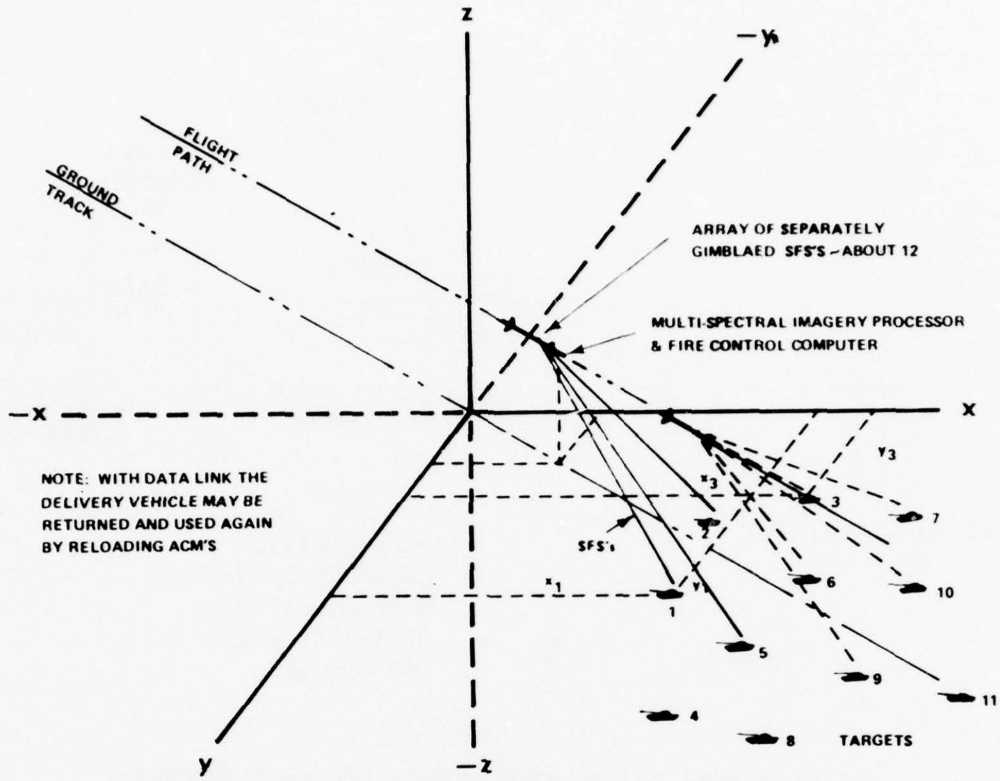


FIGURE 6

WEAPON CONCEPT FOR ANTI ARMOR SYSTEM USING MULTIPLE AIMED SFS's

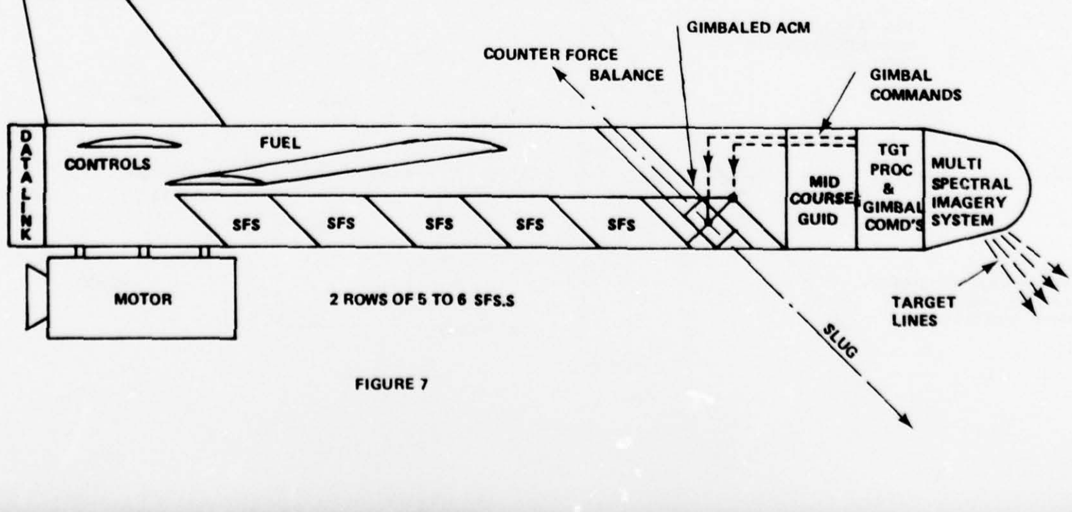


FIGURE 7

SYSTEM OPTIONS

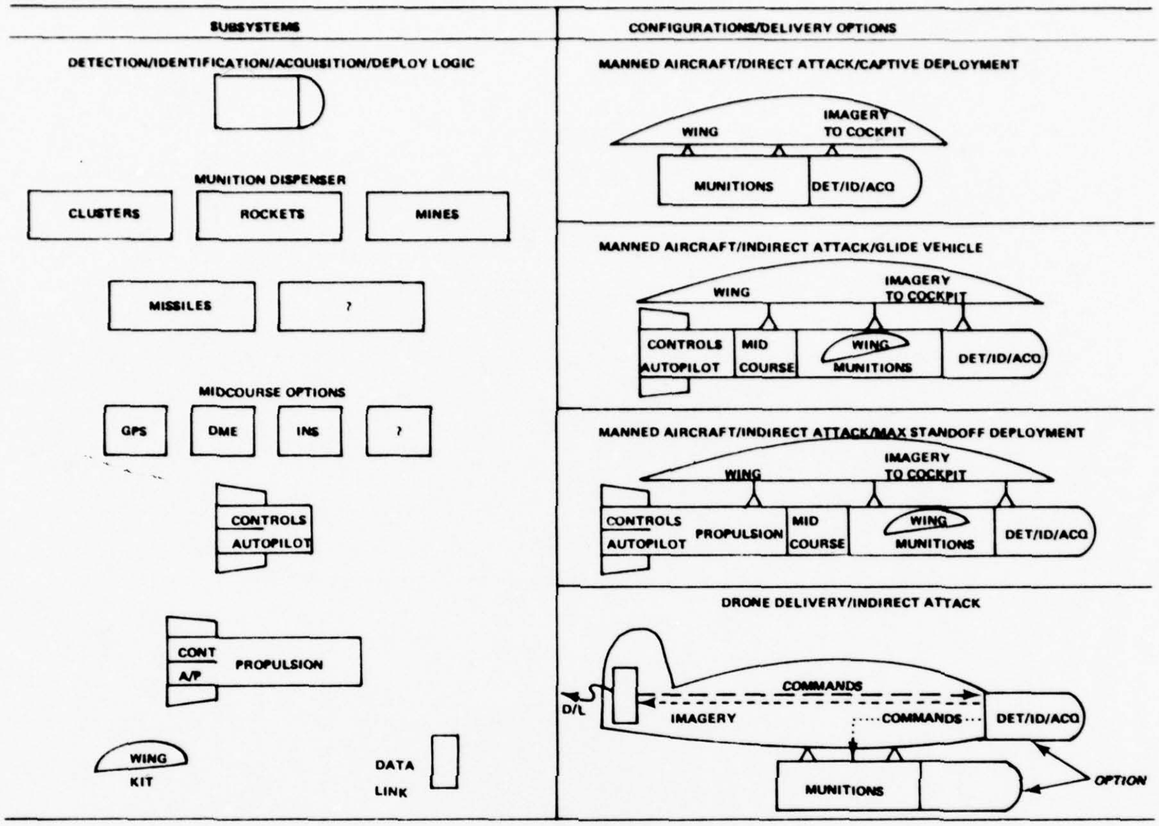


FIGURE 8

IV. CONCLUSIONS:

A very smart weapon system, capable of functioning without a man-in-the-loop for each target, is believed to be required for successful attack of the threat envisioned in this paper. The smart-package approach should be capable of autonomous operation and there appears to be a wide variety of employment options.

COST AND DESIGN ADVANTAGES DERIVED FROM THE
STANDARD ELECTRONIC MODULES PROGRAM

4-1

by
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SUMMARY

The Standard Electronic Modules Program is a highly successful design standardization program that is commanding considerable attention within the U.S. Department of Defense as a result of its achieving significant cost and reliability results. This program establishes a rational discipline for the development process for military electronic systems by providing families of functional electronic modules which are already developed, documented, and qualified, and for which a wide industrial base exists. Although this program has been heavily oriented at resolving system maintenance and logistical support problem areas, it nevertheless constitutes a readily available and highly effective "building block" approach for accomplishing research and development functions.

INTRODUCTION

The word "standardization" to the Research and Development scientist or engineer, brings fear of the heavy hand of bureaucracy restricting design flexibility which is so necessary in furthering advances in technology. Additionally, it can bring concern of too rapid obsolescence or too great constraint upon size, weight, and performance. These factors must be carefully considered when structuring and applying a technology program for standard hardware development. The advantages of some standards, however, even in the R&D phase, can be considerable when compared to whatever limitations they may impose.

Most efforts at hardware standards are initiated by the hardware users who find that their logistics problems are overwhelming and, consequently, demand commonality of supporting components even with restrictions to system capability. Although the purpose of the Standard Electronic Modules (SEM) Program is to provide a much needed aid to the hardware user, it nevertheless offers the R&D designer a substantial and readily available technilogy base with only very minor restrictions.

Even when a major new weapon system is developed, most of the supporting electronics require no new concepts--control logic, amplifiers, and other similar components can be drawn from existing technology. The SEM Program provides a broadly used, high reliability series of functional electronic components of great flexibility for application to new systems design. This permits the designer to develop only the components needed to demonstrate the new principles and saves the cost in time and money needlessly spent on designing electronics not essential to the proof of the R&D objective.

BACKGROUND

In following the evolution of semiconductor electronics, one finds that they typically share a common life cycle (see Figure 1). A developer of a new system, recognizing that a period of ten years may be required before significant use of the system in the field occurs and that for a following period of twenty years logistic support is to be needed, must make a decision on semiconductor circuits of a technology that is a complex balance between too early obsolescence and too high a development risk. This decision, while of concern to the R&D developer, is of paramount concern to the final system hardware user. This situation ultimately provided the impetus for the establishment of the SEM Program and offers a compatible methodology for finally bridging the gap between the R&D and system user communities.

† Published in CP-257 (Supplement) Classified.

* Not available at time of printing.

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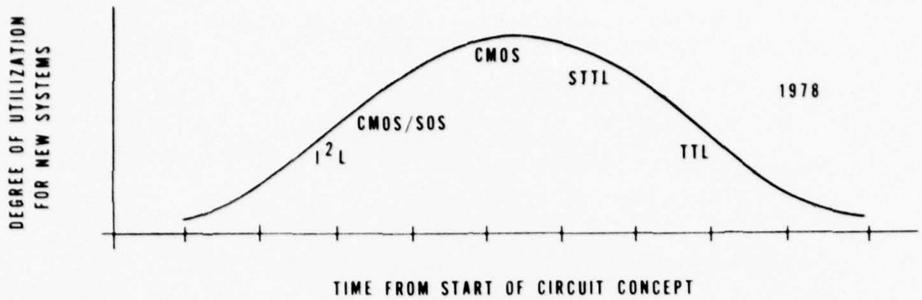

Figure 1. Technology Life Cycle

PROGRAM DESCRIPTION

The SEM Program is an electronic module standardization program that is being actively promoted throughout the U.S. Department of Defense. The purpose of this electronics standardization program is to establish a family of reliable electronic modules that will reduce the cost and facilitate the design, production, and logistics support of military electronic systems.

The concept of the SEM Program is based upon the principle of limiting redundant design through the use of standard functions, thus achieving cost benefits through consequent large production volumes and wide competition. As the program continues to gain further acceptance, the cost and performance benefits will become even more significant.

The basic objectives of the SEM Program are:

- partitioning electronic functions so that they can be common to a majority of equipment applications

- documenting modules with functional specifications (to preclude dependence upon specific vendor, design, or technology), enabling long-term availability and cost savings through vendor innovation and competition

- achieving high reliability through stringent quality assurance requirements for module design and production

- discarding modules upon failure (made possible due to high reliability and low cost)

- providing flexible modular mechanical packaging requirements which accept various circuit and packaging technologies and adapt to various equipment mechanical configurations

- easing the logistics support burden on the congested supply system by extensive inter-system commonality of a limited number of modules

- prohibiting the use of components or processes in module specifications that would inhibit competitive procurement over an extended period

- establishing two separate classes of environmental requirements that meet the needs of most shipboard, shore-based, and avionics applications

- providing configuration control policies to ensure the interchangeability of all modules of the same function and key code.

Unlike many other standardization efforts, the SEM Program is a dynamic program, which is open-ended and can accept new module functions and all technological advances. By eliminating redundant design efforts and the need for repair facilities and large inventories of unique parts, the SEM Program significantly reduces the cost of the key elements in equipment life cycle cost.

SEM CONFIGURATION

The basic standard module configuration is the single-span, single-thickness (1A size) increment. The principle dimensions were derived based on the amount of circuitry required to perform various functions, the maximum number of necessary interface connections, the size of the keying and retaining mechanisms, and the likely tolerable cost for a "throw-away" module. These considerations led to the development of a basic module increment with overall dimensions of 2.62 in. (66.55mm) in width, 1.95 in. (49.53mm) in height, and 0.290 in. (7.37mm) in thickness (see Figure 2). There are provisions for module growth increments for use in the expansion of modules of multiple span and thickness. Modules can be increased in span by increments of 3.00 in. (76.20mm) and in a brief listing of a few of those technologies that appear significant, particularly if appropriate combinations are made. If one examines computation capability, it is well recognized that we will be seeing several orders of magnitude increases within the next two decades. The anticipated computation

thickness by increments of 0.300 in. (7.62mm) (see Figure 3). Figure 2 also identifies the features of a typical SEM Program module:

4-3

Fin - Provides the marking surface, extraction interface, and thermal interface for heat dissipation.

Extraction Holes - Provide the common interface by which modules are removed by means of an extraction tool.

Guide - Provides a surface for guiding the module into the mating connector and mounting structure as well as a thermal interface for heat dissipation.

Contacts - Bladed contact pins based on a 0.100 in. (2.54mm) grid system form the module connector. Each module increment may have a maximum of 40 contacts.

Pin Skirts - Provide a convenient protective and marking surface for module contacts.

Key Pins - Provide the means of uniquely polarizing modules of different functions to ensure that they are not wrongly inserted into the mounting structure.

A three-letter key code marked on the fin defines the configuration and rotational positions of two uniquely configured keying pins inserted into each module header surface. SEM Program modules having the same key code must be both mechanically and electrically interchangeable with each other.

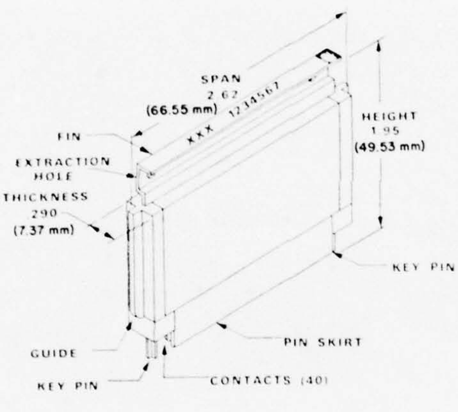

Figure 2. Basic Module Configuration

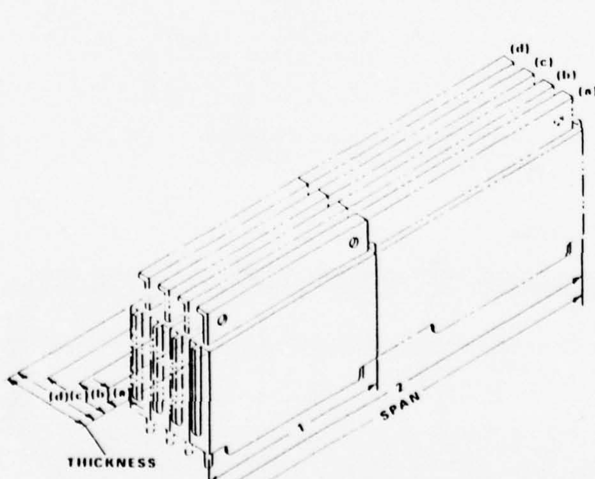

Figure 3. Multiple Growth Method

projections and the medium for a synthetic visibility capability.

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Figure 4 illustrates a double-span, single-thickness (referred to as a 2A size) module of 80 pins, and which represents an 8-Bit Central Processing Unit constructed rom two 2901 microprocessor devices.

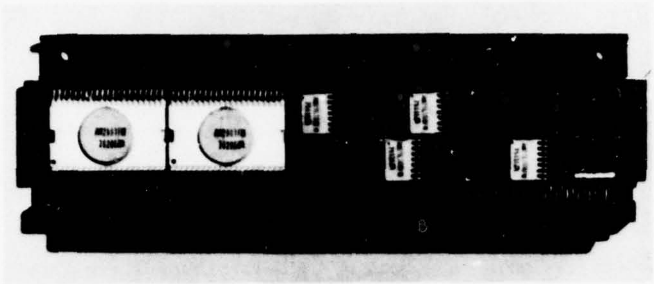

Figure 4. 8-Bit Central Processing Unit

Figure 5 depicts an array of SEM, illustrating modular growth in both the span and thickness planes.

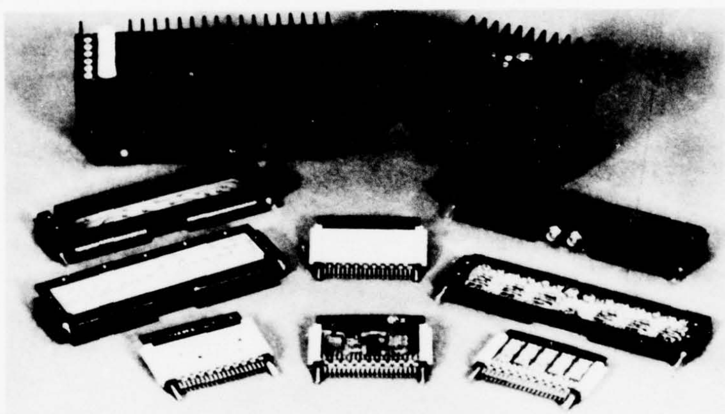

Figure 5. Module Array

The SEM Program is currently investigating Several proposed modifications to the current mechanical form factor for SEM which would improve its capability for accommodating functions of increased density, without losing physical compatibility with existing program requirements. By increasing the available component mounting area, 2A module designs will be able to accommodate up to 18 dual-in-line devices and, as a result of increased effective conductive interface area between the module and the card cage, will be capable of dissipating increased thermal loads of up to 10 watts.

Figure 6 depicts a prototype of a 9900 microprocessor module fabricated by Texas Instruments Inc., and comprised of more than 30 leadless chip carrier devices and constitutes of 15,000 gate equivalents.

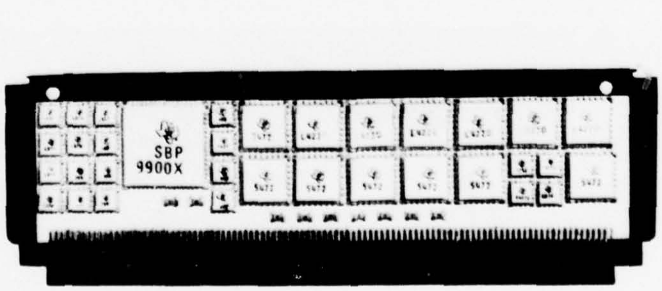

Figure 6. Proposed Improved SEM Configuration

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ELECTRICAL REQUIREMENTS

By adopting certain widely employed industrial standards and design practices, specific design guidance was established for SEM developers, thereby concentrating design efforts in a manner that would yield compatible module designs for use by others. The following is a summary of such requirements and guidelines:

Functional Partitioning - Each standard module function is a complete function or group of functions, specifiable and testable without dependence upon another module. Multiple function capability may be incorporated within a module by means of either pin programming or voltage control.

Logic Levels - Standard digital logic levels have been established.

Module Connector Requirements - Module connectors are designed to meet stringent environmental as well as electrical requirements. Requirements controlling such factors as contact current rating, contact working voltage, contact resistance, dielectric withstanding voltage, and durability are provided.

Module Contact Pin Assignment - Module power supply, circuit ground, frame ground, and signal lines have been assigned to specific contact pins on the module connector. These requirements have been established to permit the use of commonly used power and ground bussing techniques, thereby enabling the simplification of back plane wiring.

MODULE FAMILIES

Although modules are heavily oriented towards digital logic functions of varying complexity of all technologies with an established and supportable industrial base (e.g., adders, shift registers, to highly functional microprocessor and memory modules), the program also offers an extensive variety of interface circuits, converter modules, analog modules, and power supplies.

RELATED HARDWARE

Being restricted to families of functional circuit modules, the program has nevertheless spawned an extensive array of associated hardware components which have become de facto standards in their own right. As illustrated in Figures 7, 8, and 9, such hardware components range from module components to module mounting structures, back plane wiring assemblies, and interconnection components. The SEM Program and various Navy laboratories and industrial concerns have also established SEM and associated piece-part pools, for which the objective is to provide R&D and prototype developers with limited quantities of SEM and related piece parts on a quick-reaction basis and at prices which reflect volume production discounts.

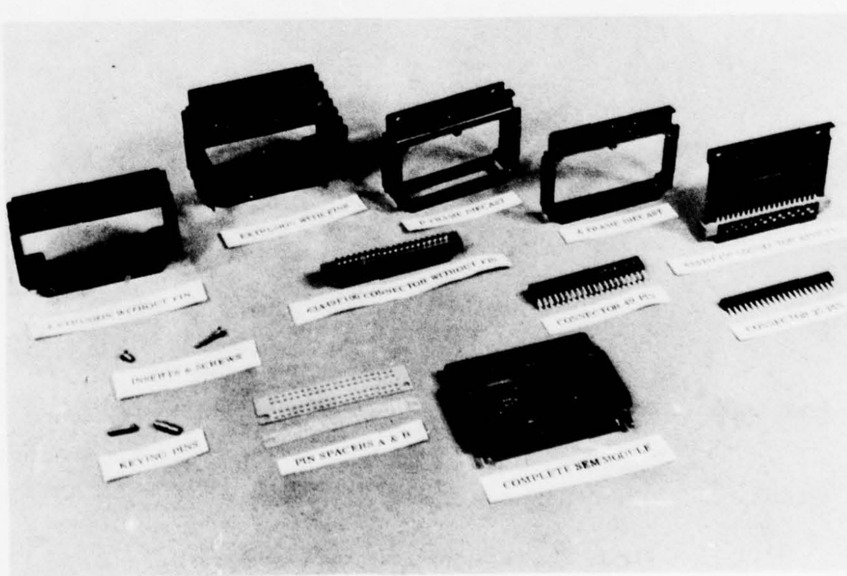

Figure 7. Typical Module Construction Components

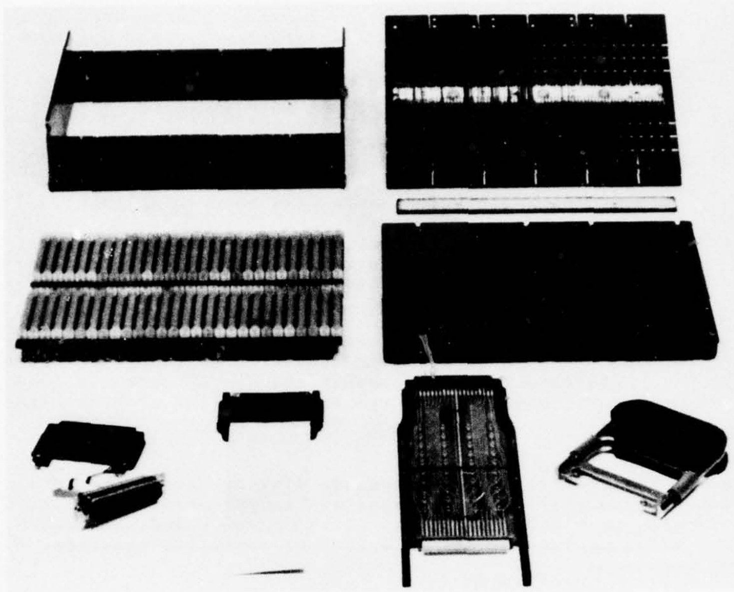

Figure 8. Related Hardware Components

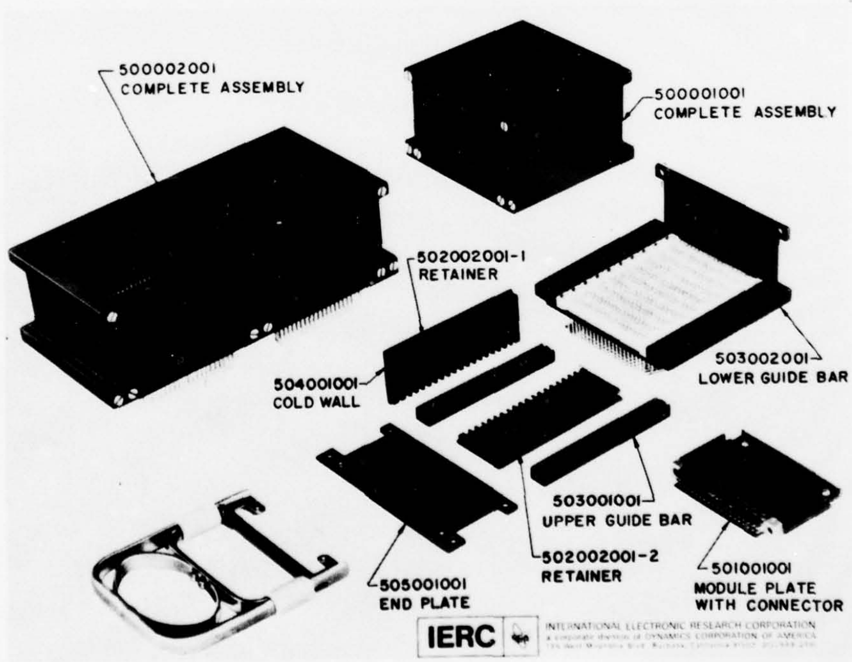

Figure 9. Mounting Structure Components

To further facilitate system design application of SEM, a broad range of computer-aided design software programs are available. These programs range from Boolean logic equation translation, module placement and wiring layout, to thermal design programs for air and conduction cooling.

Through the employment of SEM and such proven and widely available hardware and software aids, the R&D development process can indeed be simplified, not only in shortening lead times but also in substantially reducing cost. This is especially true in preproduction phases, where system prototypes become in essence production hardware, requiring minimal transitional redesign for production. In fact, SEM and its associated "building blocks" have become an invaluable tool for R&D by permitting the concentration of resources upon research objectives rather than diverting them to the more ordinary aspects of hardware development.

PROGRAM ORGANIZATION

The SEM Program is organized with the Naval Electronic Systems Command (NAVELEX) designated as the Technical Management Activity; the Naval Avionics Center (NAC), Indianapolis, Indiana, of the Naval Air Systems Command, serving as the Design Review Activity; and the Naval Weapons Support Center (NWSC), Crane, Indiana, of the Naval Sea Systems Command, serving as the Quality Assurance Activity. The Naval Electronic Systems Command Acts as the agent for the Chief of Naval Material in managing the program within the Department of Defense.

The Technical Management Activity is responsible for the operation of the overall SEM Program and:

- establishes SEM Program objectives consistent with Department of Defense and Navy standardization requirements

- organizes, implements, and controls program requirements necessary to meet Navy objectives

- organizes and directs the SEM Program work at field activities

- promotes the SEM Program within the Navy and sponsors related activities.

The Design Review Activity has as its primary responsibility the review and classification of each SEM proposed for development. As a result, they classify special or standard SEM Program module key codes and assign specification numbers for their development.

The Quality Assurance Activity is primarily responsible for the qualification of module designs and vendors. They establish and maintain SEM Program quality assurance requirements; perform initial and periodic module production qualification testing; perform correlation of SEM Program vendor test equipment; perform failure analyses; and compile SEM module reliability data.

SEM PROGRAM SPECIFICATIONS

The mechanical and environmental requirements for SEM Program modules are specified in documents which also describe the electrical, functional, and reliability requirements for each module type. These specifications are prepared in accordance with MIL-STD-1378, Requirements for Employing Standard Electronic Modules Program. The specifications for SEM are prepared originally by the developer for approval and control by the SEM Program. Normally, they specify requirements for form, fit, and function rather than detailed design requirements. This documentation technique permits module vendors to produce modules without unnecessary restrictions on components and specific design details, as long as the functional requirements of the specification are maintained.

Though the details of design are left to the module developer, it is essential that performance standards and reliability requirements be maintained. Therefore, the basic mechanical configurations from which the designer may choose are covered by a design standard; MIL-STD-1389, Design Requirements for Standard Electronic Modules. This standard also describes the design requirements which will enable modules to satisfy the quality assurance requirements specified in MIL-M-28787, General Specification for Standard Electronic Modules.

In addition to these standards and specifications, several other documents have been prepared cataloging available modules and describing their function and application. Figure 10 shows the SEM Program documentation organization. Under the auspices of the Council of NATO Armament Directors, the AC/67 is forming a Special Working Group to deal with the subject of module standardization. It is anticipated that these documents will be available from that group.

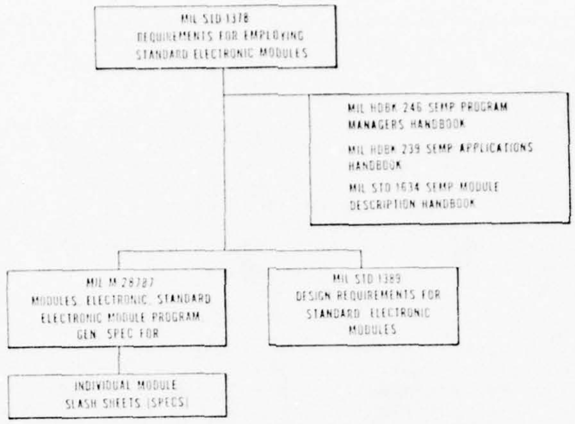

Figure 10. Documentation

STATUS OF THE SEM PROGRAM

As a result of extensive use over the last ten years, the SEM Program has evolved to where it consists of approximately 300 module types, with more than four million modules committed to production. To date, SEM Program modules have been applied in over 80 system applications, spanning virtually every operating environment. Because of the large number of modules required in already committed SEM Program systems, combined with an even brighter future for the SEM Program, approximately 15 module vendors have become qualified.

RELIABILITY EXPERIENCE OF SEM

The SEM Program quality disciplines have resulted in hardware that has demonstrated reliability far beyond initial expectations. For example, on a particular shipboard fire control system which has been deployed long enough to have statistical significance in reliability results, digital modules are demonstrating a failure rate of less than .04 failure per million hours. This is compared to the original estimate of 1.9 failures per million hours, and the observed failure rate of similar module types (but which do not adhere to the SEM Program disciplines) of 5.5 failures per million hours. In recent tests on a submarine sonar system, a system MTBF (mean time between failures) of 2,100 hours was demonstrated compared to the estimated 1,368 hours. In this demonstration, no failures occurred among SEM Program modules.

The SEM Program has from its inception emphasized the importance of reliability in system design. To ensure its module reliability and ultimately system reliability, the SEM Program has done, and continues to do, what every module or system designer should do to ensure the reliability of its product. First, it uses thoroughly screened military grade parts. Second, during the module design phase, the designer is required to address reliability by performing a thorough reliability prediction. Third, the SEM Program Quality Assurance Activity (QAA) requires that each module be thoroughly documented. Fourth, the design is extensively tested by the SEM Program QAA to ensure that the module specification is correct and that the design meets the specification. In line with this, each vendor who manufactures standard modules must be qualified by the SEM Program QAA. This qualification is maintained by periodically submitting a sampling of modules for qualification testing. This ensures that the vendor's modules comply with the module specifications. As can be seen, the striving for reliability takes a continuing effort on the part of the SEM Program, with commensurate rewards occurring in the proven performance of its modules. The total QAA effort requires a considerable investment--no one will argue this point; but the payoff in improved systems availability far outweighs this expense.

COST IMPACT OF SEM

Electronic modules constitute more than 50% of system procurement costs, and thereby offer a fertile area for achieving cost reduction with SEM. Combine this with the potential discount that can be realized through large production volume procurements and the results are significant. Table 1 depicts typical discount rates as a function of quantity and applies to all electronic modules as well as most electronic components.

Table 1. Typical Discount Rates

No. of Items	0-5	200	500	1000	5000
Discount	0%	10-15%	15-25%	25-35%	35-50%

As indicated, a production volume of 5,000 units commands discount rates as high as 50%. This is significant, for without attempting to standardize and create a production base of some consequence, we have been traditionally working at the low end of the curve with the "one of a kinds."

To illustrate how the SEM Program has benefitted a typical development program from inter-system commonality and extended demand, SEM Program costing data has been compiled from the Naval Sea Systems Command AN/BQQ-5 and AN/BQQ-6 submarine sonar development programs. The AN/BQQ-5 Sonar development program was initiated in 1970 and extensively employed SEM. A total of 16,000 modules were required per system and were comprised of 138 module types. Twenty-one standard types accounted for over 12,000 of the 16,000 module total. As a result of the subsequent AN/BQQ-6 Sonar development program also employing SEM, this program was able to satisfy system requirements almost exclusively with existing modules which are common to the AN/BQQ-5 and other Navy systems. Consequently, the AN/BQQ-6 Program needed to develop only 30 new module types to fulfill its system requirements. Table 2 presents a brief tabulation of the more obvious life cycle cost savings identified on these two sonar programs.

Table 2. Savings Resulting from SEM Commonality

Area of Savings	Amount (Thousands)
Development	$2,580
Volume Procurement/Production	3,069
Residual BQQ-5 Modules	330
Supply Administration	150
Spare Support	1,500
Total Savings Quantified	$7,629

Savings shown here were based on calculations for a limited number of AN/BQQ-5 and AN/BQQ-6 systems. Actual total savings are significantly greater and relate to the actual number of systems as well as additional savings in the areas of training, testing, support equipment, and documentation. It should be noted that these factors are only the "tip of the iceberg" relative to life cycle cost savings achievable through the use of the SEM Program. Savings should go far beyond what has been identified here and, if the "obscured" items related to logistics and long-term availability can ever be properly quantified, the true life cycle cost savings for user systems will be tremendous.

SYSTEM APPLICATIONS

To illustrate the systems design flexibility of SEM and that it is applicable to virtually all system environmental platforms, several examples of SEM system applications will be noted.

Airborne Radar System

The Air Force has recently sponsored the development of two separate radar systems, both of which employed SEM. Development models of an airborne cargo aircraft radar (see SEMR - Figure 11), for the Air Force Avionics Laboratory, Wright-Patterson Air Force Base, and a Tactical Weather Radar for the Electronics Systems Division, Hanscom Air Force Base, have been delivered and are to undergo test and evaluation. In addition to employing a significant number of existing SEM, both Air Force radar systems use numerous modules originally developed by the Navy for the "2175 Modular Radar" (Prime Search Radar) development program.

Figure 11. SEM Radar System

Table 3 depicts the inter-system commonality of SEM among the various radar programs, indicating a high degree of inter-system commonality achieved.

Table 3. Radar SEM Commonality

	"2175" TYPE IC		AF SEMR		AF TACTICAL WEATHER	
	No. Types	No. Modules	No. Types	No. Modules	No. Types	No. Modules
Total SEM	66	194	68	191	51	116
Existing SEM	19	93	18	92	17	63
Candidate SEM	47	101	50	99	34	53
SEM Common to All Systems	23	98	23	80	23	52

Sonar Receiver Set

The Raytheon Company, Submarine Signal Division, Portsmouth, RI, has also recently completed the development of the AN/BQR-24 Sonar Receiving Set (see Figure 12). This equipment is a computer-based processing system for use in conjunction with the basic AN/BQR-13 submarine sonar. The heart of the system is the processing unit which contains approximately 1,200 SEM in addition to memories and power supplies, and which are all contained in a standard Raytheon cabinet expressly designed for SEM. One significant aspect of this program is the building of the first prototype unit in a production configuration in approximately 12 months. Raytheon attributes this accomplishment to the fact that SEM was used to implement the design and that the majority of these modules were already developed, qualified, and readily available to the equipment engineer. Raytheon stated that it would have normally taken 18 to 20 months to reach a production configuration if new module designs were used. After testing, the system was sent to sea where it completed a successful evaluation. Because the SEM prototype unit was in a production configuration, transition to manufacturing did not require another design iteration, thus allowing Raytheon to respond to a short production schedule.

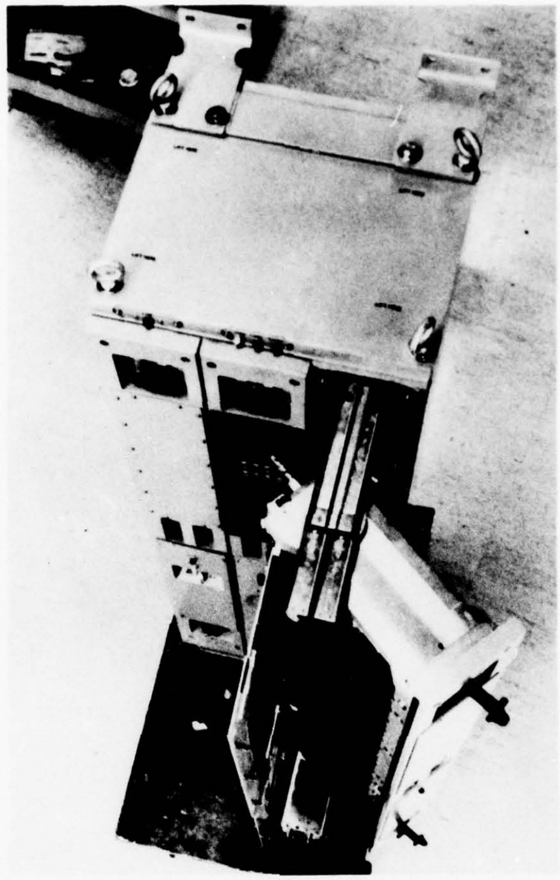

Figure 12. Sonar Receiving Set

Airborne Weapon Control System

Figure 13 depicts an airborne weapons control system designed for the A6 aircraft. A total of 91 SEM Program modules were utilized to implement this function, 75% of which were SEM standards.

Missile Guidance System

Figure 14 illustrates a prototype of a missile guidance system.

Groundbased Computer System

Figure 15 illustrates a groundbased general purpose computer system developed by the Raytheon Company, and which was mechanized entirely of SEM.

BENEFITS

As can be seen, the SEM Program is a thorough approach to instituting a rational discipline for the development and deployment of military electronic systems. By properly applying its principles and requirements, the SEM Program can benefit the following program phases.

Development Phase

Reduced development costs. Availability of standard modules results in a reduction in design effort, reduction of performance verification testing at the module level, and a reduction in the development of documentation required to define and support the modules.

Reduced development time. Lead time is reduced since complete functional modules and piece parts are available with adequate documentation.

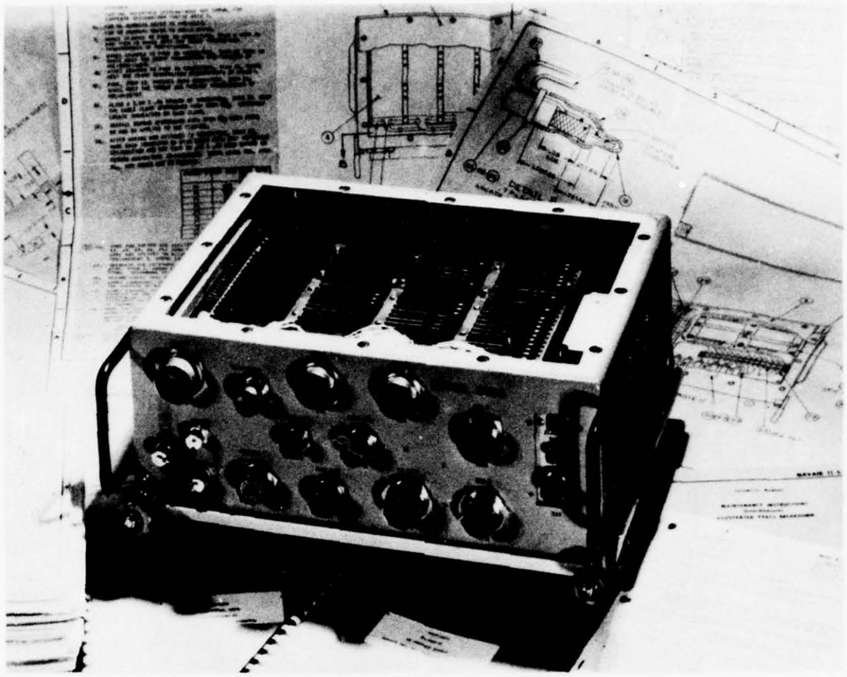

Figure 13. Airborne Weapon Control System

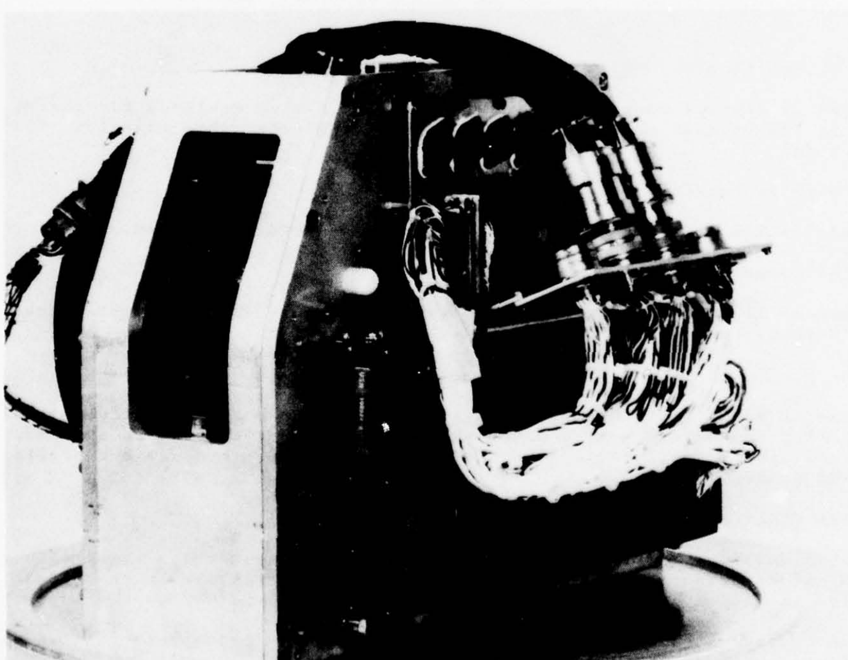

Figure 14. Missile Guidance System

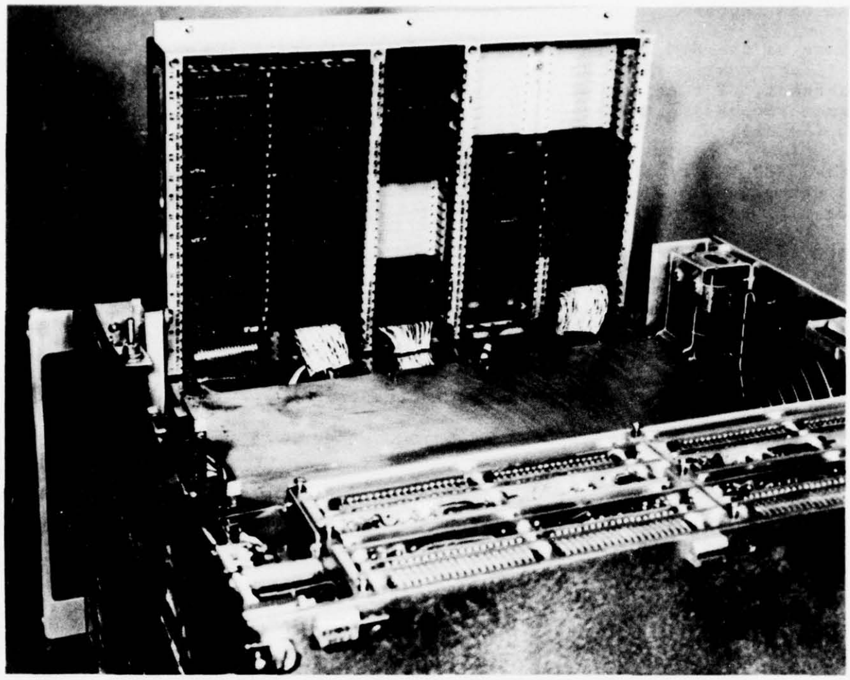

Figure 15. Groundbased Computer System

More realistic cost bids for system R&D. By specifying use of standard functional modules in R&D requests for proposals from contractors, the tendency for a contractor to "buy in" is reduced since he cannot be assured of winning the production follow-on by being locked in with proprietary or insufficiently documented hardware.

Production Phase

Reduced cost of test equipment. The module test equipment required to support the system can be used on other systems that use common modules. This reduces the need for unique test equipment for each separate system.

Reduced cost of production modules. This occurs through multiple sources of standard modules (competition), high volume production, and common quality and process procedures.

Reduced time to volume production. Standard modules are readily available from multiple sources which reduces or eliminates production tooling time and process development time.

Operations Phase

Reduced operations and maintenance costs. By reducing the number of different module types and increasing module commonality among systems, savings will occur by a training requirements reduction due to similarity of hardware, reduction in the variety of test equipment required, and simplified maintenance at all levels because of the SEM "discard upon failure" maintenance concept.

Reduced logistics costs. This is probably the greatest advantage of the SEM Program. Use of interchangeable modules between systems reduces the number of different spare part types required to be supported in the logistics system. In addition, the completeness of the functional specifications and availability of competitive sources eases spares procurement problems and lowers module cost.

Improved system reliability. The SEM Program has demonstrated that highly reliable hardware can be produced and still be cost competitive. In addition, since the same module types are used from system to system (do not become obsolete after one application), they can be continually improved through field use performance and failure analyses and appropriate corrective action.

FUTURE OF THE SEM PROGRAM

The future of the SEM Program is dedicated to extending its applicability to a greater number of military systems for, as its applications base broadens, the benefits derived will also increase. For this reason, efforts are being continually carried out to make the SEM Program responsive to the majority of users so that it can be applied to new development programs.

Currently, SEM R&D efforts are being directed toward extending the number of complex digital functions, analog and communications functions, and developing higher density hardware that will be more readily applicable to avionics.

RECOMMENDATION

We in the R&D community should look into the feasibility of applying not only the modules of the SEM Program, but also its basic philosophy, that being:

Standardize on a limited number of items,

Do not develop items for marginal increases in capability, and

Use functional specifications to encourage competition and avoid obsolescence.

If we can agree on a flexible and well-disciplined foundation through the use of SEM, it will not only aid us in our immediate objectives, but will carry through and positively influence all the phases of the system life cycle--yielding not only better equipment, but better equipment at lesser cost. By this approach, we can assure the continued availability of R&D resources by fulfilling the needs of the operational community--our ultimate customer.

GLOBAL POSITIONING SYSTEM TACTICAL MISSILE GUIDANCE

by
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5-1

SUMMARY

The concept of GPS tactical missile guidance is discussed from the standpoint of advantages gained by high level functional integration between the missile and a GPS-equipped launch aircraft. The conflicting requirements of high performance and low cost are shown to be attained by elimination of missile guidance functions that can be performed by the aircraft GPS system and transferred to the missile immediately prior to launch. The importance of integrating missile GPS receiver and inertial guidance system measurements for achieving maximum performance in a jamming environment is discussed, as well as the filter form employed and resulting performance. The unique operational advantages of this GPS missile guidance system for weapon delivery are described, including those gained by integration with a GPS-equipped aircraft.

I. INTRODUCTION

The NAVSTAR Global Positioning System (GPS) is a satellite navigation system designed to provide highly accurate three-dimensional position and velocity data to users anywhere on or near the earth. The GPS will consist of a constellation of satellites, a ground control system, and an unlimited number of users — aircraft, ships, vehicles, satellites and manportables. The system concept evolved from the United States Air Force and Navy studies initiated in the mid-60's. The resulting developments — USAF Program 621B and Navy Timation — were merged into NAVSTAR GPS in 1973.

All U.S. Armed Services are participating in NAVSTAR. USAF is the executive agency and manages the development from the NAVSTAR Joint Program Office of Air Force Space and Missile Systems Organization in El Segundo, California. Nine participating NATO nations have expressed a desire to participate in the GPS Phase II Program. Participation will consist of the establishment of a NATO group at the Joint Program Office in California for the purpose of contributing to the GPS Program and coordinating NATO requirements and operational applications.

Fundamental characteristics of GPS — all weather, worldwide availability, unsaturability, autonomy, high accuracy, and the jam-resistance inherent in spread spectrum modulation — make it an attractive candidate source of positioning data for guidance of tactical air-to-surface missiles. However, since missiles are expendable, it is essential that a missile GPS guidance system be relatively inexpensive. The requirement for low cost can be met most effectively through a system concept which combines a high degree of integration of the aircraft GPS system with an efficient integration of GPS and inertial sensors on-board the missile. The functional requirements for aircraft/missile and GPS/inertial integration are discussed below. A system concept which implements these integrations and which is currently under development by the U.S. Air Force Armament Laboratory at Eglin AFB, Florida, is also presented.

NAVSTAR GLOBAL POSITIONING SYSTEM

The NAVSTAR GPS consists of three major segments: space, control, and user. The concept of operation is as follows: each satellite continuously transmits pseudo-random noise (PRN), spread spectrum navigation signals. The signals carry information regarding the satellite ephemerides and clock behavior. GPS users calculate their position/velocity in navigation coordinates by multilaterating on the satellites. If the users were equipped with precise clocks they could synchronize on the PRN signals to make range measurements from three satellites. The user's position would be represented by the intersection of three spheres, each centered at one of the satellites. To avoid the requirement for a precise clock, the multilateration process is extended to a fourth satellite. The reception of four navigational signals permits three independent range difference equations to be formed. These equations may be solved to calculate the intersection of three hyperboloids of revolution which uniquely define the user's position.

In its operational configuration, the GPS satellite constellation will consist of 24 satellites in three circular, 20,200 km orbits with inclination of 63 degrees. This satellite deployment ensures that a minimum of six and an average of nine satellites are in view from any point on the earth, thus ensuring satellite coverage for three-dimensional navigation on a world-wide basis.

Each satellite transmits PRN signals on two carrier frequencies: 1575 MHz, termed L_1, and 1227 MHz, termed L_2. The signals are coherently generated and can be used to determine the magnitude of ionospheric signal propogation delay. Both navigation signals are modulated with a PRN sequence of primary digits, called the Precision (P) code, at a chipping rate of 10.23 mbps. The PRN sequence acts to spread the carrier bandwidth to 20 MHz. L_1 is further modulated with a 1.023 mbps Clear Acquisition (C/A) code, which aids users in gaining rapid signal acquisition. Other data, such as ephemeris, are provided at 50 bps.

A typical GPS user set consists of, an antenna, receiver, data processor and control display unit. The receiver generates replicas of the PRN codes and correlates them with received satellite signals to develop pseudo-range and pseudo delta-range measurements to each of four satellites. The processor then combines the measurements with computed satellite position and velocity to calculate the user's three-dimensional position and velocity in earth-centered coordinates.

The GPS is being developed within a three-phase program of concept validation, system validation, and production. The corresponding schedule of deployment is shown in Figure 1. The concept validation phase which began in December 1973 includes fabrication, test, and evaluation of developmental user sets, using both a simulation facility (the inverted range) at the Yuma Proving Grounds, Arizona, as well as a validation constellation of satellites. The constellation will be formed with six prototype satellites plus spares. The first of these prototypes, NAVSTAR I, was launched on February 22, 1978. Successful user equipment tests have already been conducted using signals from NAVSTAR I in conjunction with signals from ground satellite simulators at the inverted range. As additional satellites are placed in orbit during Phase I, testing of GPS user equipment will be extended to oceanic test ranges, and to Eglin AFB, Florida.

A constellation of six or more satellites will be available throughout Phase II, during which system effectiveness and supportability will be established. During this phase, the control system segment will evolve into its fully operational configuration, and a final determination of user equipment configuration will be accomplished. Initial operational test and evaluation (IOT&E) will be conducted during this phase. During Phase III, production satellites will be launched to complete the full 24 satellite constellation. NAVSTAR Initial Operational Capability (IOC) will occur in the mid-1980's.

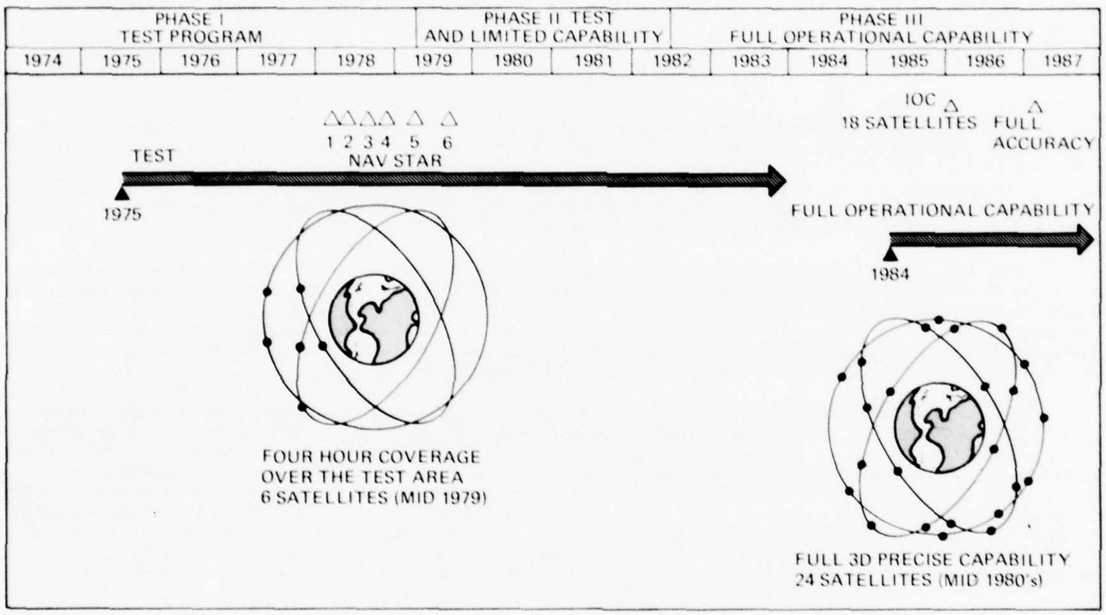

Figure 1. GPS Development Schedule

II. MISSILE GUIDANCE REQUIREMENTS

In considering the use of GPS for tactical missile guidance, it is important to consider the principal attributes the system should have. Consider the three design objectives identified in Figure 2. They are: 1) electronic countermeasures (ECM) resistance; 2) guidance unit cost, and 3) performance. This order is, in fact, the approved priority for the guidance systems under development within the Tactical GPS Guidance Program.

Traditionally, design priorities for missile guidance systems have been ordered as performance, unit cost, and ECM resistance. The reason for the interchange of performance and ECM resistance for the GPS guidance system is that the primary application is midcourse guidance, where the performance demands are considerably less, and are dictated principally by the acquisition "basket" of a terminal guidance sensor. Inherent GPS accuracies not only easily meet terminal acquisition requirements, but also meet most terminal guidance requirements for area munitions. On the other hand, with the increased emphasis on use of ECM in tactical warfare, it has become abundantly clear that considerable effort is required to provide a high probability of reaching the target area in the face of a determined defense.

Unit cost, although second priority, is also a very important factor in considering GPS for missile guidance, particularly for tactical missiles. A significant reduction in the cost/complexity of the current developmental aircraft-type GPS equipment of comparable performance is required to be attractive for the "throwaway" application of tactical missile midcourse guidance. The relative importance of the three design objectives of Figure 2 varies somewhat with the missile application. ECM resistance and performance are emphasized in the long-range cruise weapons such as Tomahawk; whereas, low cost is more important to the shorter-range, low-cost weapons such as GBU-15 and WAAM.

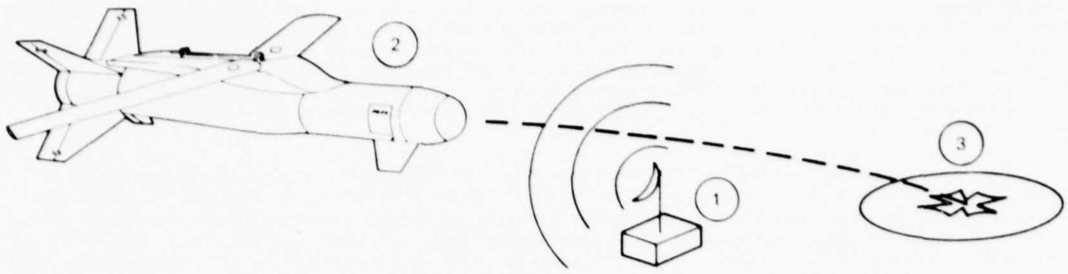

Figure 2. GPS Guidance System Design Objectives

INTEGRATED GPS/INERTIAL

The desired level of ECM resistance places several requirements on the guidance system functional design. To begin with, the missile receiver (M-Receiver) must be designed for maximum antijam (AJ) capability. This is achieved by tracking the pseudo-random Precision (P) code signals and using minimum acquisition and tracking bandwidths. However, velocity aiding must be supplied to the receiver to achieve these narrow bandwidths in the face of anticipated missile dynamics. Aiding is also essential to provide for extrapolation of range and range-rate to the NAVSTAR satellites, permitting reacquisition of GPS signals following periods of signal loss. Secondly, a pure inertial guidance capability is necessary, as this provides a means to continue to the target/acquisition basket if the GPS signals are not regained. This capability is essential for scenarios where powerful jammers are collocated with the target. The twin requirements of receiver velocity aiding and navigation without GPS measurements dictate an integrated GPS/inertial navigation guidance system. An integrated system is shown functionally in Figure 3.

In the closed guidance loop (lower portion of figure), the vehicle dynamics are sensed by an inertial measurement unit. The measurements are then used to perform inertial navigation, providing vehicle position and velocity estimates which can be compared to desired target coordinates. Differences become a basis for the generation of steering commands generated for an autopilot. In the absence of any other information, the missile will perform with accuracy determined primarily by the quality of the inertial measurement unit. The GPS Class M Receiver provides additional independent measurements of vehicle position and velocity which are compared in a Kalman filter with inertial derivations of like data. Differences in these quantities are then used to estimate both the dominant GPS

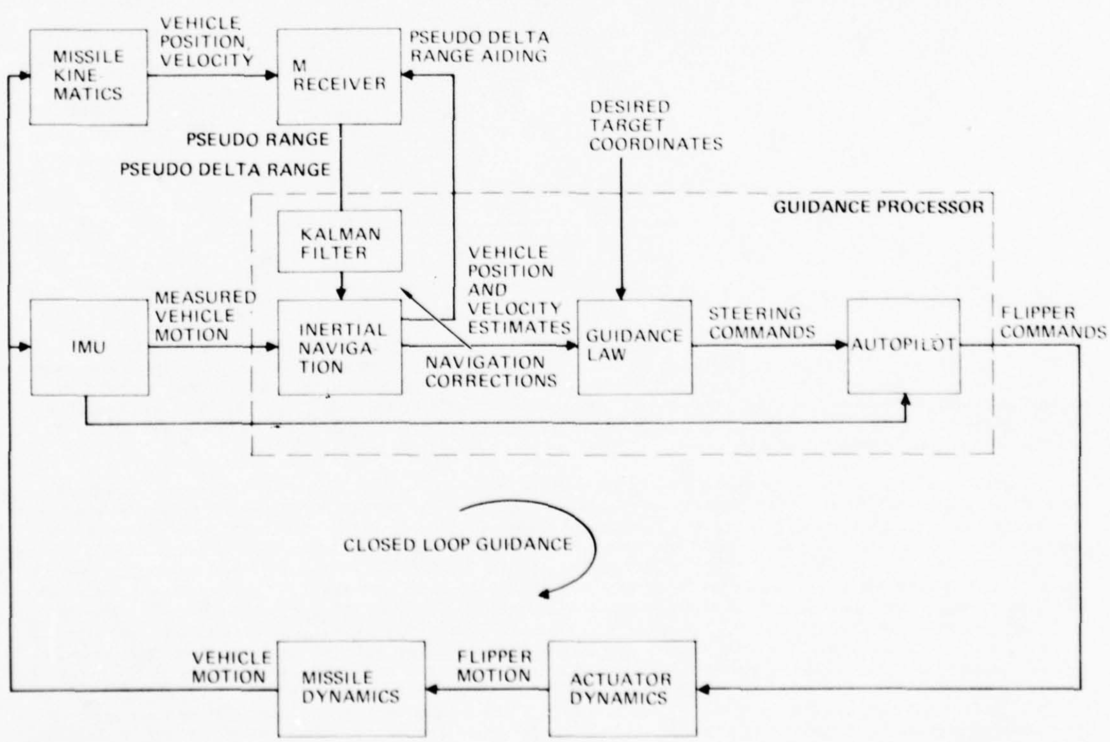

Figure 3. GPS Guidance System Functional Design

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and dominant inertial error contributors, allowing compensation in the navigation computations. The integration of GPS and inertial information via the Kalman filter is ideal from a systems view point. The GPS receiver provides highly accurate, long-term data which are used to bound the growth of inertial errors. In fact, these data can be used to calibrate and compensate for the principal errors of a low-quality (low-cost) inertial unit, yielding improved system navigation performance. On the other hand, the inertial sensors provide velocity aiding to the GPS receiver, improving its performance in a high ECM environment.

The integrated GPS/inertial concept results in two distinct advantages for a missile guidance system. The inertial aiding of the GPS receiver coupled with the bounding of inertial errors with GPS measurements allows formation of an optimal guidance system from the combination of suboptimal (low-cost) subsystems. And most importantly, this integrated guidance automatically reverts to simple inertial guidance whenever GPS signals are lost for any reason. In the presence of severe jamming, the missile continues to guide to the target or terminal acquisition point, suffering only a graceful degradation of accuracy rather than a catastrophic loss of guidance.

III. GPS AIRCRAFT NAVIGATION/MISSILE GUIDANCE INTEGRATION

A launch aircraft with an integrated GPS/inertial navigation system as a part of its avionics suite offers unique advantages to the aircraft/missile weapon system. This combination provides both increased operational flexibility and reduced cost and complexity of the missile GPS guidance system. Figure 4 illustrates the desired aircraft/missile guidance system interface.

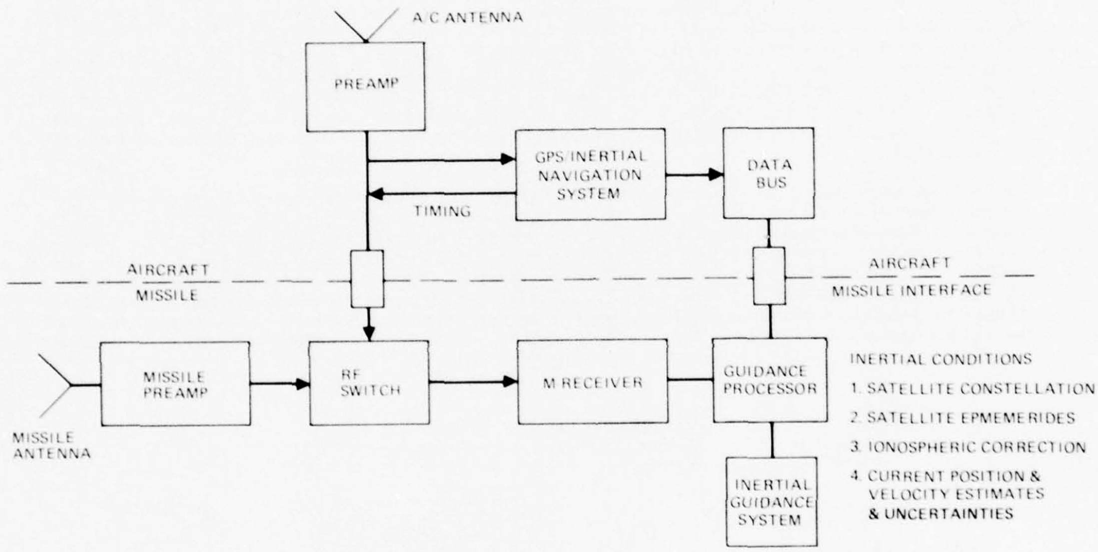

Figure 4. Desired Aircraft/Missile Interface

AIRCRAFT/MISSILE INTERFACE

Two specific interfaces are required: 1) an RF link to carry GPS signals from the aircraft antennas to the missile (superimposed on the RF link are discrete time synchronization signals to synchronize aircraft and missile GPS receivers, facilitating rapid signal acquisition), and 2) a path for digital initialization data from the aircraft GPS equipment to the missile guidance system via the aircraft digital data bus. The significance of these interfaces to the simplification of the missile GPS receiver is more fully described below.

Providing RF signals from the aircraft antennas yields two significant advantages. The first advantage is that verification that the missile guidance system is functioning properly can be made prior to weapon launch. The second is the availability of higher signal-to-jammer reception of the satellite signals by the aircraft antenna system than can be provided by the necessarily smaller and simpler missile antenna system. This is an important advantage since the missile receiver requires higher signal-to-noise conditions for initial acquisition of the signal than for lock and track. Having once acquired the satellite signals and established accurate time synchronization, the missile receiver has the capability to reacquire the satellite transmissions after launch under lower signal-to-jam conditions. Additionally, tracking before launch provides GPS measurements to the on-board Kalman filter to allow alignment of the strapdown inertial guidance system and estimation of the phase and frequency offsets of the missile receiver clock. When the missile is launched, the guidance system loses the NAVSTAR signals for several seconds until the missile is clear of the aircraft. The guidance system then initiates a reacquisition of the signals. Estimation and compensation for the receiver clock offsets reduces the uncertainty in reacquisition timing. This allows reacquisition with a narrower bandwidth and consequently lower required signal-to-noise.

MISSILE RECEIVER SIMPLIFICATIONS

The principle goal in the design of the GPS missile guidance system is to achieve adequate performance in an ECM environment with a minimum cost of equipment in the expendable missile. This goal has been realized by Hughes in the system design developed during the Tactical GPS Guidance Development Program for the U. S. Air Force Armament Laboratory, Eglin AFB. In that design, discussed below, a high level of functional integration is achieved between the aircraft GPS/inertial navigation system and the missile GPS guidance system. Specifically, the design concept eliminates functions from the missile guidance system that can be performed by the launch aircraft and supplied to the missile immediately prior to launch.

A typical GPS receiver performs algorithms which enable it to: select the optimum four satellites to track at a given location and time; search, acquire, and track the Clear Acquisition (C/A) code; transition to search, acquire and track the Precision (P) code; demodulate the 50 bps navigation messages to obtain ephemeris and other navigation data; and compare L_1/L_2 time of arrival to compute ionospheric propogation delay. To accomplish these tasks, the typical receiver design must employ: memory of almanac information on the entire system of 24 satellites; complex hardware and software to search, acquire, and track both C/A and P codes; phase tracking of the carrier to permit demodulation of the 50 bps data; and dual RF hardware to allow simultaneous L_1 and L_2 receiver operation. Figure 4 indicates the initial condition data that can be provided by the aircraft GPS equipment to facilitate the direct acquisition and track of the satellite P code signals by much simplified missile equipment. The following paragraphs describe how this simplification is achieved.

It is presumed that the launch aircraft has acquired and is tracking the desired GPS satellite constellation signals prior to launch. By passing the identity of the desired set of satellites and the parameters required to describe the ephemerides to the missile guidance system, the algorithm for selection of the optimum combination of satellites and the almanac describing the ephemerides of all 24 satellites are avoided in the missile.

For the shorter range missiles, such as GBU-15, the ephemeris parameters passed from the aircraft are adequate throughout flight, eliminating the need for a carrier tracking loop capable of extracting the 50 Hz navigation message from the satellites' signals. For longer flight times, such as for the cruise missile, it may be necessary to change a satellite inthe constellation. Since the satellite ephemerides are predictable, the launch aircraft can anticipate the problem before launch and assign the desired new satellite and the time to make the transition. The longer flight times of the cruise missiles may also require 50 bps data demodulation during flight.

M-Receiver acquisition of the satellite signals is greatly simplified by the aircraft receiver supplying a precision timing mark on the RF link. This allows direct acquisition of the P-code signal, eliminating an M-Receiver requirement to carry a C/A code generator, which would be needed to acquire the 1.023 MHz C/A code.

Significant M Receiver cost savings are further achieved through the deletion of the two-frequency operation, which is used to determine the ionospheric propogation delay by time difference measurement. This propogation delay varies slowly with time, with a worst-case time gradient of 20 ns (feet) per hour. The apparent average gradient is at least an order of magnitude less; thus, these data on the delay can be provided to the M Receiver immediately prior to launch, eliminating the requirement for the receiver to make this computation. Initialization of ionospheric propogation delay also enables the missile receiver to function while tracking only one GPS frequency instead of two, since the second frequency exists to aid the delay computation process.

A constant ionospheric delay correction factor is adequate for missile flight times less than 30 minutes. For longer flights, the correction factor for each satellite can be modeled by a polynomial expression whose coefficients are supplied by the launch aircraft.

Figure 5 shows the corresponding hardware simplications that can be made in implementing an M-Receiver by taking advantage of the aircraft GPS equipment capability. Another significant savings in M-Receiver hardware shown is the change to a sequential receiver design and elimination of three of the four carrier tracking channels. This simplication is not a result of the aircraft interface, but rather of integration of the GPS receiver with a missile inertial guidance system (IGS). In sequencing through the four satellites, any given satellite is only tracked for one-fourth of the time; e.g., 5 seconds if the total period is 20 seconds. The satellite must, therefore, be reacquired each sequencing period. Thus, the missile's inertial navigation capability must provide dead reckoning of position and velocity sufficiently accurately to position the receiver back onthe correct code chip, avoiding reacquisition. This accuracy is strongly dependent on the initial alignment of the IGS, and subsequent periodic updating using GPS measurements.

MISSILE INERTIAL GUIDANCE SYSTEM (IGS) ALIGNMENT

In-flight alignment of the missile's IGS can be accomplished in one of two ways, both of which rely on information from the launch aircraft. Traditionally, a missile inertial navigation system is aligned by a transfer of alignment from the launch aircraft inertial navigation system. This typically is accomplished by periodic transfer of aircraft velocity component estimates to the missile via the data bus, where a Kalman filter compares these estimates, which it assumes to be very accurate, to its own velocity estimates and uses the differences to estimate the misalignment quantities.

However, since highly precise position and velocity information is available, alignment of the IGS can also be accomplished directly from the GPS receiver measurements in the missile, providing a significant performance advantage. This alignment would be performed in the identical filter that processes GPS pseudorange and pseudo-range rate measurements during flight to provide navigation updates and allows estimation of both the dominant inertial component errors (e.g., gyro bias) and dominant GPS errors (e.g., clock phase and frequency). The accuracies of these two alignment approaches are comparable.

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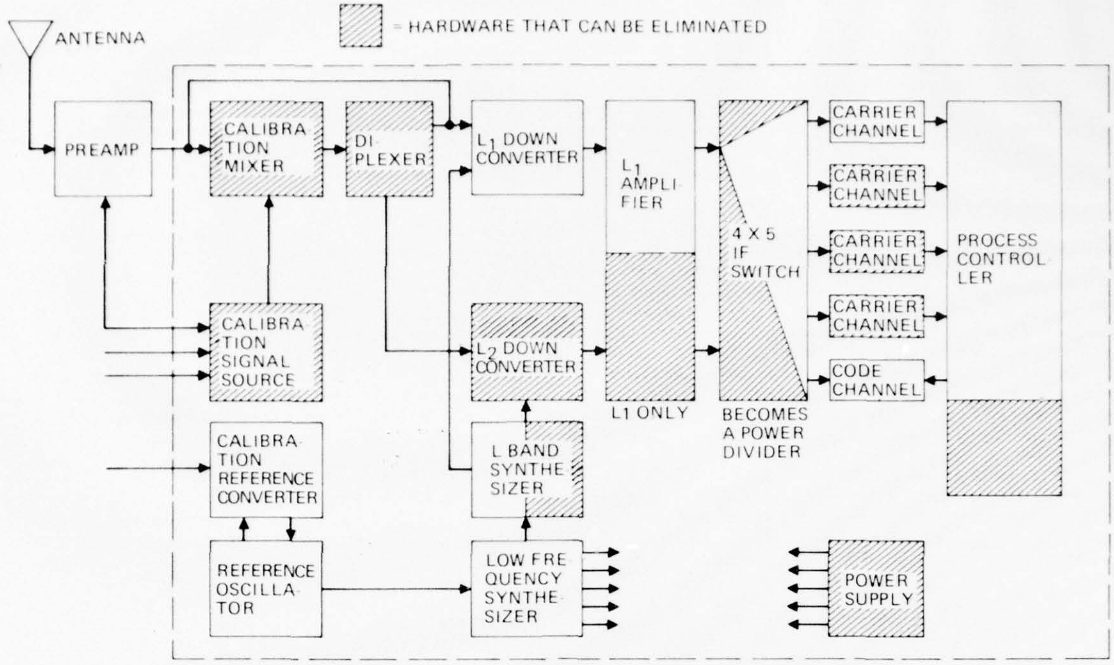

Figure 5. M Receiver Hardware Simplifications

An Upper-Diagonal (U-D) form of Kalman filter is used to generate the integrated GPS/inertial navigation solution. This filter form is the most computationally efficient of the Kalman filter forms that are designed to ensure a stable estimation solution. The M Receiver range measurements are used sequentially by the filter, one every 6 seconds, to estimate the dominant inertial and GPS errors. The 14 states used in the Hughes Tactical GPS Guidance System filter are shown in Table 1. This filter provides the mechanism for aligning the inertial sensor package to the navigation coordinate system prior to launch. Alignment is achieved via an "S" turn maneuver in the horizontal plane to produce horizontal changes in velocity, allowing observation of the azimuth misalignment. The maneuver and corresponding velocity history are shown in Figure 6. The transient response characteristic of the filter under these conditions can be seen in Figure 7. Several simulation runs were made with different interferring noise histories. The shaded area indicates the resultant dispersion in performance.

TABLE 1. DEFINITION OF FILTER STATE VARIABLES

Filter State	System State
X_1	North Velocity Error, m/sec
X_2	West Velocity Error, m/sec
X_3	Up Velocity Error, m/sec
X_4	Latitude Error, meters
X_5	Longitude Error, meters
X_6	Altitude Error, meters
X_7	Misalignment About North, degrees
X_8	Misalignment About West, degrees
X_9	Misalignment About Up, degrees
X_{10}	Clock Phase Error, meters
X_{11}	Clock Frequency Error, m/sec
X_{12}	Gyro Drift About Forward, deg/hr
X_{13}	Gyro Drift About Left, deg/hr
X_{14}	Gyro Drift About Z, deg/hr

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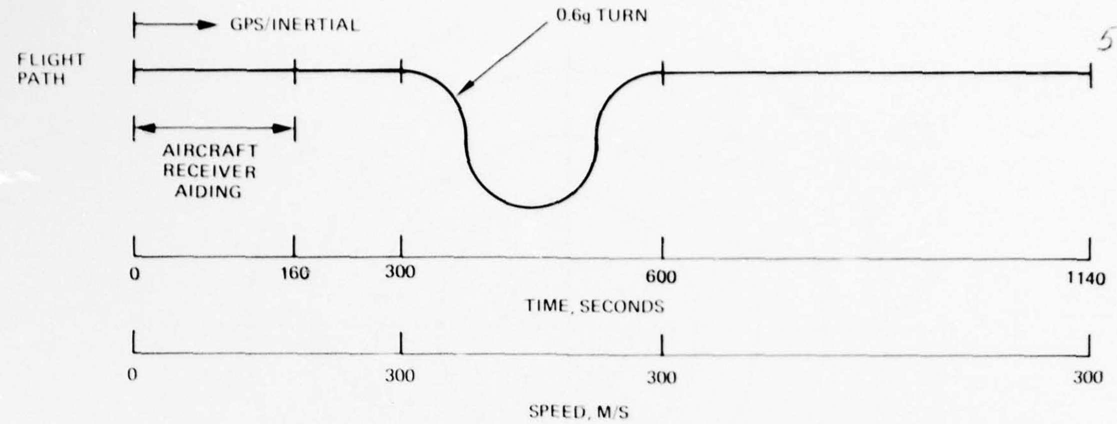

Figure 6. Flight Conditions for System Alignment

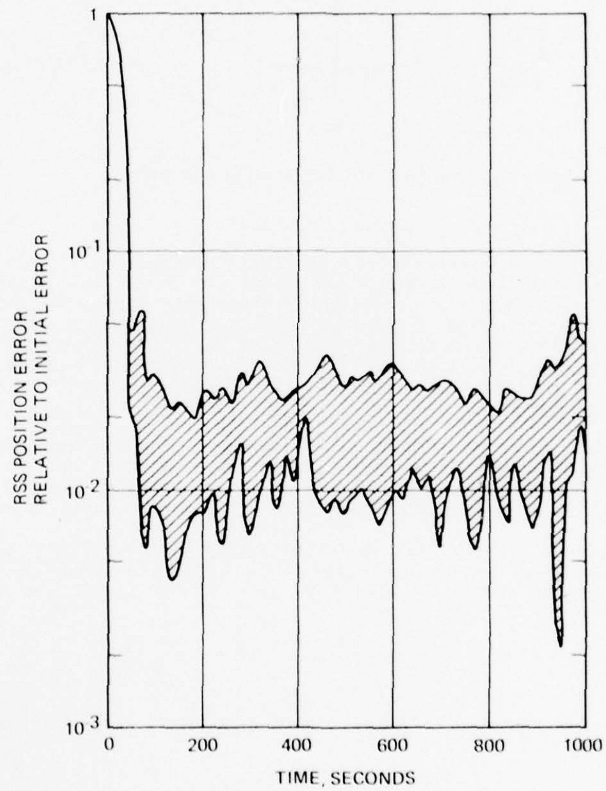

Figure 7. RSS Position Error

NAVIGATION PERFORMANCE

The filter also plays a key role in the ECM resistance of the weapon's guidance system. When four satellites are being tracked, the frequency and accuracy of the GPS measurements provide an accurate navigation solution even with significant receiver and inertial sensor errors. However, in a hostile jamming environment, it is likely that the signals from one or more satellites will be lost during portions of the missile's trajectory, particularly in the vicinity of a heavily-defended target. When satellite signals are lost, navigation accuracy can degrade rapidly if the major errors are not limited. The filter provides the mechanism for estimating and compensating for these errors.

The navigation accuracy of the guidance system has been investigated via computer simulation studies. The particular satellite constellation assumed is shown in Figure 8. The navigation performance of the system has been determined for various numbers of satellites from four to none.

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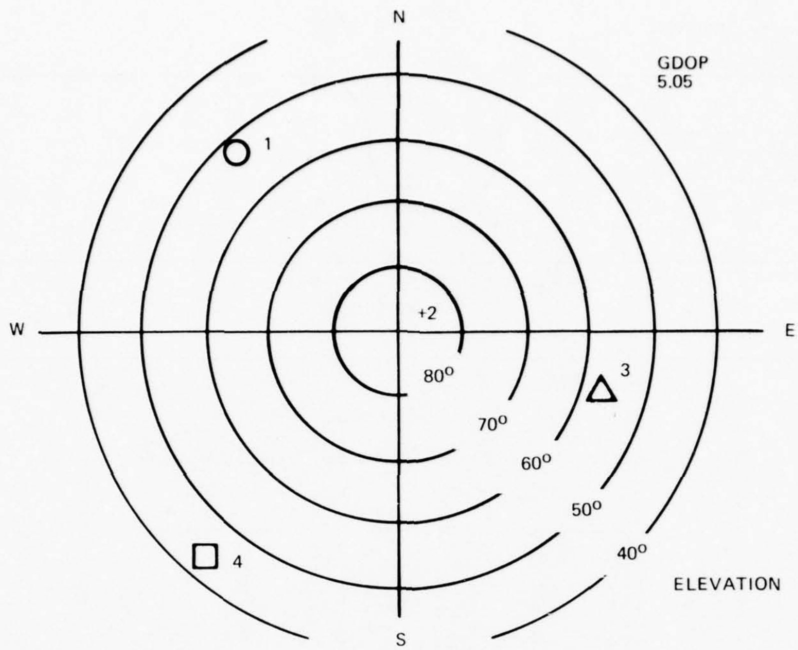

Figure 8. Satellite Local Elevation and Azimuth

The principal measure of guidance system performance is its accuracy in determining the missile's position in navigation coordinates. Studies have shown that achieving adequate navigation accuracy is assured by simply maintaining track of the satellite signals. If the receiver is able to track the satellite signals, regardless of the signal-to-noise environment, the navigation accuracy is acceptable for midcourse guidance; therefore, the problem is essentially one of maintaining signal tracking in a jamming environment. This capability is closely related to the accuracy of velocity aiding provided to the receiver. For this reason, performance is measured in rss velocity error and is presented here relative to nominal performance.

Figure 9 illustrates the effect on performance of losing the overhead satellite for a period of 60 seconds. Surprisingly small errors are developed under this condition. This is primarily a result of the ability to estimate the dominant clock errors of phase and frequency offset. This performance shown is comparable to that resulting from losing any one of the four satellites for an equivalent period.

The loss of two satellites, as could be anticipated, is much more detrimental to performance. The effect of losing satellites 1 and 2 for a 60-second period is shown in Figure 10. This performance is representative of that experienced with the loss of any pair of the four satellites being tracked.

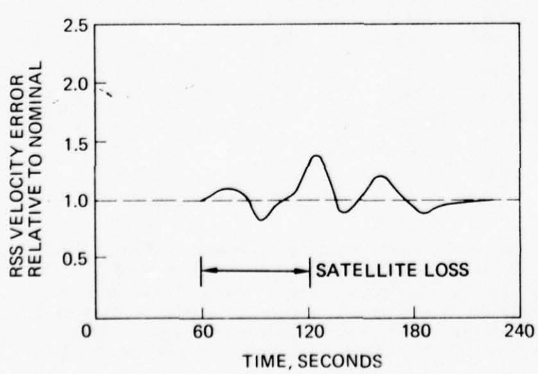

Figure 9. Effect of Losing One Satellite on Velocity Estimation Error

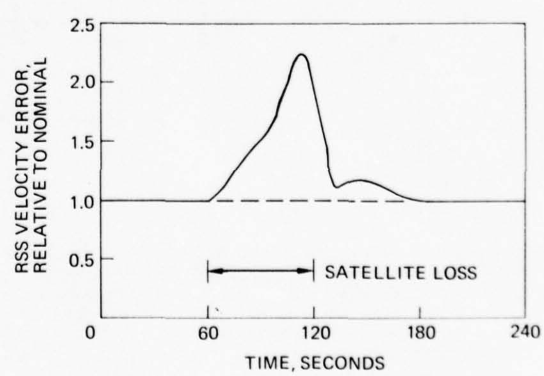

Figure 10. Effect of Losing Two Satellites on Velocity Estimation Error

IV. TACTICAL WEAPON DELIVERY

In the integrated configuration described above, GPS is an especially attractive guidance aid for tactical weapons. In addition to great accuracy, this radio navigation system offers three attributes highly desired by military commanders: flexibility of employment, operational simplicity, and stealth.

NAVIGATIONAL ACCURACY

The navigational accuracies of most GPS receivers are expected to average 8 to 10 meter spherical error worldwide, 24 hours a day. Table 2 depicts the GPS error budget which leads to this figure. For sequential tracking receivers, such as the M-Receivers currently being developed within the USAF Tactical GPS Guidance Demonstration Program, the ranging error may be somewhat larger. In addition, for long flight time, GDOP could degrade before the target is reached unless the receiver is designed and programmed to change satellites during flight (or a constellation is chosen such that GDOP is unconstrained at launch, but is optimum at the target). Atmospheric delay error may also grow, depending on missile time of flight and type of propagation delay compensation.

TABLE 2. GPS NAVIGATION ERRORS
(Meters, 1σ)

Error Source	GPS Receiver
Satellite Ephemeris	1.5
Uncorrected Atmospheric Delay	2.5
Multipath	1.8
Satellites Unmodeled Clock Errors and Signal Delays	0.9
Receiver Ranging Measurement Deviation	1.5
RSS	3.8
GDOP*	2.2 to 3
One Sigma Spherical Error, meters	8 to 11

*Geometric Dilution of Precision - a factor which describes the degradation of accuracy in three dimensions due to the nonoptimum geometric configuration of satellites.

FLEXIBILITY OF EMPLOYMENT

Weapons using GPS guidance will provide great flexibility of employment in several ways. Since the GPS is a radio navigation system, these weapons will be capable of all-weather, day-night operation. The NAVSTAR satellite constellation is designed to provide uniform signal coverage over the entire earth. GPS guidance will thus be usable worldwide. Weapon trajectories will be unconstrained by terrain (mountainous, flat, or ocean), or by distance from friendly forces. Operational flexibility is further enhanced by rapid response to targeting. Weapons with GPS guidance will be able to accept targeting commands before launch, via program tape; in-flight before launch via pilot or weapon system operator inputs over the aircraft digital bus; or even postlaunch via secure data link, such as the Joint Tactical Information Distribution System (JTIDS).

OPERATIONAL SIMPLICITY

Under the dynamic scenario of conventional combat, operational simplicity is a feature to be desired of any weapon system. Since missiles with GPS guidance and prelaunch targeting are autonomous (with the exception of the satellite down-link), they are imminently suited for standoff, launch-and-leave tactics. The NAVSTAR system can serve an unlimited number of users simultaneously, eliminating the need for prestrike coordination. Finally, the built-in ECM resistance of a tactical GPS guidance system aids operational simplicity by reducing or eliminating requirements for supporting ECCM. The relative magnitude of this ECM resistance can be seen by examining the jamming margin of a GPS guidance system. The total jamming margin results from, signal gain at the satellite transmitter output, gain of both the satellite and receiver antennas, and the processing gain of the receiver.

Receiver processing gain results from the fact that the GPS employs spread spectrum communications. The principle of operation is shown in Figure 11. The PRN modulation causes a spreading of the baseband signal in the transmitted frequency domain. The receiver then compresses this wide spectrum signal back to its original bandwidth, by correlating the received signal with a replica of the PRN code. At the same time this correlation acts to spread any interfering source, such as J, by the same amount that the spread signal was compressed. Narrowband filtering is then used to reject that interference energy lying outside the compressed bandwidth. The consequent improvement in signal-to-noise ratio results in receiver processing gain.

The substantial effectiveness of this processing gain is illustrated by calculating the transmitted power that would be required of a conventional communication link, to achieve the same threshold performance as the GPS spread spectrum link in a given jamming environment. Consider a weapon approaching a target and a 110 mile nonspread spectrum communications link with this weapon. The

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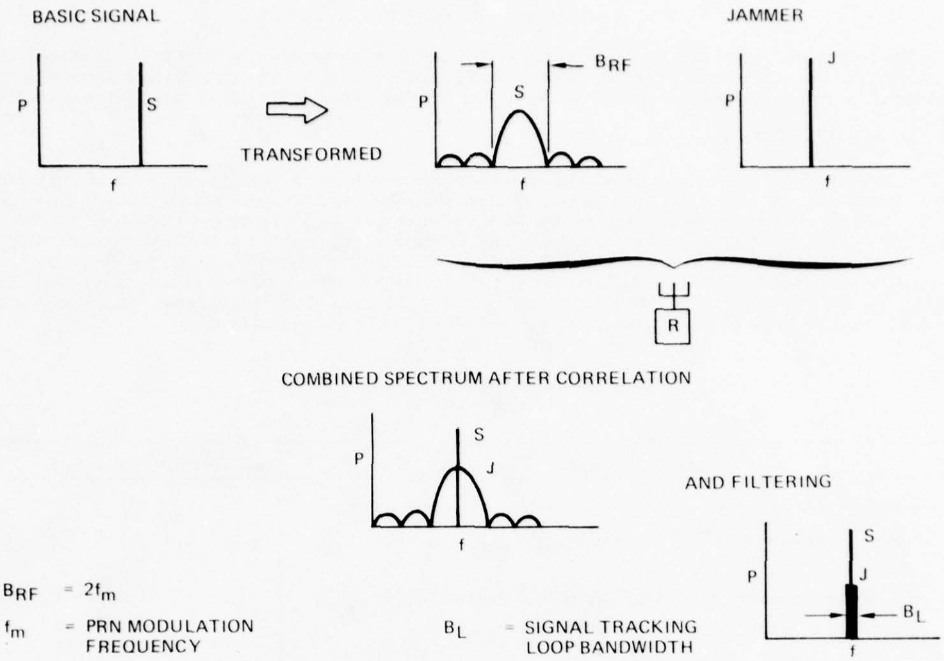

Figure 11. Pseudo Random Noise - Speed Spectrum

postsignal processing signal-to-noise ratio required by this receiver is identical to that required by a GPS receiver for equivalent detection and tracking performance. For either receiver

$$\frac{S}{N} = \frac{CP}{(KTB_L + J)\, d^2}$$

$$\approx \frac{CP}{J d^2}$$

for $J \gg KTB_L$.

where

d = transmitter-to-receiver distance

C = proportionality constant

P = transmitted ERP

KTB_L = receiver thermal noise

J = jammer power

and where antenna gains are neglected.

Equating postprocessing signal-to-noise ratio for the two receivers and solving for communications link transmitted power gives

$$P_{CL} = \left(\frac{d_{CL}}{d_{GPS}}\right)^2 \times \left(\frac{J_{CL}}{J_{GPS}}\right) \times P_{GPS}$$

where the subscripts CL and GPS denote conventional (nonspread spectrum) and GPS links, respectively. The design goal for processing gain of the demonstration GPS Class M-receiver is 60 dB, while the processing gain of a conventional receiver is 0 dB. Assuming identical CW jammer power impinging on both systems; the jamming power is effectively reduced by the magnitude of receiver processing gain, yielding:

$$P_{CL} = \left(\frac{110}{11,000}\right)^2 \times \left(\frac{1}{10^{-6}}\right) \times P_{GPS}$$

$$= 100\ P_{GPS}$$

Hence, to achieve an equivalent degree of jamming resistance, a conventional nonspread spectrum weapon communication link having 110 mile range would require 100 times greater effective radiated power (ERP) than the NAVSTAR satellite ERP!

STEALTH

The final attribute of GPS guidance is stealth. A weapon equipped with GPS guidance is entirely passive; it does not broadcast its presence to opposing forces. It also does not have any inherent altitude limitations; it is free to take maximum advantage of low altitude contour tactics and terrain masking. The implications are higher survivability, increased effectiveness, and, lower cost per strike.

UTILITY FOR TACTICAL WEAPONS

How do these operational characteristics impact the candidate tactical weapons? Certainly, a first consideration is that for GPS guidance, accuracy is independent of range to target. This guidance attribute is more important for longer range weapons than short. Because of the high flexibility of employment, GPS guidance is ideally suited for weapons responsive to dynamic scenarios. Stealth and ECM considerations favor low altitude weapons. Given these considerations, weapons that could make maximum use of GPS capabilities would include medium range and over-the-horizon antiship missiles, and air or sea-launched cruise missiles.

An antiship missile is currently limited in range by either midcourse guidance accuracy or target position uncertainty at the end of flight. The total missile-to-target position uncertainty at the end of the midcourse phase must be sufficiently small to allow acquisition with the terminal sensor. The ability to predict a target's position is currently several times better than the midcourse guidance accuracy of antiship missiles using inertial navigation. This guidance inaccuracy necessitates a pop-up maneuver at considerable distance from the target, to search and acquire the target and steer out errors. GPS guidance accuracy would allow closing to much shorter ranges, greatly increasing both weapon effectiveness and survivability.

GPS guidance would be nearly as beneficial for air-launched conventional standoff weapons, wide-area, antiarmor weapons and ground-launched cruise missiles; but for these weapons, ECM resistance appears to be the most important feature. Subsonic glide and boost-glide weapons would also make use of the most prominent GPS characteristics, but guidance system cost would receive a stronger emphasis.

V. SUMMARY

A high level of functional integration between the delivery aircraft avionics and the missile has been shown to be particularly important to the design of a GPS tactical missile guidance system. The system design described achieves ECM resistance and positioning accuracy which approaches those of a GPS aircraft navigation system, but with simplified missile equipment having far less cost. These results have been achieved by eliminating the performance, in the missile equipment, of functions which are performed by aircraft equipment and can be supplied prior to launch.

A key element of the system implementation is a UD filter which mixes the GPS measurements of ranges to the satellites with inertial measurements of vehicle motion, producing estimates of the dominant GPS and inertial errors and reduction of their effects. The ability to estimate receiver clock errors is shown to greatly reduce the sensitivity to loss of one of the four satellites. The effect of losing two satellites for a period of 60 seconds is much more severe, causing a doubling of the errors in velocity estimation for receiver aiding, which is critical to maintaining signal tracking in an ECM environment.

The advantages in the use of GPS for midcourse guidance of tactical long range air-to-surface missiles in combination with a GPS-equipped launch aircraft are clear. These advantages provide a weapon delivery capability unmatched in both performance and operational flexibility. With inherent high position accuracy, GPS guided missiles can be delivered worldwide, in any weather, over any terrain. Using GPS, missiles can fly at extremely low altitudes with a completely passive guidance system, yielding a maximum of stealth and consequent dividends in both weapon effectiveness and survivability. The antiship role provides a graphic illustration of the stealth advantages, coupled with the potential for operation at ranges limited only by targeting uncertainties.

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DIGITAL FLIGHT CONTROL SYSTEM ARCHITECTURE AND IMPLEMENTATION

7-1

by
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SUMMARY

Digital flight control systems must be designed to match their specific functional requirements if they are to satisfy the integrity, performance and cost targets. The architecture must be simple and modular so that it can be readily analysed. There must be a one to one correspondence between software and hardware to ensure visibility, particularly for the Failure Mode Effect analysis. Modularity enables flexibility to be maintained during development and permits development costs to be shared between projects. A modular system is described and its associated communication and control system, which uses a standardised interface, is outlined. The advent of the digital microprocessor has extended the range of viable architectures and has made multiprocessor configurations (particularly dual processor) attractive. In such configurations, 16 bit microprocessors perform auxiliary functions such as data management and/or self test. The degree of self test can extend from preflight testing to full monitoring, in which the microprocessor undertakes a dissimilar check of the main processor, thereby protecting against common mode failures which can occur in a multiple, similar redundant system.

1. INTRODUCTION

A modern automatic flight control system can vary in sophistication from a simple, single function device (eg a roll or yaw damper) to a multi-functional system embracing 3/4 axis autostabilisation, autopilot functions, thrust management, flight envelope protection, and flight management (navigation and performance computation). The development of sophisticated automatic flight control systems has been accelerated by the advent of:-

a) Control Configured Vehicle design in which the inherent aerodynamic stability of combat aircraft is reduced to improve manoeuverability and structural efficiency.

b) Active control of commercial aircraft which embraces manoeuvre load control (the redistribution of aerodynamic loads to reduce structural loads), elastic mode suppression and gust alleviation.

c) Three axis, full authority fly-by-wire primary control systems.

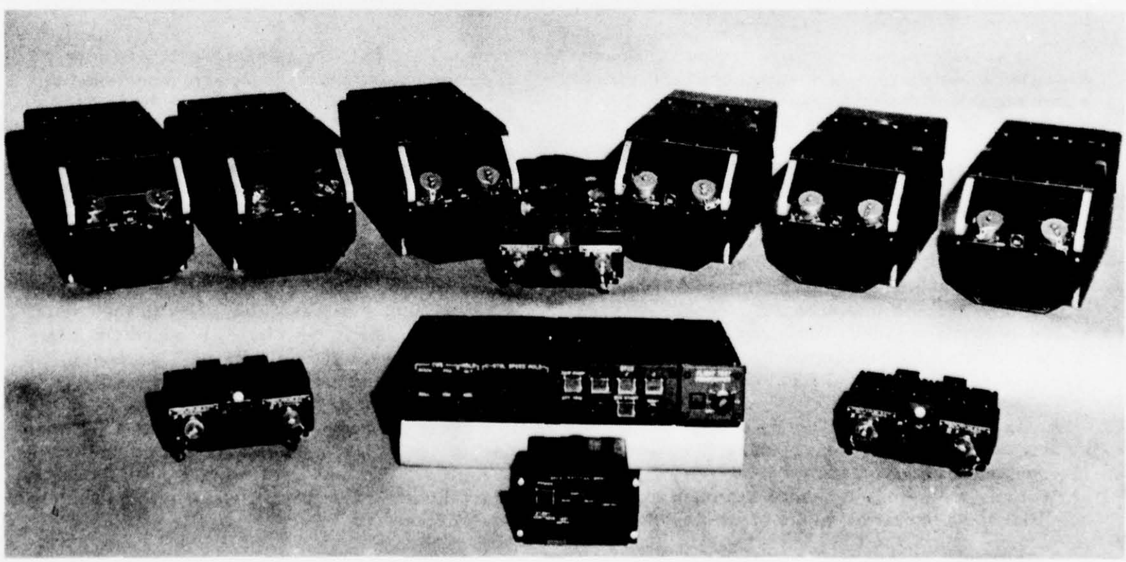

Figure 1 Digital AFCS Equipment

Operational survival of a single failure can be ensured by the use of either a triplex or a duplicate monitored arrangement, and higher integrity, double failure operational systems can be achieved by means of either a quadruplex or a triplicate monitored system. The Boeing YC-14, for example, employs triplex redundancy for fail operational systems, and reduced levels of redundancy for less critical functions. A current full authority fly-by-wire application employs double triplex actuators for the primary control surfaces, four lanes being driven by unconsolidated quadruplex channel outputs and the remaining two lanes by the consolidated outputs from the quadruplex channels.

As the level of sophistication of these systems grows, it is increasingly necessary to adopt a 'total system' approach, whereby design is optimised to strike the best compromise between various conflicting design factors such as integrity, performance, crew workload, availability, and life cycle costs. Moreover, the individual factors are themselves giving rise to more stringent design criteria.

The probability of failure of a high authority, fail-operational system must be extremely low, and this calls for a high integrity design incorporating both redundancy and self monitoring techniques to obviate the catastrophic effects of control surface hardovers under single or multiple failure conditions. Digital processors do not degrade gracefully and the consequences of a hardware failure or software error may be complicated and variable, dependent on the nature of the computation being performed. Such failures can result in an abrupt and extensive system malfunction unless suitable failure detection and correction techniques are incorporated. Special safety assessment procedures have been developed which take account of the failure characteristics peculiar to digital systems. High integrity hardware and software architectures are designed to provide visibility of operation while meeting performance and cost specifications.

This paper describes how the advances in technology are leading to systems which, through the use of standardised interfaces and modular design, are flexible and provide adequate computing power and integrity whilst meeting life cycle cost targets.

2. MINIMISATION OF LIFE CYCLE COSTS

Procurement agencies are cost conscious, and minimisation of life cycle costs, involving an optimum trade off between acquisition and operating costs, is a key design factor. The ability to adopt a low risk, 'task orientation' approach whereby multisource MSI and LSI devices are suitably configured to meet a particular customer specification with the minimum of special-to-type hardware, while taking advantage of continuing hardware development programmes is a key factor in reducing life cycle costs. This approach permits the use of standardised modules and test procedures for most major functional elements without detriment to the design flexibility, thus enabling development overheads to be minimised for each application.

Operating costs are minimised by devoting effort during the design phase to improving reliability and ease of maintenance. Software reliability is achieved by adherence to a hierarchically structured programming technique to ensure design visibility, together with the use of a disciplined modular structure with minimum inter-modular communication and interaction. Software/hardware integration testing requires the use of independently verified hardware, and includes closed loop testing with simulated aerodynamics for all operating modes throughout the flight envelope.

A variety of measures are adopted to ensure hardware reliability. Component quality is carefully specified to optimise procurement cost and reliability, environmental burn-in tests are undertaken to eliminate infant mortality defects, and derating criteria are applied to prolong component life. The reliability of semiconductor devices and high density LSI packages is also critically dependent upon adequate cooling, and conducting cooling together with cold wall heat exchanger techniques are used to eliminate potential hot spots and to reduce reliance upon the availability of cooling air.

Equipment is specifically designed for maintainability with regard to modularity, accuracy of fault location, and rapidity of testing and fault diagnosis using ATE. In particular, computer data highways interface directly with ATE, and data associated with in-flight defects can be stored on non-volatile memory for post-flight examination at the various levels of servicing. The high diagnostic confidence factor, and hence a low wrongful removal rate,together with the high speed and simplicity in the use of BITE, minimises labour costs at first line. Second and third line cost are similarly reduced and thus is aided further by development of low cost ATE which matches the specific application. Moreover, the effectiveness of BITE and ATE yields a significant economy in spares inventory investment.

As digital systems employ a large percentage of time shared hardware which performs a variety of system tasks, special test software is required to exercise all the hardware functions in a manner more rigorous and yet independent of any given flight software program. This test software is used during environmental testing to ensure thorough exercising of the equipment over the full range of environmental conditions and forms the basis of the post maintenance safety tests.

3. TRENDS IN DIGITAL TECHNOLOGY

In recent years LSI microprocessor technology has rapidly progressed to a level which matches the avionic requirements. Current microprogrammable computers enable the designer to select the

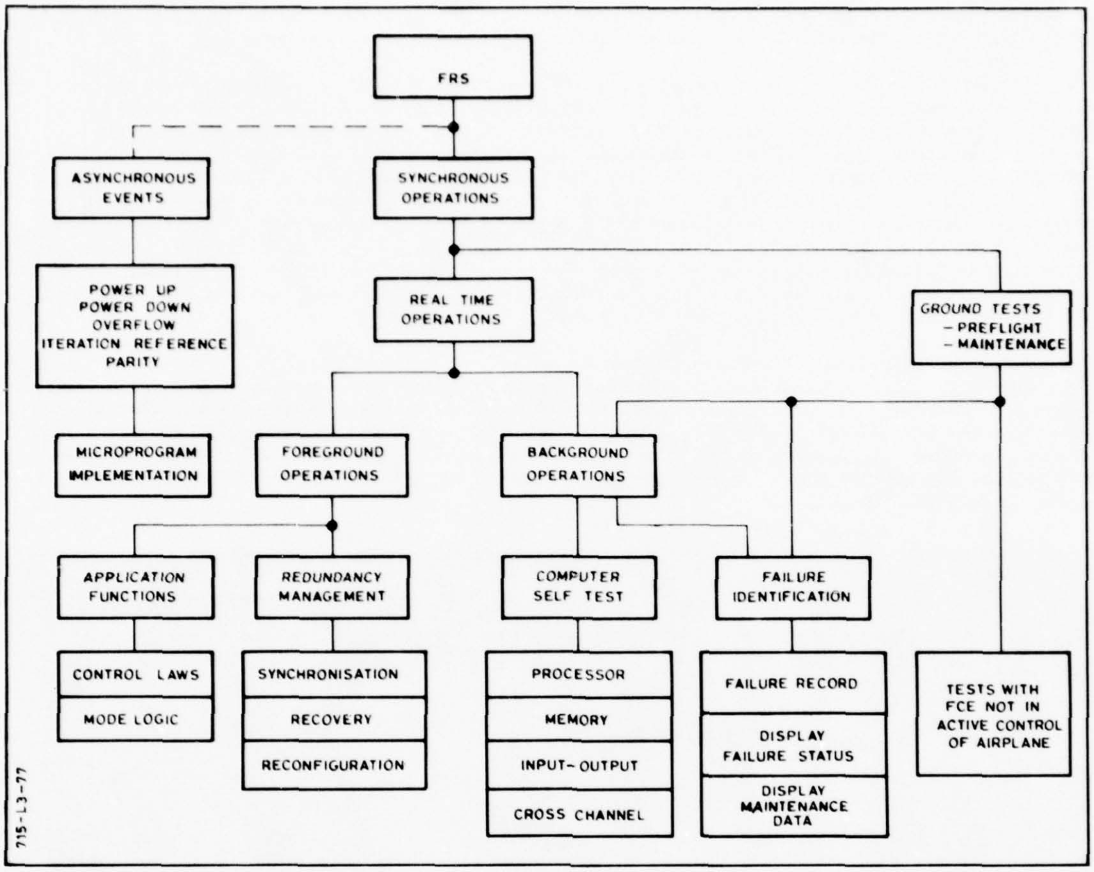

Figure 2 Flight Resident Software Structure

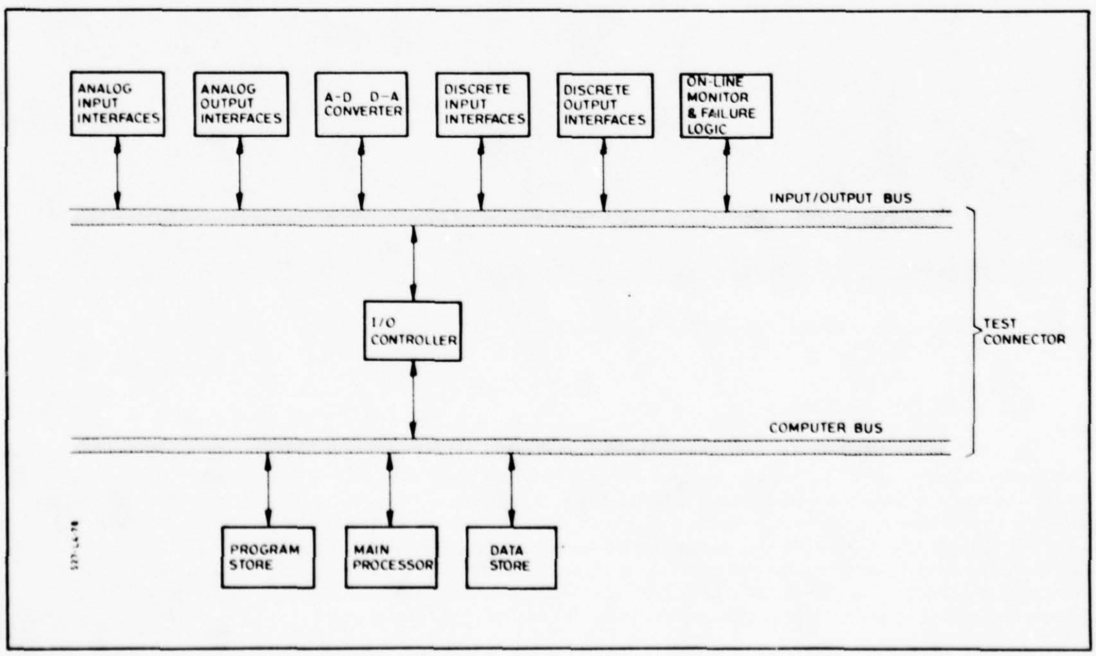

Figure 3 Multiple Bus Architecture

hardware, software and firmware mix appropriate to a specific application; the easily modifiable firmware providing the interface between application software and device hardware.

Since the advent of the planar transistor in 1959, semi-conductor complexity (components per chip) has, on average, doubled each year, culminating in the LSI microprocessor which offers great scope for increasing reliability and reducing weight and cost. There are currently two main microprocessor families: single chip devices having minimal space and power consumption requirements, and the more powerful bit-slice devices which comprise typically 2 or 4 bit LSI components connected in parallel to implement a processor of the required word length. The latter devices are intrinsically faster than single chip microprocessors by virtue of their technology (low power Schottky TTL & ECL architecture). Their instruction sets can be microprogrammed to achieve a high degree of parallel operation, and hence significantly increase processor throughput and flexibility. The average instruction time for a bit slice device is of the order of 1 μsec or less, compared with 5-10 μsec for the smaller and simpler single chip microprocessor.

Semiconductor memory technology has kept pace with, if not moved ahead of, processor improvements. Current LSI devices provide much greater bit density and manufacturing yields, with no increase in packaging size or power consumption. Thus 64K bit dynamic RAMs are under development for temporary data storage, and larger capacity devices - up to 1 Megabit - are envisaged, based on electron beam photo engravure to obtain the necessary resolution. 32K bit ROMs and 16K bit PROMs will be available within 2 years and 16K bit EPROMs have recently been introduced, which can be used during system development.

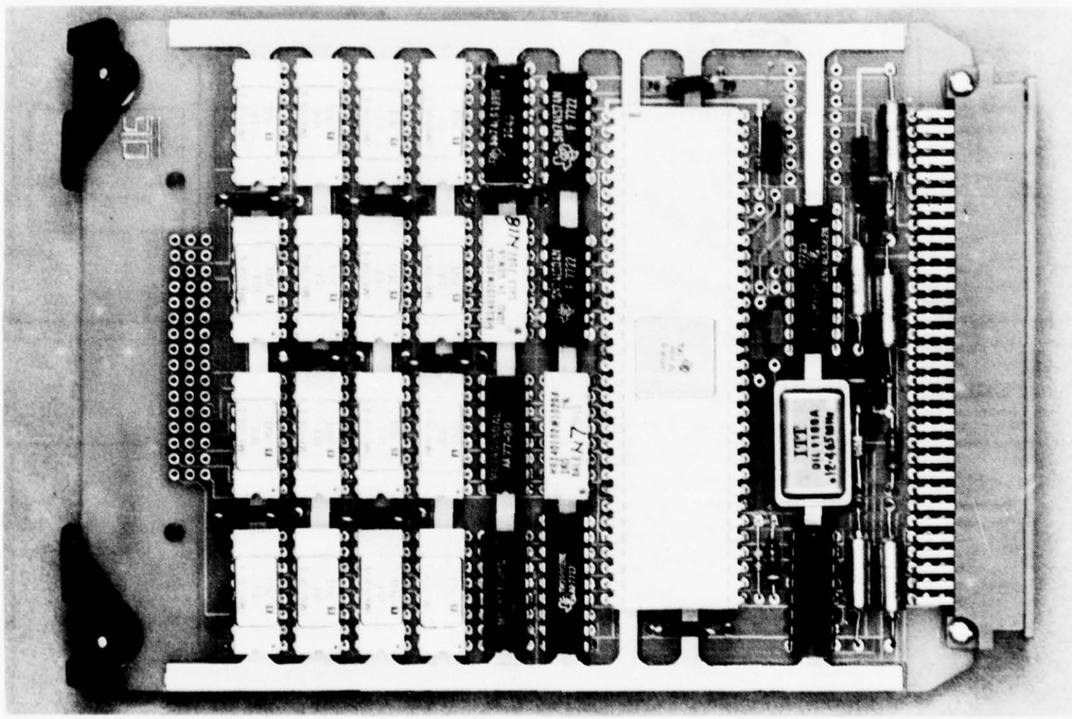

Figure 4 Typical Module Including Microprocessor

4. FLEXIBILITY OF DESIGN

The key to the modularity and flexibility of the flight control systems are the bidirectional, 16-bit multiple bus architectures. The standardisation of I/O control and data exchange, inherent in the use of bus organised architectures, provides the flexibility necessary to satisfy a variety of diverse military applications, embracing various levels of redundancy and degrees of self monitoring and involving a multiplicity of different peripheral interfaces. It merely requires the use of a range of standardised functional modules interfaced via standardised multi layer motherboards. Its simplicity contrasts markedly with the variety of device configurations employed hitherto for both the analogue and the earlier generations of digital mechanisation of high integrity systems.

The use of bus architectures, coupled with adoption of common data exchange standards (eg MIL-STD-1553 for military applications and ARINC 429 for civil applications), facilitates expansion by the addition of input or output devices to the I/O bus. Maintainability is considerably enhanced at LRU and module level as ATE is easily interfaced with the internal LRU buses. Transfer of data within typical flight control computers is facilitated by the Direct Memory Access (DMA) technique in which an I/O Controller 'handshake' mechanism determines priority and duration of I/O access. The use of 'cycle stealing' techniques allows processing to continue uninterrupted during data transfer.

5. HIGH INTEGRITY SYSTEMS ARCHITECTURES

High integrity system architectures enable bounds to be placed on the effects of individual system malfunctions commensurate with the degree of hazard involved. Multiplexed systems employing only cross channel comparison for safety monitoring are potentially susceptible to common mode failures, common design errors and dormant failures, and close attention to the design, analysis, and testing is therefore essential to preclude such conditions. Common mode failures may be induced by the external environment and by cross channel or cross-lane fault propagation. The incidence of common design errors is minimised by design visibility, coupled with adherence to strict software development procedures together with rigorous testing. The use of dissimilar monitoring in each computing channel can also provide fail safe protection against such errors, and can be used to detect dormant failures.

Visibility of hardware design is achieved by a combination of modularity, the minimisation of complexity consistent with efficient task mechanisation, and the minimisation of undefined and asynchronous operations such as interrupts. The use of a separate microprocessor for DMA I/O Control, operating in parallel with the main processor, results in a clear hardware and timing interface between the real world and the software. The main processor is specifically designed for the flight control task, and a compromise can be achieved between program size and processor complexity which minimizes design errors in either software or hardware. Aided by the hardware design, software can be readily structured into small independent modules, whose addressing and control sequence can be made independent of time and instantaneous parameter values thereby facilitating the analysis and testing of intermodule communications.

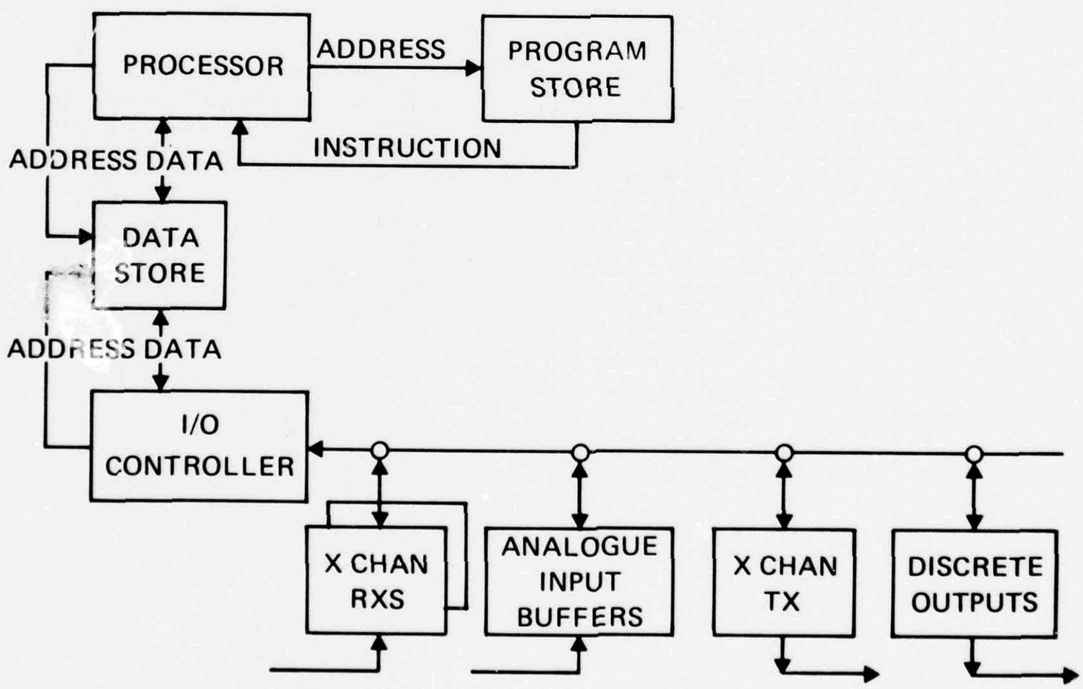

Figure 5 Dual Processor - I/O Control

The use of a separate micro processor for the DMA control of I/O operations reduces the task of the main processor, and provides considerable scope for increasing the extent of self monitoring throughout the system. Hitherto, self monitoring of a single processor has been used mainly to augment the cross comparison monitoring of redundant systems to guard against dormant and common mode failures and to increase system availability. Such self monitoring is essentially passive in that signal stimuli are not injected into the system. More rigorous, active, monitoring techniques can now be persued. Self monitoring is implemented using test software specifically designed to exercise all hardware digital functions over their entire operating range in a manner independent of any given flight software. Moreover, it constitutes a dissimilar form of monitoring and can provide worst case testing. Where dissimilar software monitoring is also executed by a separate processor, protection is provided against main processor design faults.

For certain critical elements of digital systems, dedicated hardware monitors are used to complement the self test software routines. The most likely configurations for near term future systems will employ an optimum mix of self monitoring, cross comparison monitoring and dissimilar monitoring, as governed by the particular system requirements.

6. MULTIPROCESSOR CONFIGURATIONS

Future flight control systems will employ more elaborate multiprocessor configurations, in which the partitioning of functions among distributed processors can provide a measure of graceful degradation for the overall system, as well as relaxing size, performance and power consumption constraints which apply to present processor designs. Possible configurations range from:

a) Several processors interconnected by a data bus accessing common data and program stores and the interfaces.

to b) Several processors, each with its own program store, local data store and I/O interfaces communicating via either a common data bus or a common data store.

Task allocation in such configurations can also take a variety of forms, for example, each processor can either perform a particular system function thereby restoring the axis segregation characteristic of analog systems with the minimum hardware impact. Alternatively, each processor can undertake a multijob task on a next-available priority basis, which enables tasks to be redistributed amongst processors under failure conditions and permits the use of 'hot spares' to improve reliability. Distributed task configurations can also be used with microprocessors performing or controlling interface functions.

7. CONCLUSIONS

It is now possible to develop flight control systems which have the flexibility to meet a range of requirements. This flexibility is based on bus orientated design and modular input-output and processor structures. For most applications the required levels of integrity can be achieved by the use of cross channel and self monitoring. For the highest levels of integrity, such as full time fly-by-wire, the problem of common failure can be overcome by designing for visibility and by introducing elements of dissimilarity where visibility cannot reach the level required.

In systems where full time operation is required and where environmental factors will lead to derangement of more than one channel then some form of hardened standby facility must be provided which survives the derangement. Where the environmental factors cause only temporary disturbance the system can be designed to recover from the transient by detecting the transient and entering a realignment mode.

DEVELOPMENT OF THE INTEGRATED FLIGHT TRAJECTORY CONTROL CONCEPT

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SUMMARY

Operational missions into heavily defended target zones with the likelihood for deviations from the intended routes and redirections from Command and Control centers will impose heavy workload demands upon the pilot/aircrew. This paper describes the approach selected by the Integrated Flight Trajectory Control (IFTC) program: 1) combine the functions of flight control systems and navigation computers, 2) develop techniques for four dimensional trajectory generation and display, and 3) develop procedures for operating on information received via data link. The system is complementary to the pilot and, by its logical operation, reduces the potential for pilot error in high stress situations. The paper also describes the operational advantages offered by the system and the method of evaluating its performance.

1. INTRODUCTION

The proliferation of the enemy military forces in numbers and in sophistication has been impressive in the past decade. This growth and the ability to apply that military force quickly and anywhere in the world, stresses the importance of a demonstrated deterrent capability which can react immediately and apply the appropriate force in any contingency. Strategic power, tactical power, global mobility and a precise, well-ordered strike capability will be necessary parts of that required capability.

The next confrontation will, in all likelihood, be intense tactically and non-nuclear. The battle area will be heavily defended by a multitude of defensive weapons (SAMs, AAA, etc) supported by enemy fighter aircraft. The airpower of the enemy will be formidable. Enemy fighters will be encountered in numbers far greater than in past conflicts. In fact, the number of enemy fighters will probably exceed the number of friendly aircraft.

Not only will the friendly pilot have to be concerned about the unfriendly aircraft as a threat, but the ground defenses, as well, will be numerous and quite lethal. An accurate, up-to-date knowledge of enemy air and ground defenses and the capability to maneuver through those defenses in a path of minimum exposure will certainly increase the survivability, and hence effectiveness, of our friendly aircraft.

Communication and precise time-space coordination have been highlighted as key capabilities required to successfully handle this conflict. The ability to put forth a well-ordered strike force with capabilities to strike at night and in adverse weather and to redirect to targets of higher priority, help to offset the difference in numbers of aircraft that the enemy will possess. However, as the pilot is given increasingly greater capabilities (more sophisticated weapons, aircraft and control freedom) and is required to meet more precise times on target after being redirected, the cockpit workload may become prohibitively high. Reduction of pilot workload, through levels of automation and simplification of controls and tasks in the cockpit, appears necessary to maintain a manageable level of effort for the pilot.

The Integrated Flight Trajectory Control (IFTC) program is concerned with these problems that the pilot will face in the future. To better appreciate the expected problems, the projected tactical environment and operational techniques need to be examined in more detail.

As indicated above, the tactical scenario is characterized by large numbers of enemy air and ground elements, & heavy concentrations of air-to-air and ground-to-air defense systems on both sides of the battle zone. The enemy's superior numbers of land and air elements requires the friendly air forces to maintain tactical air superiority through more efficient utilization of aircraft. Friendly aircraft will be required to fly missions into enemy territory under night and adverse weather conditions to neutralize and/or hamper enemy ground movements. To accomplish this, a high level of command and control (C^2) will be utilized for efficient allocation of available air and ground elements. A jam resistant, digital communication link will supply, in real time, command and control directives and tactical information from various sources to the cockpit for the pilot to assimilate and act upon.

This secure data link capability is currently being developed by the U.S. Department of Defense under the name, Joint Tactical Information Distribution System (JTIDS)[1]. Examples of the tactical information sources for JTIDS are: near real-time reconnaissance from Quick Strike Reconnaissance (QSR) aircraft, ground and air threat locations from the Precision Emitter Locator Strike System (PELSS), and aircraft track information (friendly and unfriendly) from the Airborne Warning and Control System (AWACS) and from other JTIDS user aircraft. Coupled with the tactical information, a precision navigation system such as the Global Positioning System (GPS)[2] or the JTIDS's Relative Navigation Capability [3] will be used for the tactical navigation grid to supply the accuracy needed for night/adverse weather operation.

Air operations in this dense air and ground threat environment will be characterized by complex, time critical mission profiles needed to coordinate the mix of strike, airlift and support aircraft. This precise time-space coordination will enhance,

- air assault missions where timely support from air defense, stand-off jammers, and gunships is critical for mission success and survivability,
- interdiction missions which require time scheduling of the suppression of enemy defenses while the strike aircraft proceeds with the weapon delivery in concert with other strike aircraft,
- airlift missions where air defenses and ground defense support in air drop areas is required for the time intervals when the transport aircraft are highly vulnerable to enemy attack, and
- night/adverse weather missions flown under instrument conditions which significantly contribute to the cockpit workload.

The real time redirection capability and the increased availability of tactical data provided by the JTIDS will obviously enhance the capability of the friendly forces. However, this capability for redirection, though beneficial, will increase the already high cockpit workload if not implemented properly. Targets will change, refueling will be rescheduled, ingress/egress routes will vary and all this can and will occur after takeoff, thus affecting the mission profiles and schedules of a number of air elements.

Restating, the IFTC program's concern is that in the dense threat environment, coupled with the vast availability of information and a redirect posture, the potential for increased pilot workload will in all likelihood make the pilot the limiting factor in the execution of time critical missions. Thus, the operational improvements that can be provided by C^2 and the advanced tactical systems may not be reliably achieved due to the inability of the pilot to translate the tactical situation information supplied to the cockpit into appropriate aircraft control actions.

Current state-of-the-art cockpit equipment (autopilots, flight directors, and Inertial Navigation Systems (INS)) provide pilot relief and steering cues for flying under constant attitude, heading, altitude, or speed conditions and for flying straight line profiles between stored mission destination points. Today, this equipment, along with navigation maps and hand calculations, are used to navigate along the mission route and meet all specified target and rendezvous times. Determinations of fuel quantities to be used during the mission segments and refuel points are computed during the pre-mission planning. Deviations from the original mission plan, which could be caused by the need to avoid the lethal airspace around a new enemy SAM location, the need to divert from the current course because of an approaching enemy aircraft, or profile changes made to take advantage of terrain features to minimize detection by enemy radars, will require the pilot to work with navigation maps and the new tactical data to determine the desired path back to the original mission profile. These disruptions will force the revisions of the profile, the time schedule, and the fuel predictions.

Mission redirects, especially redirects that require precise space-time coordination with other air elements, will require even more work by the pilot. The new mission route will need to be plotted on the navigation maps based on JTIDS alphanumeric messages, and using aircraft performance charts. It must be determined if the time schedule can be met and if there is sufficient fuel to execute the task and return to base or rendezvous with an available refuel tanker. Survivability must also be considered. In revising the mission profile, the threat situation needs to be considered. If the mission is achievable and warrants the risk, data describing the mission route will be entered into the cockpit equipment used for flying the aircraft along the new profile. If the pilot notifies the commander that he cannot comply with the mission redirect, the process will start over and a target that could be destroyed may be lost.

The level of this type of cockpit work is formidable for a pilot or aircrew under non-combat situations and is probably impossible to manage with the heavy stress expected in combat situations. Furthermore, detailed planning for mission changes and redirects, which have not been needed for effective operations in past military air conflicts, reduces the valuable time needed for operation of radar, communication receivers, and jamming equipment, and for early visual detection of enemy aircraft - a fundamental concern of all military pilots.

The IFTC program objective is to expand the flight management capabilities of on-board equipment to reduce the pilot workload needed to operate in a hostile tactical environment with both real time mission redirects provided by data links or pilot inputs under autonomous operation. This program is developing a system which adds flight management capabilities through the use of digital computers to integrate guidance and control with control/display, navigation and data link systems. 83

2. THE INTEGRATED FLIGHT TRAJECTORY CONTROL CONCEPT

The Integrated Flight Trajectory Control program is developing an on-board system that will provide pilots increased capability for real time management of their tactical missions. The expanded flight management capabilities will be achieved through the following:

- Real time computation of four-dimensional (x,y,z and time) trajectories in response to pilot or data link inputs. On-board computation of the mission profile relieves the pilot of this time consuming and mentally fatiguing task.

- Automatic guidance and control of the aircraft along the four-dimensional (4-D) trajectory to provide the capability of achieving assigned mission time-of-arrivals with minimum pilot effort while flying missions during night/adverse weather conditions. Precise aircraft control in both space and time is achieved using accurate radio and inertial navigation systems. This aircraft control capability complements the all weather target detection, weapon delivery and landing systems that exist or will exist on tactical aircraft.

- Integration of the control/display system with the 4-D trajectory generator and guidance and control functions to effect an efficient technique for pilot interaction with the IFTC system. Graphical representation of either a horizontal or vertical view of the tactical situation, including the computed trajectories, will be displayed on an electronic display. For pilot communication with the system, an interactive control/display consisting of a multifunction keyboard, dedicated mode keys, and an electronic display for presentation of alphanumeric data has been developed which minimizes the effort the pilot needs to use to create, modify, and/or engage new mission profiles.

The IFTC system will add flight management capabilities to both military transport and fighters with the trajectory generation, guidance and control, data link integration, and interactive controller/display techniques applicable to both assault and airlift missions.

2.1 System Description

Figure 1 illustrates the IFTC integration of trajectory generation, aircraft control, tactical communication systems, navigation systems, and control/display. Digital processing is the means through which this integration is achieved. The trajectory generator, the control law, and interactive control/displays processing make up the heart of the computer program. The computer processes, 1) aircraft position, velocity, and attitude inputs from the aircraft's navigation system, 2) tactical situation and C^2 data from the communication system, and 3) pilot inputs inserted through the interactive control/display hardware. The computer outputs aircraft control commands to the flight control system and data to the cockpit's displays and indicators.

The trajectory generator computes the 4-D trajectory based on data defining the mission profile, data describing the threat environments, and knowledge of the aircraft capabilities and constraints. The mission profile data is a sequence of points (waypoints, targets, rendezvous points, mission task initial points, and approach points) on the flight path with additional point parameters such as Time of Arrivals (TOA) and speed, defining points on the speed and time schedule. The trajectory generator computes a set of parameters that define all points on the horizontal and vertical parts of the trajectory, the speed and time schedule profiles, and the fuel requirements for the mission. The trajectory is adjusted to either avoid the lethal airspace around threats, or if necessary, penetrate them with a minimum of aircraft exposure. The aircraft performance capabilities, the operational limitations, and predicted winds are constraints on the computed trajectory.

Guidance and control of the aircraft is handled by a control law that tracks the computer generated profile according to the computed time schedule. The control law has both automatic and manual modes. In the automatic mode, which relieves the pilot of the tracking task, the control law generates elevator, aileron, and throttle commands to minimize the difference between the current aircraft state (position, velocity and attitude) and the desired aircraft state on the reference trajectory. Predictive information from the reference trajectory is included as terms in the control law's command to minimize the transient path tracking errors that could occur during trajectory direction and speed changes. Measurements of the current aircraft state are provided by the aircraft's navigation systems. The commands are fed to the servo actuators in the aircraft's flight control system for translation into control surface and throttle displacements. In the manual mode, the control law provides attitude and throttle commands on the Attitude Director Indicator (ADI).

The pilot's ability to communicate with the system's digital computer will determine the system's usefulness in the cockpit. Pilot workload is reduced by the automatic computation of the mission trajectory. These trajectories must be displayed relative to the tactical situation so the pilot can make the decision to engage, modify, or reject them. Graphical presentation of this information is desirable, thus the system utilizes an electronic display, referred to as the Situation Display, to present the tactical situation. Aircraft present position and track, engaged and alternate trajectories, lethal coverage areas of threats, and friendly and unfriendly aircraft locations are projected in a selectable horizontal or vertical format.

Operations using the interactive control/display have been structured to minimize both the mental and manual effort required for the pilot to communicate with the system. Understandable, logical, and reasonably simple procedures have been established for pilot interaction using a combination of electronic displays, dedicated and multifunction control keys, a hand controlled crosshair (on the Situation Display), and computer automation.

Data supplied via the tactical digital communication system is automatically input to the trajectory generator, the interactive control/display, and the Situation Display. Modifications of the present profile or a new trajectory are computed with the trajectory generator and displayed on the Situation Display in response to new threats or C^2 inputs. If required, the system presents an anticipatory engage profile presentation on the interactive control/display to simplify the pilot's flight management task.

2.2 Operational Capabilities

The United States Air Force flys a large percentage of its peace-time missions in controlled airspace with a mix of civil and military aircraft on prescribed, preplanned routes. Current cockpit equipment has sufficient capability to meet the demands of this type mission. However, in times of conflict when seemingly well-planned missions become confused by enemy ground forces (AAA, SAMs) and enemy aircraft, a more flexible system is required to unburden the pilot during and after these distractions.

The IFTC system has been designed with this volatile operational environment in mind. The IFTC trajectory generator accepts new points from the data link (C^2) and computes a new trajectory, after considering aircraft performance parameters, threats, and mission constraints, to produce a flyable trajectory. The Situation Display presents the trajectory. This newly computed trajectory is presented as a dashed line to distinguish it from the engaged profile. This method of presentation was selected to aid the pilot in recognizing that he has been directed to another target or that a threat avoidance zone has been discovered in this engaged flight path. This relieves the pilot of a significant amount of work such as computing has speed/time profiles and fuel requirements to complete the mission since this dashed profile is defined as being a physically flyable alternative. This change need not have been inserted via data link, but could have been input by the pilot using the keyboard to enter latitude and longitude or the range and bearing of new trajectory points. This operation is greatly simplified by using the hand controlled crosshair to designate new point locations on the Situation Display.

The capability for acceptance of data linked information into the cockpit for display and automatic input into the trajectory generator exhibits the potential for a significantly greater amount of pertinent tactical information to be received and evaluated by the pilot. This automation is accomplished while maintaining the necessary feature of permitting the pilot to scrutinize the incoming data and have the final decision authority. Without this capability, the visual recognition and the manual insertion of incoming data would saturate the pilot in very short order.

The parameters that can be specified with each point of the trajectory are not limited to latitude, longitude, altitude and time but such parameters as heading, flight path angle, turn radius and speed are accepted. These parameters can be inserted and the trajectory generator automatically computes the profile. This capability lends itself to weapon delivery missions in low visibility conditions where the aircraft's heading, flight path angle, and speed must be controlled for the desired delivery path, which without a trajectory generator places a significant workload requirement on the pilot.

It is widely felt that the battle zones of the next conflict will be highly saturated with SAM and AAA implacements. In this situation the IFTC threat avoidance/least exposure computations are of significant benefit. Present methods dictate that the mission's preplanned route avoids known implacements. The pilot must perform defensive maneuvers when warned by the on-board equipment. It is at these times that the pilot begins to lose track of his position in relation to the target and especially of his time schedule. With the increased capabilities afforded by the IFTC system the pilot will continue to have control of his aircraft to perform necessary defensive and/or offensive maneuvers, with his best intercept back to the original path or a new, more direct path being constantly computed and displayed. This continuous precise updating of the aircraft parameters, such as position, speed, time on target, and fuel remaining, which can be transmitted to C^2 via the secure data link will significantly aid the C^2 capability to utilize the strike forces to their greatest advantage. Knowing the fuel situation of each aircraft is beneficial in the prioritizing of the refueling operation without a high level of voice communication.

The flight management capabilities provided by the IFTC system when centralized C^2 inputs are provided by a digital data link have been addressed above. However, when the aircraft is operating autonomously, or with several other aircraft in a local area with voice only commands of a Forward Air Controller, the IFTC system capabilities are still beneficial in reducing the computations and data entry (workload) requirements that are imposed by present data systems. Operating without benefit of data linked information, the aircraft can respond in minimum time and with minimum work effort, to radio contacts with controllers.

A comparison of several important gimbal and strapdown characteristics is shown in Figure 4.

In summary, it is felt that the operational benefits derived from the IFTC system will meet the requirements for a quick reaction, precision time-space control system while providing the flexibility for C^2 redirects. The pilot workload is limited to a level that is equal to or less than presently encountered in either fighters or transports. 8-5

3.0 IFTC DEMONSTRATION PROGRAM

The current IFTC program, which is the outgrowth of an earlier study that applied the IFTC concept to the terminal area control of military transports [4],[5], is applying the system to a fighter vehicle and mission. The following has been accomplished.

- The 4-D trajectory generation and control law equations have been developed for a representative fighter.
- The interactive control/display operation, Situation Display presentation, and other cockpit instrument outputs have been structured for mission tasks typical of a fighter involved in air-to-ground weapon delivery.
- Methods for automatically accepting tactical situation data and C^2 directives supplied by digital communication systems in the trajectory generator and interactive control/displays have been developed.
- A hybrid computer simulation of the above capabilities has been developed with the following features:
    - A cockpit with F-16 dimensions, see Figure 2
    - An F-4 aircraft model
    - Interactive control/display, an electronic Situation Display, an electronic ADI, and other indicators are installed in the cockpit.
    - The capability for inserting tactical digital data messages as well as voiced command and control inputs into the cockpit.

Figure 3 shows the simulator configuration. The hardware elements in the simulator are as follows:

- Applied Dynamics AD-256 Analog Computer
- Applied Dynamics AD-4 Analog/Hybrid Computer
- IBM-370 Digital Computer
- Digital Equipment Corporation Model PDP-11/20 and PDP-11/03
- Hughes Conograph Graphic System
- Single Seat Cockpit

Man-in-the-loop testing in the fighter cockpit will demonstrate and evaluate the IFTC system.

4.0 FOUR-DIMENSIONAL TRAJECTORY GENERATION

The backbone of the IFTC concept is the ability of the on-board digital computer to generate a four-dimensional trajectory in space in response to pilot or data link inputs. The fundamental requirements of the trajectory generator include the ability to,

- construct a curved, three-dimensional path between points in space
- construct a "time" profile
- verify a flyable trajectory

Each trajectory is constructed from a sequence of points and specified parameters associated with these points.

The 4-D trajectory is divided into, 1) horizontal curved paths consisting of constant radius turns and straight line segments used to join successive points in the mission profile, 2) vertical paths made up of constant flight path angles to change altitude, and 3) a time schedule and corresonding speed profile. The basic computational blocks of the trajectory generator are shown in Figure 4. The threat avoidance equations determine if any mission points lie in the defined lethal airspace volume around each threat, or if any straight path between successive points intersects the threat volumes. If any intersections exist, new trajectory points are created that define a profile that avoids the threat volume.

4.1 Trajectory Point Parameters

The curved horizontal paths computed by the trajectory generator afford a greater flexibility for mission synthesis by allowing, but not requiring, many parameters at each point to be specified. As a result a complex, curved trajectory is computed with fewer number of x, y, z (only) points needed for definition than is possible with conventionally defined profiles. The complete point definition may include the following parameters in addition to the required space coordinates:

- TIME COORDINATE (DESIRED TIME-OF-ARRIVALS)
- TRACK ANGLE
- AIRSPEED
- VERTICAL FLIGHT PATH ANGLE
- TURN RADIUS
- WIND VECTOR

4.2.1 Functional description

Generally the function of the point in the mission determines which parameters will be specified. For example, the simple waypoints used to define the enroute navigation pro-
8-6 file may only require latitude, longitude, and altitude for specification, while weapon delivery initial points (IPs) may additionally require the specification of track angle, airspeed, and time-of-arrival.

4.2 Threat Avoidance

As described in the introduction, a dense threat environment will exist in the future tactical arena. The trajectories computed by the IFTC system will either avoid the lethal airspaces around the threats or, if required, penetrate the lethal airspace with flight paths that minimize the aircraft's exposure to the threat. The IFTC trajectory generator models the ground based threats as vertically oriented cylinders with radii and height approximating the lethal range of the threats. This threat envelope is adequate to demonstrate the action of the trajectory generator relative to the threat data present in the IFTC system.

Currently, the threat avoidance equations use the known threat locations and types and determines those profile segments which will penetrate the lethal range envelopes. The equations will add points to the original sequence of points defining the trajectory that will cause the trajectory to avoid the threat volume (or volumes) by stretching the horizontal or vertical paths. Associated with these points are the inbound and outbound turn radii, ground track angles, and flight path angles needed to construct the avoidance segments. The avoidance segments are generated to minimize the total deviation from the original trajectory. This strategy reduces the effect on any time-of-arrival constraints placed on the mission.

Before the IFTC program is completed the threat avoidance equations will be developed further to purposefully allow penetration of the threat volumes. There are several reasons to deliberately penetrate a threat volume:

- The extra path length to avoid the threat and the corresponding time delay may preclude meeting the specified priority time schedule.
- The threat may be unavoidable because of aircraft dynamic limitations.
- The need for weapon delivery on a target within the coverage of one or more threats.

The strategy for deliberate penetration, of course, will be to minimize the aircraft exposure to the threat.

4.3 Horizontal Path Generation

To determine the total horizontal path, the sequence of points defining the trajectory are processed sequentially in pairs by the IFTC horizontal path generation equations until the mission list is exhausted. The horizontal path generator determines the shortest path between the two points in each pair using an initial turn, a straight segment and a final turn. This path type is required only if the second point, referred to as the "to" point, has a specified ground track angle to be satisfied at the instant of flyover, as might be the case for weapon delivery or rendezvous initial points (IPs).

If the track angle is not specified, the horizontal path reduces to a simple turn followed by a straight segment toward the "to" waypoint. Points with no specified track angle are used as general navigation points, and the trajectory generator does not force "flyover" of the points. The profile will smoothly and predictably transition from the inbound course to the outbound course.

The parameters determined by the trajectory generation equations and transmitted to the Simulation Display, interactive control/display, and control law, when the trajectory is engaged, are shown in Figure 5. For two successive points, P_1 and P_2, defined by lattitude and longitude and other pertinent parameters, the horizontal path is generated by computing,

a. L and Ω, the length and heading of any straight-line segment between points of tangency with turns at P_1 and P_2.

b. XTR1, YTR1 and XTR2, YTR2, the coordinates of the points of tangency.

c. XCR1, YCR1 and XCR2, YCR2, the coordinates of the centers of the turns at P_1 and P_2.

d. LCR1, LCR2, the length of the curved path segment around the turns.

If a turn radius is not specified, the trajectory algorithm establishes the radius as a function of the specified, or a predicted airspeed, and a nominal aircraft bank angle.

4.4 Vertical Path Generator

For an altitude change between point pairs the vertical path generator uses either one flight path angle (FPA) for the entire path between the two points, when a flight path angle has not been specified, or two FPAs consisting of a segment with a specified FPA and

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a zero FPA segment. Whenever aerodynamic constraints or a pilot-specified FPA prevents the aircraft from changing altitude with only FPA changes, a spiral descent or ascent is generated to indicate to the pilot, via the Situation Display, how the altitude change can be made. 8-7

When unspecified, the FPA for the i^{th} point pairs is computed as

$$\gamma_i = \tan^{-1}((H_{i-1}-H_i)/i^{th} \text{ segment length})$$

For this type of altitude change the aircraft flies with a constant γ_i, for the total segment. If the FPA is outside the allowable range of flight path angles, which is established by drag, available thrust and weight, a spiral profile is generated using the technique described below.

The two-FPA type is used for situations involving altitude increases or decreases with specified FPAs. The generator determines the position along the horizontal profile at which commencing an ascent or descent at the specified FPA will also satisfy the specified altitude at the "to" point.

A spiral maneuver, is computed whenever: 1) the altitude change for the point pair cannot be achieved with the specified flight path angle, or 2) the altitude change cannot be accomplished with a vertical flight path between the maximum and minimum flight path angle constraints of the aircraft. The spiral maneuver consists of circular turns with an acceptable FPA either following the first point in the point pair or preceding the "to" point.

4.5 Speed/Time Profile Generation

The speed/time profile generator computes both a speed profile and a time schedule for the 4-D trajectory. The speed profile, which is an airspeed profile, and time schedule are both functions of the distance along the horizontal trajectory. The speed profile agrees with any specified airspeeds, while the time schedule, which is calculated using available wind information, is computed to minimize the differences between specified time of arrivals (TOAs) and computed TOAs. The speed profile is generated with a minimum number of speed changes to achieve a profile that is logical and understandable from the pilot's point of view.

The technique used in generating the speed profile and time schedule involves computing an acceleration profile for the aircraft. This profile, which is also a function of distance along the horizontal path, consists of constant acceleration (or deceleration) segments positioned along the horizontal path. Integration of the acceleration profile establishes the speed profile. The speed and wind profiles are used to determine the time schedule.

Figure 6 shows the primary computation blocks of the speed/time profile generator, which iteratively adjusts the acceleration profile to minimize the difference between the computed time schedule and the specified TOAs. This processing technique accounts for the specified speeds, limits the speed profile so it does not exceed the aircraft's maximum and minimum airspeeds, and constrains the acceleration profile so that at any point on the trajectory the computed acceleration (or deceleration) does not exceed the available aircraft acceleration or deceleration. The available acceleration and deceleration are computed in terms of flight path angle, drag, weight and available thrust.

For initialization the speed/time profile generator ignores the specified TOAs and determines an acceleration profile that integrates into a speed profile satisfying all specified speeds. The initial point on the speed profile is the current aircraft airspeed, since the first point on all 4-D trajectories computed by the IFTC system is the current aircraft state. The initial acceleration profile has a minimum number of non-zero acceleration segments - only the number required to change the speed to agree with the specified speeds.

The first step in the iteration process, see Figure 6, is to calculate the speed profile and time schedule over all the point pairs up to and including the point with the first specified TOA. Then the difference between the specified TOA and computed TOA for this last point is computed. If the TOA error is sufficiently small, this part of the profile is complete. (The process is also complete if the trajectory does not have a specified TOA.) Otherwise, the acceleration profile is perturbed to reduce the TOA error. (The technique for perturbing the acceleration profile is discussed below.) The iteration process continues with the speed profile and time schedule being computed, the TOA error determined, and acceleration profile perturbed until either the TOA error is nulled, or has converged to a steady state value. Analysis and simulation runs performed during this program indicate this iterative technique is convergent.

A non-zero steady state TOA error means the aircraft speed and acceleration constraints have been reached and the acceleration segments have been positioned to minimize the magnitude of the time error. In this situation, the trajectory generator is indicating the aircraft will not meet the assigned TOA. This information is displayed on the Situation Display for the pilot, in case he wants to modify the 3-D trajectory or specified speeds to achieve the TOA.

8-8 After computing the speed profile and time schedule between the aircraft and the point with the first specified TOA, the process is repeated for each set of point pairs between successive specified TOAs until the trajectory is completed.

The technique used to perturb the acceleration profile and minimize the TOA error uses parameters that define the sensitivity of the TOA error to changes in the acceleration profile. The time error sensitivity to accelerating (or decelerating) the aircraft earlier or later on the trajectory is used first to adjust the location of the acceleration segments to reduce the time error. If the equations determine that segment repositioning cannot further reduce the TOA error, then additional non-zero acceleration segments are added to introduce an intermediate speed in between successive specified speeds. The time error sensitivities to the value of this intermediate speed and the acceleration (or deceleration) value used to make the speed change are then used to make the final perturbations on the acceleration profile.

While minimum fuel usage was not a design objective, the 4-D trajectory generation equations can be modified to add this capability. The technique will take into account the economy speed/altitude profile parameters output from a minimum fuel algorithm.

5.0 THE CONTROL LAW

The IFTC system has the capability to automatically track the computer generated three-dimensional trajectory according to the calculated time schedule, or provide guidance cues for display to allow the pilot to fly the computer synthesized 4-D trajectory. The control law processes measurements of the current aircraft state and parameters defining the 4-D trajectory that has been engaged by the pilot. For the automatic operation, the control law determines aileron, elevator, and throttle commands. For the manual mode, the control law determines roll, pitch, and throttle change commands that are displayed on the ADI for pilot interpretation and appropriate action.

The control law processing in the automatic mode, as shown in Figure 7, is divided into the following computations: the reference state $\underline{X}_0(t^*)$ and nominal control commands, the trajectory tracking error vector $\delta\underline{X}(t)$, and the control command, $\underline{U}(t)$. The sum of tracking error feedback and nominal commands make up the total control command. Feedback of the tracking errors minimizes the steady state spatial and time differences between the 4-D trajectory and the aircraft. Feed forward of the nominal commands, from the reference trajectory, minimizes the transient tracking errors that can occur during turns, changes in altitude, and periods of acceleration.

The definition used for the reference state, $X_0(t)$, which is differenced with the aircraft state to compute the tracking error, has a pronounced effect on the control law performance. For the IFTC system the reference state's x, y components are defined by the location on the desired horizontal path that is the closest point to the aircraft. The time t^* is the time the aircraft should have been at the reference state. When the tracking error vector is calculated with this reference state, the cross track distance error and time error can be made separate components. This separation prevents cross axis coupling between time error and lateral error being introduced by the tracking error feedback through the control law; i.e., time errors do not cause cross track errors and cross track errors do not cause time errors. Figure 8 represents the distinctions between the current aircraft locations $(x(t), y(t))$, the desired aircraft location on the reference trajectory $(x_0(t), y_0(t))$, and the reference state $(x_0(t^*), y_0(t^*))$. The cross track distance error is δx_c and the time error is $\delta t = t^*-t$. This reference state definition is consistent with what other investigators [6,7] that are concerned with tracking a time based trajectory have used in their control law mechanization.

Expressed in terms of the tracking error components and the nominal control commands, the equations for the aileron, elevator, and throttle change commands are as follows:

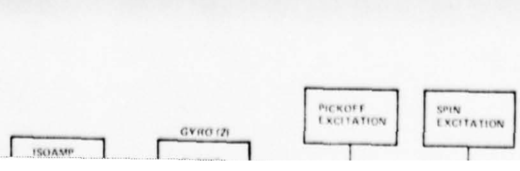
$$\delta_\phi = G_{11}\delta x_c + G_{12}V_G\delta\psi + K_\phi(\phi_0 - \phi) \tag{1}$$

$$\delta_e = G_{23}\delta z + G_{24}V_G\delta\gamma \tag{2}$$

$$\delta_T = G_{35}\delta t + G_{36}V_G^{-1}\delta V + K_T A_0 \tag{3}$$

with $\delta\psi$ the track angle error, δz the altitude error, $\delta\gamma$ the flight path angle error, δV the ground speed error, V_G the computed ground speed, ϕ_0 the nominal bank command, A_0 the nominal acceleration command, and ϕ the current aircraft roll angle. Each trajectory tracking error is the difference between the value on the reference trajectory at time (t^*) and the current aircraft measurement. The nominal commands, which include the flight path angle γ_0 that is part of the $\delta\gamma$ error, are valid at time t^* with anticipation times added when the reference trajectory has an abrupt change in roll, flight path angle, or acceleration.

The reference state and control law have been structured so that each command will handle a single dimension of the tracking problem, with proportional plus rate feedback provided in each command. The proportional feedback nulls out the displacement error and rate feedback provides path damping. In the IFTC program where the control law is implemented in a hybrid simulator, aircraft stability is augmented with roll rate



damping added to the aileron command, a blend of pitch rate and normal acceleration added to the elevator command, and yaw rate feedback for Dutch roll damping and lateral acceleration feedback for turn coordination. The feedback gains have been selected using classical analysis techniques. The gains will be verified during the testing with an F-4 aircraft model in the simulator. 8-9

To be consistent with the trajectory generator's method for computing turns, the nominal roll command ϕ_0 will be time varying during each turn. The trajectory generator uses constant radius turns with a constant airspeed in generating curved trajectories. Consequently, the desired ground speed and bank angle are functions of the predicted wind magnitude and direction and the current aircraft location on the turn.

In the simulator the pilot will be able to override the control law's automatic aileron and elevator commands by applying force to his flight control stick, or he can override the throttle command by disengaging the throttle servo. This is an operational advantage because it allows the pilot, with a minimum of effort, to deviate from the engaged profile for such things as evasive maneuvers and terrain shadowing.

However, after the manual override has been removed it is difficult for the linear feedback control law given by Eqns. (1), (2) and (3) to return the aircraft to the engaged 4-D trajectory if there are large position, speed or time tracking errors present. To handle these errors the control law operation has been integrated with the trajectory generator. When a large position tracking error exists, the trajectory generator computes a 3-D capture segment from the current aircraft location to the next point on the engaged trajectory and computes a revised speed and time schedule for the complete trajectory, including capture segment. When a large speed or time error exists, only a revised speed and time schedule are calculated. The control law then uses this new trajectory as its reference, and since the tracking errors are now small, a stable control system operation results.

6.0 CONTROL DISPLAY INTEGRATION

Real-time, in-flight management by the pilot of complex four-dimensional trajectories imposes new and stringent requirements on the display system.

Pilots presently spend approximately 2-3 hours preplanning for each hour of mission flight time. During the planning process they consider the waypoints, arrival times, flight routes and aircraft fuel and performance limitations to arrive at a combination of flight segments (a trajectory) that will accomplish the mission. The pilot is assisted in this planning process by printed material that helps him visualize the aircraft potential path in space during such complex maneuvers as steep curved descents. As a result of this mission preplanning, the pilot acquires a thorough understanding of the mission profile and the information requirements and constraints on which it is based.

The IFTC trajectory generator generates a complex trajectory that is a function of time-space position coordinates. This mission profile is new to the pilot and it may have to be modified in flight to meet changing mission requirements, or even to achieve the original mission requirements after unforeseen disruptions of the original profile.

Thus an analysis of control display system requirements for a system containing an on-board trajectory generator reveals two significant differences from current systems. These new requirements are found in the area of:

1. Providing sufficient information to the pilot to understand the newly generated trajectory in preparation for acceptance/engagement.

2. Providing in-flight confidence building information of aircraft situation with respect to command situation.

Research programs (STOLAND [6], Terminal Configured Vehicle - TCV) have demonstrated the usefulness of electronic horizontal map displays to provide the pilot with a readily understood graphic view of the aircraft with respect to the trajectory and significant terrain features.

The STOLAND and TCV also demonstrated the desirability of electronic displays of attitude and flight director command information and the need for an electronic alphanumeric display for data entry. The initial feasibility study of the IFTC application in a transport aircraft utilized a display system containing electronic attitude and flight director information, horizontal map information and alphanumeric system information [4]. It demonstrated the use of a special crosshair symbol on the Situation Display as a graphic means of creating new or modifying old waypoint and trajectory information. The transport program also demonstrated the usefulness of providing a level of computer/pilot interaction.

The control display implementation for the fighter demonstration is built on the knowledge gained in the transport program. Electronic displays provide attitude information (with roll, pitch, and throttle commands), situation (horizontal and vertical) information, and system alphanumeric information. The concept of computer/pilot interaction was developed to a higher level during the fighter demonstration study by providing mission oriented mode controls and expanded man/machine interaction (through digital processing) for mission and data management.

The gyro torquing current is used to develop a voltage across precision resistors. A solid state switch is employed to provide two ranges of operation, thus extending the dynamic range of the electronics to that provided by the gyroscopes. A "wash", or polarity reversal, technique is incorporated to effectively eliminate contamination of the data by bias vol-

8-10

The need for additional vertical information for in-flight management of the generated flight plan was recognized during the initial transport simulation program when complex spiral descents were generated to demonstrate the system capability to link together two points that are close together in latitude and longitude but widely separated in altitude. Thus a vertical profile display consisting of a view of flight plan altitude as a function of along track distance is presented in the fighter cockpit as the Vertical Situation Display (VSD) mode of the Situation Display. This presentation was chosen because of its similarity to the altitude/distance view pictorial information generated during mission planning or briefing sessions. Under mode control of the pilot, then, the Situation Display shows either a simple horizontal map or a vertical map. This combination of horizontal and vertical profile mapping information is being investigated as a means of providing the pilot with a more rapid, complete understanding of the generated profile.

Requirement 2, the need to provide the pilot with situation information with respect to commanded situation, is addressed in the fighter demonstration program by:

1. Providing the vertical situation mode on the Situation Display, and

2. Modifying the vertical scale air data displays to provide computer driven readouts of command altitude, command Mach, and command airspeed quantitively and with respect to actual flight values.

An additional requirement imposed for the fighter demonstration study was the requirement to demonstrate the integration of digital data link command and control and tactical situation information with the IFTC system. This combination is very complementary! Data linked command and control information consisting of waypoint, threat, or target locations and TOAs is readily accepted as inputs to the trajectory generator. A trajectory is generated based on the command and control requirements and on aircraft constraints and is computed for the pilot. With the IFTC system the pilot has the advantage of being able to review the data linked information as an integrated whole on a familiar format. He knows that the trajectory generator has considered speed, fuel, maneuverability and stored tactical information with the data linked command and control requirements to arrive at the trajectory shown.

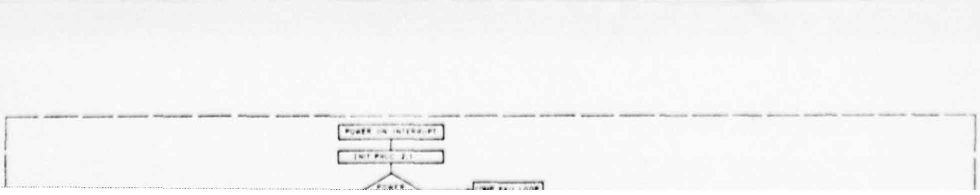
The control display system is shown installed in the fighter cockpit in Figure 9, with the following major features:

- It is designed for one-man operation.
- It considers the information requirements of the IFTC system and advanced command and control operation.
- It provides mission oriented mode selection.
- It extends the use of interactive control/display software to minimize the pilot's information management tasks.
- It explores the use of trajectory generator outputs as a means of providing the pilot with a higher level of decision making information than is possible without the trajectory generator.

The basic flight displays of indicated airspeed, attitude, altitude, and horizontal situation information are arranged in the familiar "T" scan pattern used in many Air Force aircraft. The interactive control/display and associated mode controls are located on the left side of the front panel. The alphanumeric keyboard and dedicated data management keys are located on the left console beneath the throttle.

The Situation Display, shown in Figure 10, provides horizontal situation information; i.e., a simplified map of the waypoints, threats and the trajectory (solid line) being flown between waypoints. In addition, a computed, but not engaged, trajectory is shown as a series of dashed lines. Threat information, such as surface-to-air missile envelopes and unfriendly aircraft, are displayed in appropriate locations on the Situation Display. There are two modes of horizontal situation information: Track up and North up. In addition, the HSI mode displays a compass rose with the aircraft in the center much in the same manner as an electromechanical Horizontal Situation Indicator (HSI). The Vertical Situation Mode (VSD) of the Situation Display provides an altitude vs along track distance format of vertical profile information. This vertical profile display was specifically generated to enable the pilot to get a better understanding of the three-dimensional aspects of the newly generated trajectory.

The interactive display, shown in Figure 11, is a CRT presenting system mode, trajectory related data, and status information in alphanumeric form. This information is presented in special formats to simplify the procedure for pilot-system interaction. The interaction is accomplished with a combination of dedicated and multifunction control keys, and hand-operated crosshair (on the Situation Display), and computer automation, see Figure 11. Decision trees for the control/display part of the computer program have been developed that establish a reasonable, simple and logical man-machine interaction procedure. By anticipating, in certain situations, what display format the pilot will need for the next interaction step, computer automation is being used to minimize the number of detailed instructions (and key actuations) needed to control a system with this flexibility and capability.



Over 30 formats have been developed for the interactive display to handle the data necessary for three active mission modes (NAVigation, BLD weapon delivery, and RDZ-rendezvous), waypoints, targets, refuel points, mission plans, threats, data link, and map display declutter/clutter options. Some formats highlight certain information with reverse video, where the color relationship of the characters and their immediate background is reversed, see Figure 10. The reverse video cues the pilot to particular information or designates the information selected by the pilot via the row/column switches immediately to the left of and below the display. Mission oriented mode select formats are presented whenever one of the seven mode select keys are pressed, with mode and trajectory selection being a three-step process: 8-11

1. Select the proper mode from the seven dedicated switches.

2. Review the mission oriented select format (see Figure 10) on the interactive display, editing it as desired to reflect pilot needs.

3. Engage the mode and/or trajectory -- to actuate the control law -- by pressing the ENGAGE key.

The IFTC system will accept the point information generated either by the pilot or by a command and control element via digital data link. The incoming point information is examined for completeness and processed by the trajectory generator into a four-dimensional trajectory (x, y, z and time) capable of being flown by the aircraft. If the incoming data is not complete, an appropriate message is placed on the interactive display indicating the data deficiencies. The trajectory generated is displayed on the Situation Display in a dashed line format to indicate that it is available for, but is not engaged for, flight control, see Figure 10. An appropriate select format is simultaneously (Figure 10) placed on the interactive display indicating to the pilot that the new plan may be engaged simply by pressing the ENGAGE key. Should the pilot wish to review additional information about the plan, he would press the DATA key and look at any level of detail that he wishes to examine. This example illustrates how the computer automatically computes and displays trajectories, from pilot or data link inputs, and places the next logical format on the interactive display, thus simplifying the pilot's task of engaging or changing the mission trajectory or mode.

For control of the simulator, the cockpit is configured with a standard right hand force-actuated control stick and a left hand throttle. The control stick contains multiple pushbuttons to control weapon release, Automatic Flight Control engage/disengage, aircraft trim and the intercom. The throttle grip has the control switches for the Situation Display crosshair, the speed brake and the autothrottle on/off switch.

The Electronic ADI (EADI) in the top center of the front panel (Figure 10) is a conventional electronic attitude display with horizontal and vertical flight director commands plus a moving tape throttle command display on the left wing of the miniature aircraft symbol. The two air data vertical scale instruments on either side of the EADI are modified to provide a computer-driven indication of the relative direction and magnitude of command airspeed, Mach, and altitude with respect to the present aircraft flight values.

7.0 SYSTEM TESTING

The IFTC system will be demonstrated and evaluated in a man-in-the-loop simulation by applying the capabilities of the system in a realistic tactical scenario with changing mission requirements and unannounced disruptions, i.e., SAMs in direct line of flight. Six USAF pilots, on active duty, will be used as test subjects for the evaluation. As a basis for comparison, a control/display/navigation system possessing capabilities similar to those installed in present operational aircraft will be implemented and flown by each of the subject pilots. The overall mission performance of the test subjects, using the expanded flight management capabilities of the IFTC system, will be compared to their performance using the baseline system.

Air-to-ground weapon delivery missions in a hostile, tactical environment will be simulated. Each pilot will fly a multisegmented mission in the IFTC simulator and a similar mission using the baseline system for navigation and weapon delivery. Each pilot will start the simulation on a preplanned mission that consists of an initial refuel, a route over the FEBA to the target, and a route back to friendly territory to another refuel. The mission includes a number of time critical tasks and disruptions that will be used to demonstrate the usefulness of real time trajectory generation and control capabilities. Examples of the time critical events and disruptions are the following:

- Rendezvous for in-air refuel
- Cross a waypoint on the FEBA inbound to target zone with specified time of arrival
- Enroute update of target coordinates
- Encounter unfriendly aircraft
- Encounter SAM threat on trajectory
- Weapon delivery on primary target with specified time of arrival
- Redirect to new second target with specified arrival time
- Redirect to new egress route with specified arrival time
- Set up and rendezvous with new tanker for refuel

4.2.2.4 Strapdown System Outputs

The raw data are received from the IMU and compensated. The subsequent intergration utilizes a fourth-order Runge Kutta algorithm for the quaternion representation of attitude.

The evaluation will compare the IFTC performance with the baseline performance under two operational conditions. The first condition will supply the information via simulated data link to the alphanumeric displays, in the case of the baseline system, and automatically into the trajectory generator and displays in the IFTC system. Information for the second condition will come into the cockpit from a controller via the pilot's headphone for both the IFTC and the baseline system. Each subject will then be required to utilize the capabilities available to complete the mission requirements.

The experimenter will act as a command and control element to set up the various diversions via either voice link or digital data. The pilot's data management task will be to review the incoming data, inputting it as necessary to satisfy the requirements of either the IFTC system or the baseline system. His primary tasks will be to fly the aircraft so as to make good the various required arrival times and complete the required weapon deliveries accurately and on time. He will be expected to perform the navigation functions required by each system to complete the mission.

The systems will be assessed qualitatively with questionnaires administered before and after the tests and quantitatively by analyzing tracking error and man/machine interaction time and data entry errors.

CONCLUSION

In conclusion, the Integrated Flight Trajectory Control concept, when applied to the tactical fighter and transports, will result in more efficient, timely operation in the tactical environment. IFTC is the synergistic culmination of flight control, navigation management and display technologies combined with today's highly efficient digital computers, which can and will result in a more efficient air warfare system with reduced pilot workload.

Although the pilot-in-the-loop testing for the IFTC system, as applied to the tactical fighter scenario, is still approximately one month away (June 1978), sufficient testing has been done in the analytical effort and in the transport cockpit using tactical transport scenarios to promote a high confidence level that the same efficient operation will result in the fighter.

The control functions and the sophisticated trajectory generator alone serve to relieve the pilot of many time consuming, burdening tasks. This allows him the proper time to utilize the sophisticated strike systems with the net result of a higher mission success probability. This is especially true when considering the emphasis on command and control which will result in vast quantities of information being transmitted to and from the cockpit. This information will be of great benefit when managed properly in the cockpit.

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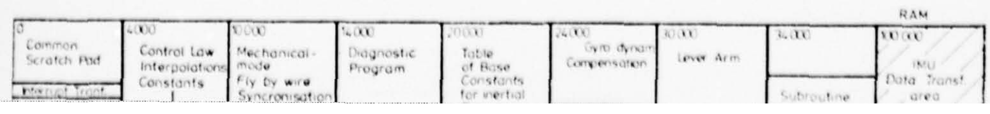

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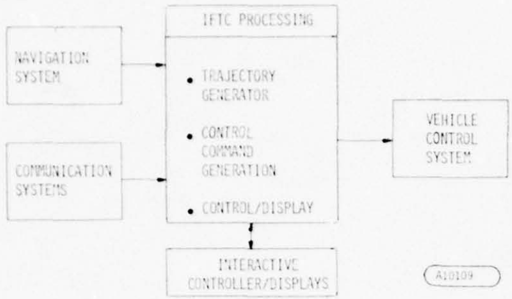

Figure 1. The IFTC Control System Block Diagram.

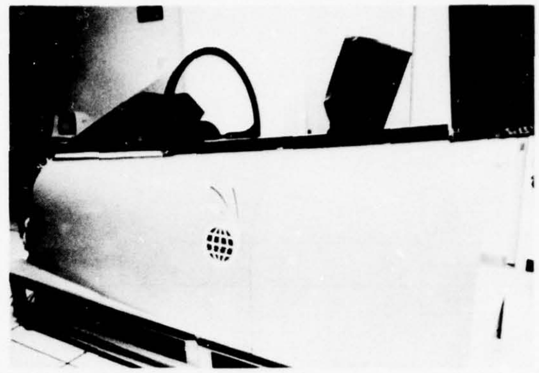

Figure 2. IFTC Cockpit Simulator.

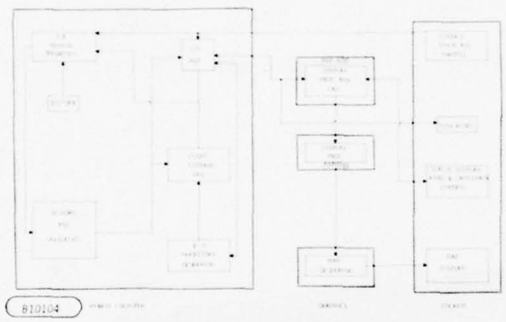

Figure 3. Simulator Elements.

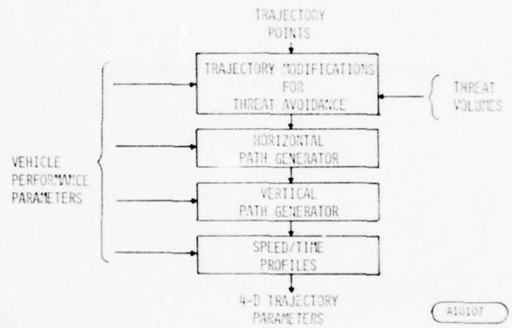

Figure 4. Trajectory Generator Block Diagram.

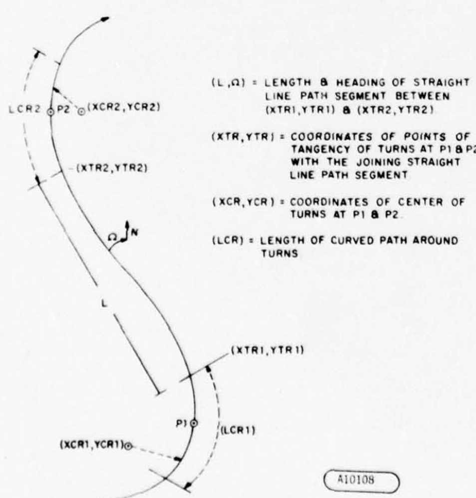

Figure 5. Parameters Calculated by Horizontal Path Generator.

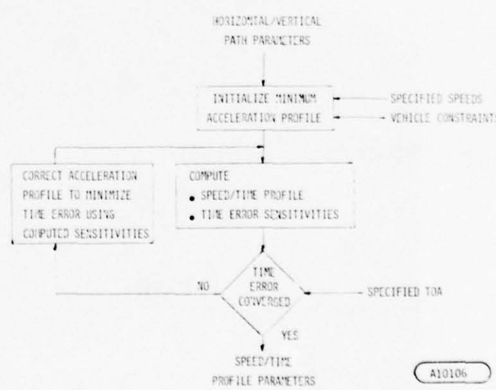

Figure 6. Speed/Time Computational Flow Diagram.

3.2.3 Flight Safety Considerations

Any equipment used in a flight-safety critical position has to satisfy two requirements.

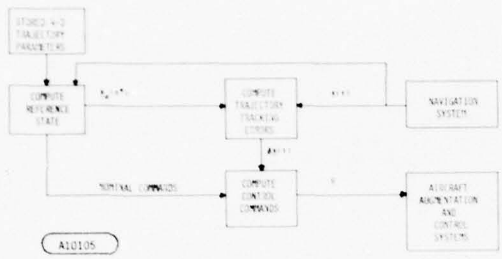

Figure 7. Control Law Block Diagram.

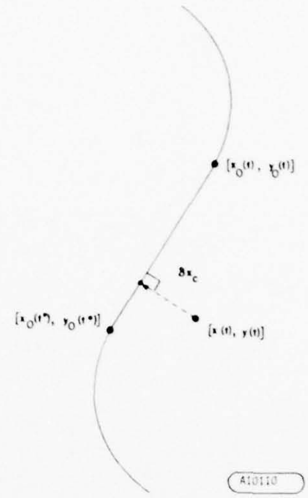

Figure 8. Reference State Definition.

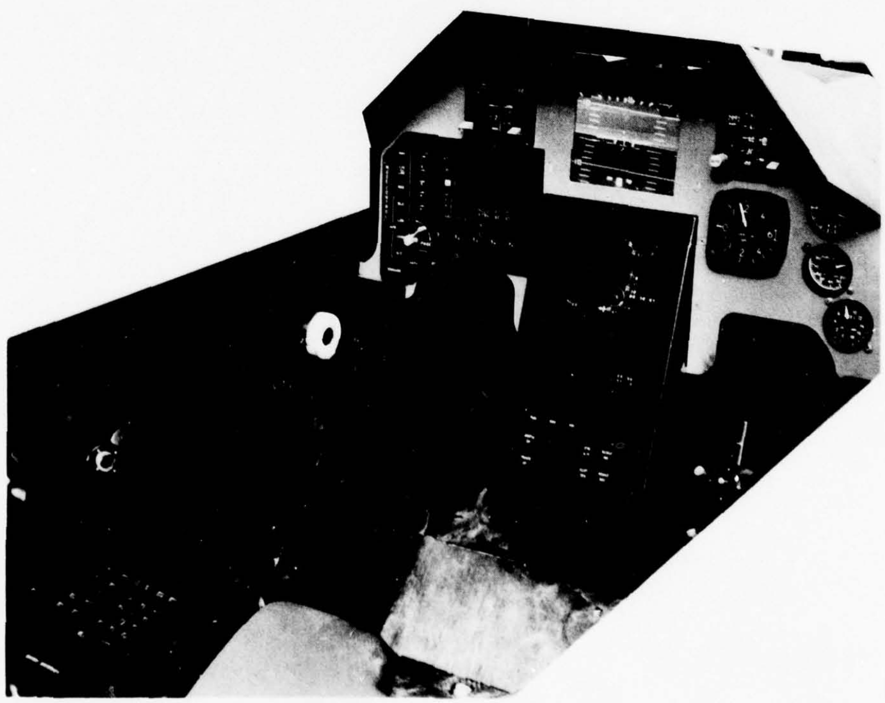

Figure 9. IFTC Control Display System.

As mentioned before, the computers run identical but nonsynchronized software. This feature, which is imposed by computer hardware, has resulted in an undesired effect. In general, it forces one to use higher failure thresholds. In case of the strapdown

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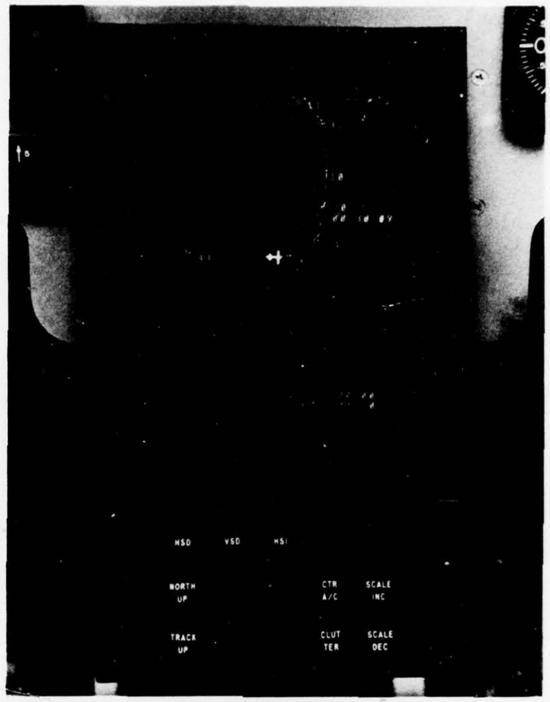

Figure 10. Situation Display.

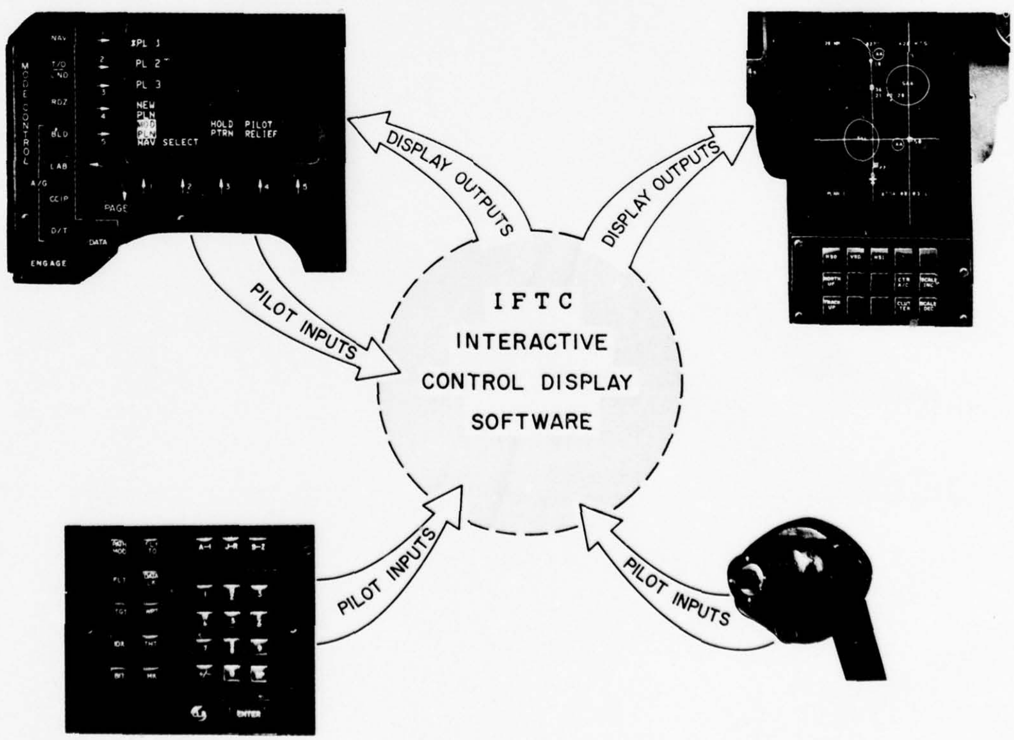

Figure 11. Interactive Control/Display.

Figure 21 may be used as a starting point for cost effectiveness considerations with the following sensor configurations.

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REDUNDANT STRAPDOWN NAVIGATION, GUIDANCE, AND CONTROL OF A CONTROL CONFIGURED VEHICLE

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1. ABSTRACT

The paper starts with design considerations and a brief description of the CCV-F104 project. A detailed part is devoted to the strapdown subsystem design consideration and realisation. The hardware and software mechanization of the integrated guidance and control system of the CCV-F104-G is explained with special focussing on the strapdown part. This includes the solution to the redundancy problem.

Finally the next feasible steps in system improvement and minimization of the inertial part are outlined.

2. DATA REQUIREMENTS FOR GUIDANCE AND CONTROL

The basic signals needed to augment the aircraft are rates. However, if the requirements are tougher, angle of attack and sideslip angle as well as attitudes and flight path angle are requested.

These requirements increase, as more as the aircraft performance requirements are increasing, leading to the Control Configured Vehicle (CCV) concepts. Since navigation and guidance and control are using the same information the natural question arises, if their needs can not be satisfied using a common signal source.

They obviously can as the following table shows.

FUNCTION	CONVENTIONAL MECHANIZATION	INTEGRATED MECHANIZATION
NAVIGATION	Gimbal INS	S/D INS
Position	" "	" "
Velocity	" "	" "
Attitude	" "	" "
CONTROL		
Attitude	Gimbal INS or AHRS	S/D INS
Att. Rate	Rate Gyros	" "
Body-Axis Acceleration	Accelerometers*	" "
Altitude	Air Data Sensors	Air Data Sensors
Angle of Attack	Incidence Sensors	Derived from S/D
Angle of Sideslip	" "	" " " **
Airspeed	Air Data Sensors	Air Data Sensors
GUIDANCE	(Required data available from above items)	

* Or computationally derived from Gimbal INS

** Feasible 'practicality' yet to be proven

Figure 1. Information requirements for navigation, guidance, and control.

As the above table expresses, there are two candidates for the heart of the sensor system

Gimbaled platforms plus Rate Sensors plus eventually Accelerometers	vs	Strapdown Navigation Systems

Increasing operational requirements lead to redundant navigation systems, a fact which is of great importance to a common signal source solution. The stabilization and control system of a modern combat airplane is governed by redundancy requirements. Cost effectiveness considerations formerly did not permit the use of redundant navigation systems in fighter aircraft. Todays strapdown development is strongly pointing towards a redundant common sensor solution which still is cost effective.

3. WHY STRAPDOWN?

It is apparent from the preceding discussion that a strapdown system is an ideal flight reference sensor. A strapdown system operated in conjunction with appropriate air data sensors, provides all information required for vehicle guidance, navigation, and control. Recent studies conducted by Boeing [1] have concluded that an integrated strapdown/air data flight reference system can be cost effective even if navigation functions are not required. On this basis strapdown provides high accuracy inertial navigation at no additional cost.

Despite the far greater utility of a strapdown mechanization there is a natural tendency to evaluate its merits by comparison with a conventional gimbal inertial navigation system.

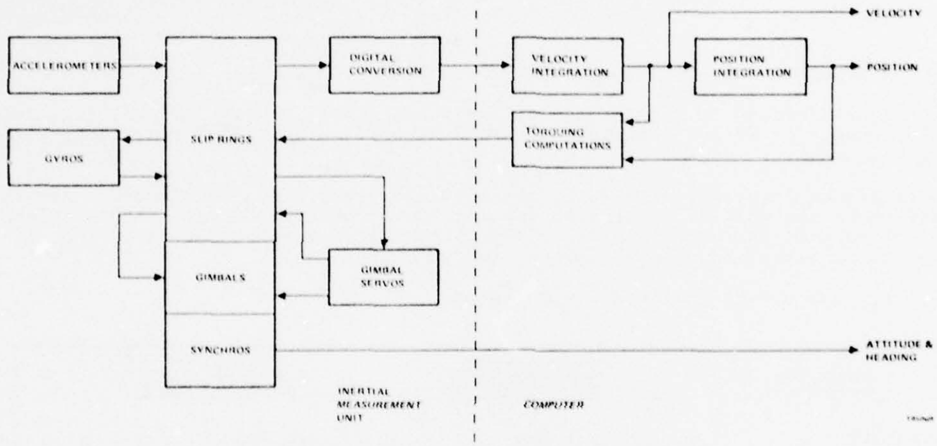

Figure 2. Simplified block diagram of a gimbaled platform

Figure 2 shows a highly simplified block diagram of a gimbal inertial system. The gyros, gimbal servos, and gimbals hold the accelerometer triad aligned with the reference coordinate system. The gyros are physically torqued by the computer in order to maintain this alignment. The primary computational functions are the integration of acceleration to obtain velocity, the integration of velocity to obtain position, and the computation of the required gyro torquing signals.

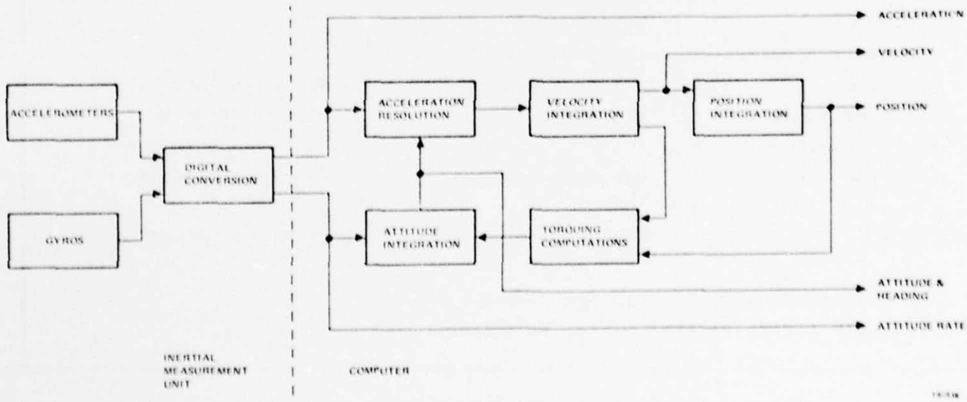

Figure 3. Simplified block diagram of a strapdown system.

The corresponding block diagram of a strapdown system is shown in Figure 3. The added computational functions involve the integration of measured gyro rates to obtain attitude, the resolution of measured accelerations onto the reference coordinate frame, and additional sensor error compensation which is required as a result of the demanding strapdown operating environment. In effect, these additional computations replace the precision electromechanical devices associated with the gimbal structure and its control.

reducing the computations and data entry (workload) requirements that are imposed by present data systems. Operating without benefit of data linked information, the aircraft can respond in minimum time and with minimum work effort, to radio contacts with controllers.

A comparison of several important gimbal and strapdown characteristics is shown in Figure 4.

ITEM	GIMBAL SYSTEM	STRAPDOWN SYSTEM
RELIABILITY	LOWER: MORE TOTAL PARTS	HIGHER: FEWER TOTAL PARTS
COST OF OWNERSHIP	HIGHER: LOWER RELIABILITY, MORE DIFFICULT TO MAINTAIN	LOWER: HIGHER RELIABILITY, EASIER TO MAINTAIN
IMU/COMPUTER INTERFACE	TWO-WAY DATA TRANSFER	ONE-WAY DATA TRANSFER
ATTITUDE ACCURACY	LOWER: CONTAMINATED BY SYNCHRO ERRORS	HIGHER: DIRECT DIGITAL ATTITUDE DATA AVAILABLE
ATTITUDE RATE DATA	NOT AVAILABLE	DIRECTLY AVAILABLE
BODY-AXIS ACCELERATION	NOT AVAILABLE	DIRECTLY AVAILABLE
ATTITUDE OPERATING LIMITS	SPECIAL PROVISIONS TO AVOID "GIMBAL LOCK"	NO SINGULARITIES

Figure 4. Characteristics comparison of gimbaled system vs strapdown systems

4. AN EXPERIMENTAL IMPLEMENTATION

4.1 Project description

In 1974 MBB received a contract from the German Ministry of Defence (GMOD) to design, develop and flight test a Control Configured Vehicle using an F-104-G as a test bed.

Since this experimental aircraft offered a good possibility to implement ohter than only the basic CCV-technology, it was decided to develop a full digital redundant integrated guidance and control system for the CCV-F104-G.

Strapdown sensors have been selected to form the heart of the CCV-sensor-system.

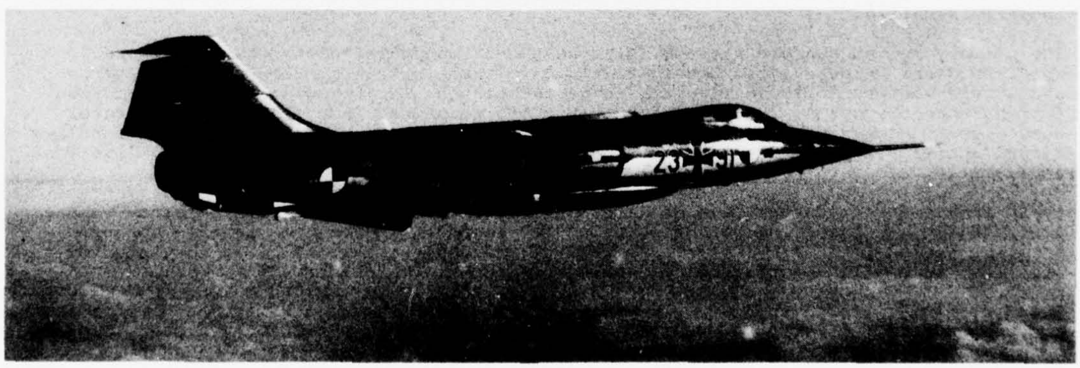

Figure 5. CCV-F104-G in flight

4.2 CCV-F104-G Guidance and Control System Description

The CCV-F104-G integrated guidance and control system provides, within a set of four redundant computers:

- Stabilization and Control
- Autopilot
- Air-Data Computation
- Strapdown Navigation
- Auto Navigation
- Redundancy Management
- Preflight Checkout

- TIME COORDINATE (DESIRED TIME-OF-ARRIVALS)
- TRACK ANGLE
- AIRSPEED
- VERTICAL FLIGHT PATH ANGLE
- TURN RADIUS
- WIND VECTOR

4.2.1 Functional description

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Using configuration measures the airplane will be destabilized up to -20% mean wing chord, a fact which will not be addressed here. However, it is very important, that the decision for the FBW system to be quad redundant was not influenced at all by the fact that the air plane is unstable. Only a (fail-op)² requirement and the decision for no mechanical back up system lead to this redundancy degree.

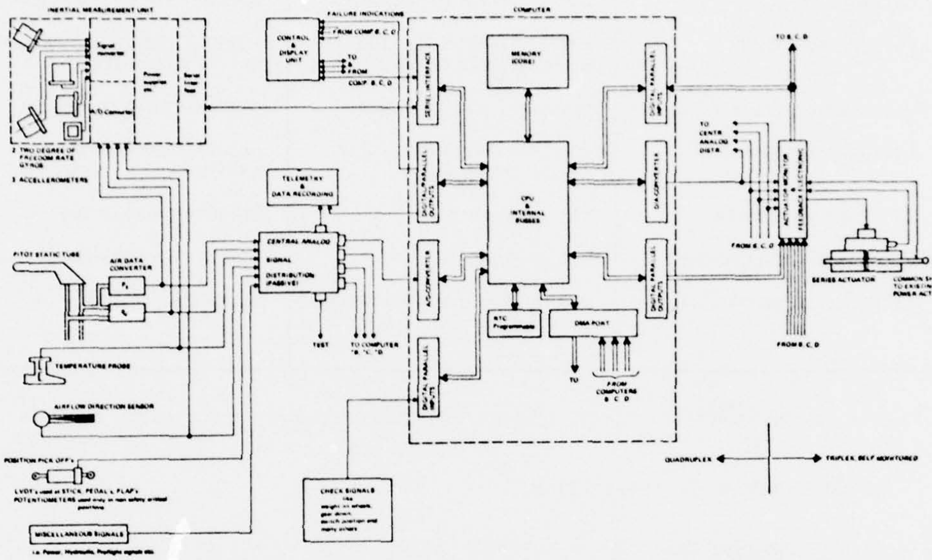

Figure 6. CCV-F104-G guidance and control system (one channel).

4.2.2 Subsystem Description

The strapdown subsystem consists of two basic units, a TDS-3D inertial measurement unit (IMU) and a TDY-43 digital computer. A simplified block diagram of the subsystem is shown in Figure 8.

The primary system sensors are two Teledyne SDG-5 two-degree-of-freedom strapdown gyroscopes and three Systron Donner Model 4841 linear accelerometers. Both the gyros and accelerometers employ analog rebalance techniques. The torquing currents required to rebalance the gyros, which are directly proportional to the vehicle angular rates about the sensing axes, provide the basic system angular motion measurements. Translational motion measurements are providing as voltages directly proportional to linear acceleration along the accelerometer sensing axes.

These analog sensor outputs are converted to digital format using high accuracy voltage-to-frequency (V/F) converters and buffered for transmission to the digital computer. Auxiliary data, including sensor temperatures and self-test signals, is converted by a separate, multiplexed, whole-number analog-to-digital converter.

Measurement data received by the computer is first compensated for various error effects. The compensated angular rate measurements are used to compute vehicle attitude. Compensated acceleration data, after appropriate coordinate transformation, is used to compute navigation data. Angular rate, acceleration, attitude and navigation parameters are then used as the data base for implementation of guidance and control functions.

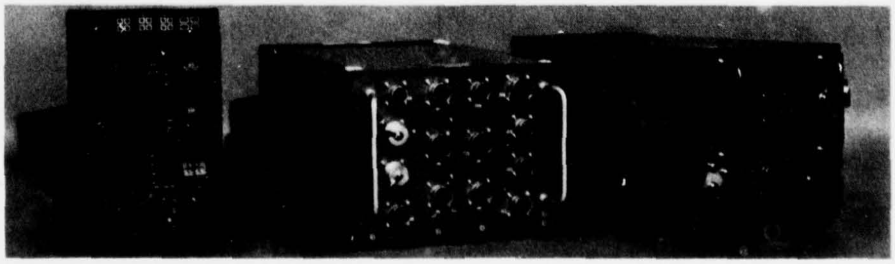

Figure 7. Control and display unit, computer and inertial measurement unit of the CCV-F104.

4.4 Vertical Path Generator

For an altitude change between point pairs the vertical path generator uses either one flight path angle (FPA) for the entire path between the two points, when a flight path angle has not been specified, or two FPAs consisting of a segment with a specified FPA and

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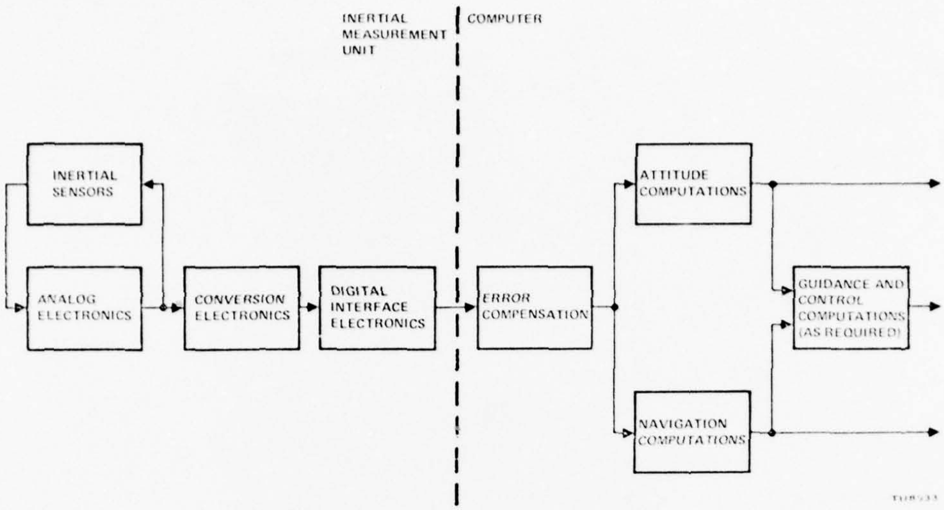

Figure 8. Simplified block diagram of the CCV-F104 strapdown system

4.2.2.1 Inertial Sensors

The inertial sensor package includes two two-degree-of-freedom gyroscopes and three linear accelerometers arranged on a single instrument mount. A photograph of this package is shown in Figure 9. The accelerometers are mutually orthogonal and mounted nominally along the primary axes of the vehicle. The gyros are mounted in a skewed arrangement with spin axes nominally 30° from the vehicle roll axes. This skewing is provided to reduce power consumption during high roll rate maneuvers.

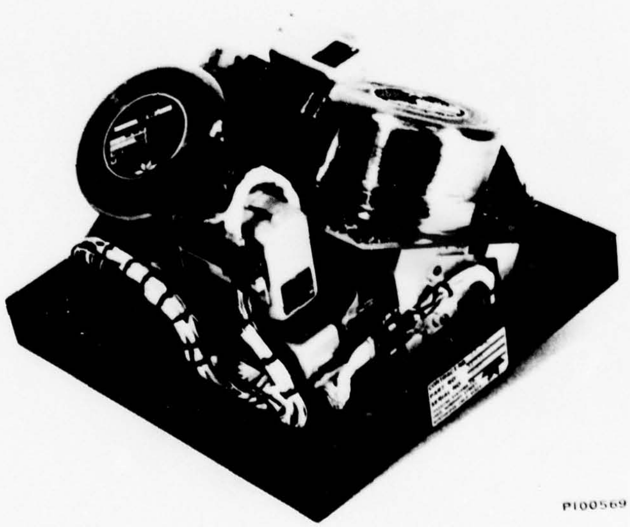

Figure 9. Sensor mount

The gyroscope which is employed is the Teledyne SDG-5. This gyro is a dry tuned instrument specifically designed for strapdown applications. Specified bias stability for the gyro is 0.01°/hr, with random drift of 0.001°/hr. Maximum input rates are specified at 400°/sec, providing a dynamic range in excess of 10^9:1. Development of the SDG gyro was initiated in 1968 and it has been produced since 1971.

the aircraft will not meet the assigned TOA. This information is displayed on the Situation Display for the pilot, in case he wants to modify the 3-D trajectory or specified speeds to achieve the TOA.

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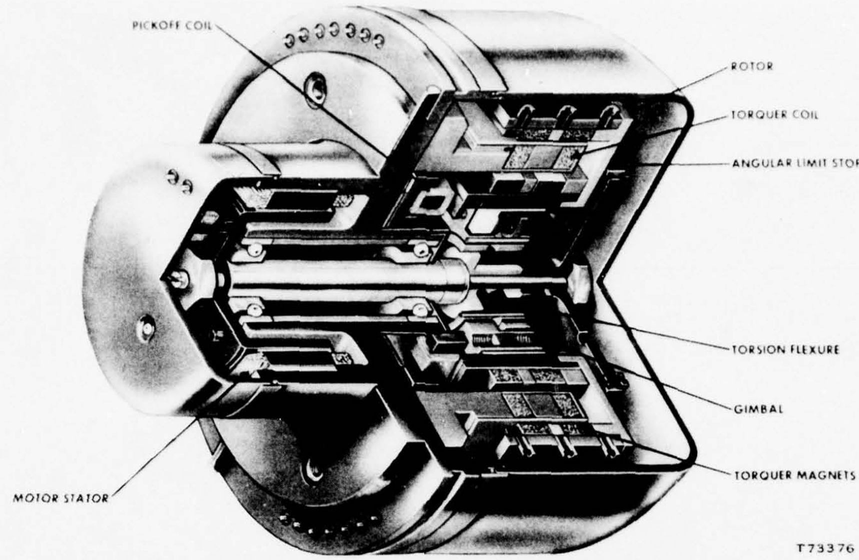

Figure 10. SGD-5 Dry tuned gyro

The accelerometer which is used is the Systron-Donner Model 4841. This instrument is characterized by bias stabilities of 50 μg. Approximately 3000 of these units have been produced and used in a variety of applications. A photograph of the Model 4841 accelerometer is shown in Figure 11.

Figure 11. Model 4841 accelerometer

4.2.2.2 IMU Electronics

The IMU electronics serve three primary functions. The analog electronics provide control of the inertial sensors. The conversion electronics convert the basic analog measurement data to digital format. The digital electronics provide IMU timing functions, buffer the converted data and interface with the digital computer.

A simplified block diagram of the sensors and analog electronics is shown in Figure 12.

path angle, or acceleration.

The reference state and control law have been structured so that each command will handle a single dimension of the tracking problem, with proportional plus rate feedback provided in each command. The proportional feedback nulls out the displacement error and rate feedback provides path damping. In the IFTC program where the control law is implemented in a hybrid simulator, aircraft stability is augmented with roll rate

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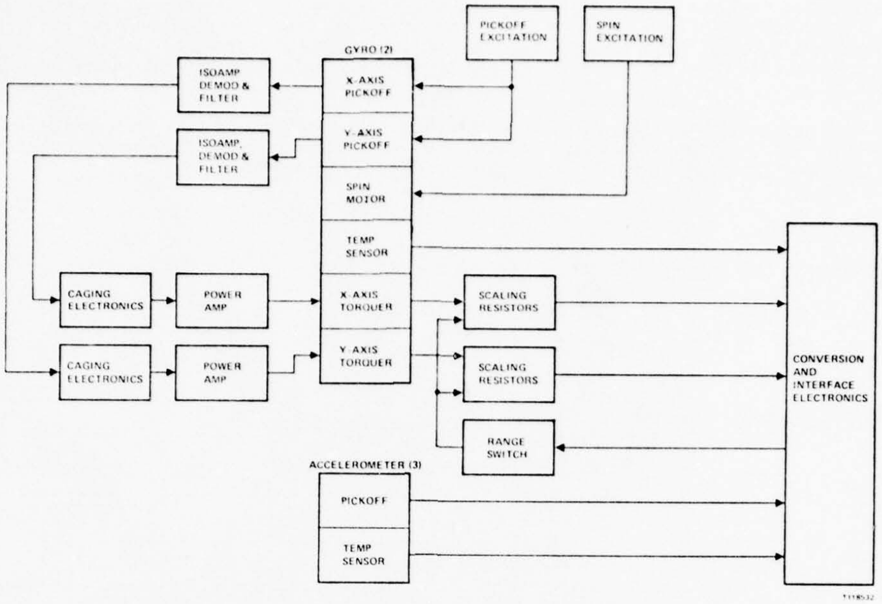

Figure 12. IMU sensor and electronic block diagram

The gyro spin motor is driven by 3 phase 400 Hz excitation. Pickoff excitation is 48 KHz. Pickoff signals are demodulated, filtered, and used to derive the basic gyro caging signals, which utilize both cross- and direct-axis pickoff information. The caging outputs are power amplified and used to torque the gyro rotor to null. The torquing current is directly proportional to input angular rate and provides the basic gyro measurement information.

Accelerometer rebalance electronics are integral to the sensor. The accelerometer output is a voltage which is directly proportional to input acceleration.

Temperature sensors are included internal to both the gyros and accelerometers. These provide data which is required for error compensation.

A simplified block diagram of the digital and conversion electronics is shown in Figure 13.

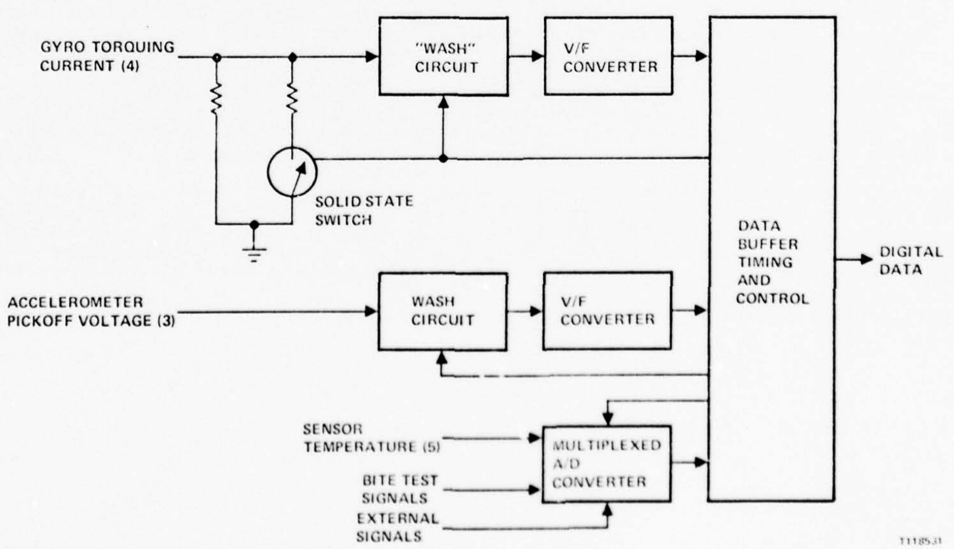

Figure 13. Conversion electronic block diagram

(with roll, pitch, and throttle commands), situation (horizontal and vertical) information, and system alphanumeric information. The concept of computer/pilot interaction was developed to a higher level during the fighter demonstration study by providing mission oriented mode controls and expanded man/machine interaction (through digital processing) for mission and data management.

The gyro torquing current is used to develop a voltage across precision resistors. A solid state switch is employed to provide two ranges of operation, thus extending the dynamic range of the electronics to that provided by the gyroscopes. A "wash", or polarity reversal, technique is incorporated to effectively eliminate contamination of the data by bias voltages.

The acceleration conversion channels are identical to those used for gyro conversion except that no range switching is required. A separate multiplexed whole number analog-to-digital converter is used to convert auxiliary data to digital format.

4.2.2.3 Software

A computational block diagram is shown in Figure 14.

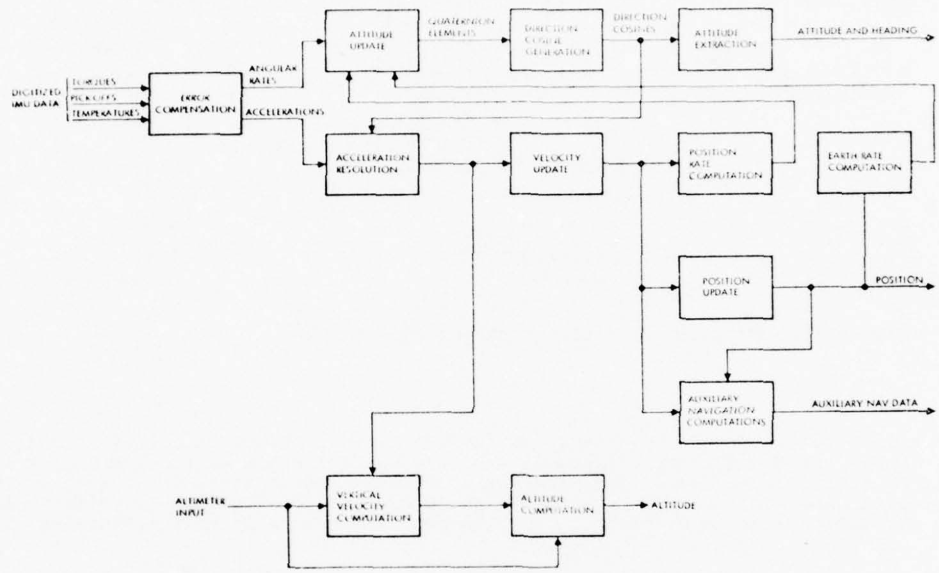

Figure 14. Strapdown computational block digramm.

Sensor measurements are first compensated for error effects. Gyro compensation includes corrections for bias, scale factor, misalignment, direct and quadrature mass unbalance, anisoelasticity and several "dynamic" error sources. Accelerometer data is compensated for bias, scale factor and misalignment. Since the sensors are not temperature controlled, compensation is also included for thermal variations in certain error parameters.

Compensated angular rate data is used to update the computed attitude. The attitude reference is maintained as a quaternion and is updated using a 4th order Runge-Kutta algorithm at a 50 Hz rate. The attitude quaternion is used to compute the attitude direction cosines from which vehicle pitch, roll, and heading are computed.

Compensated acceleration data is resolved through attitude into local-level, wander azimuth coordinates and integrated to obtain velocity and position. The vertical channel is damped by air data-derived altitude information. Earth rate and craft rate correction terms are computed and applied to the attitude update computations. Acceleration resolution is performed at a 50 Hz rate. All navigation computations are performed at 16 2/3 Hz.

A computational flow diagramm is shown in Figure 15.

keys, and hand-operated crosshair (on the situation display), and computer automation, see Figure 11. Decision trees for the control/display part of the computer program have been developed that establish a reasonable, simple and logical man-machine interaction procedure. By anticipating, in certain situations, what display format the pilot will need for the next interaction step, computer automation is being used to minimize the number of detailed instructions (and key actuations) needed to control a system with this flexibility and capability.

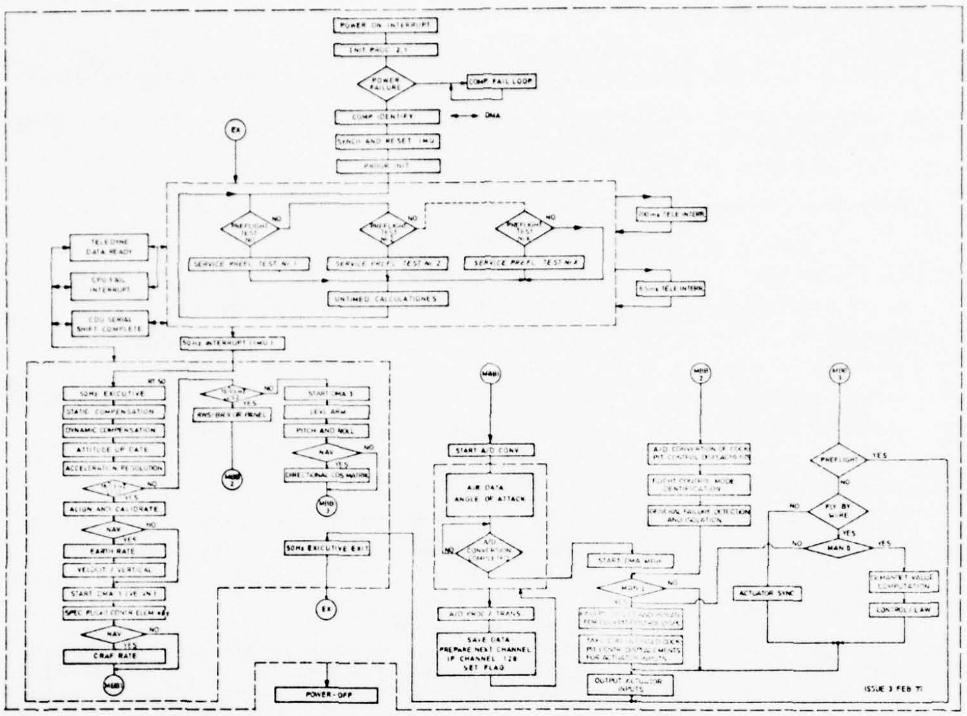

Figure 15. CCV program flow.

Timing and memory requirements for the strapdown mechanization are summarized in Figure 16.

FUNCTION	MEMORY (16 BIT WORDS)	PERCENT OF AVAILABLE TIME (TDY-43 COMPUTER)
EXECUTIVE & TEST	524	1.43
ERROR COMPENSATION	794	13.43
ATTITUDE UPDATE & ACCELERATION RESOLUTION	911	16.75
ALIGNMENT & CALIBRATION	911	N/A
NAVIGATION	809	2.09
FLIGHT REFERENCE PARAMETERS AND AUTONAV	524	2.92
ARITHMETIC SUBROUTINES	841	N/A
SUBTOTAL	5373	38.21%
CONTROL & DISPLAY PANEL	1983	N/A
TOTAL	7356	38.21%

Figure 16. Strapdown timing and memory requirements

- Redirect to new second target with specified arrival time
- Redirect to new egress route with specified arrival time
- Set up and rendezvous with new tanker for refuel

4.2.2.4 Strapdown System Outputs

The raw data are received from the IMU and compensated. The subsequent intergration utilizes a fourth-order Runge Kutta algorithm for the quaternion representation of attitude.

The following elements of the Inertial State Vector are available to the user within the computer, and most of them are also available for readout via a CDU.

a_x	-	body accelerations	p	-	roll rate
a_y	-	body accelerations	q	-	pitch rate
a_z	-	body accelerations	r	-	yaw rate
ϕ	-	roll attitude	u	-	velocity along x-body-axis
θ	-	pitch attitude	v	-	velocity along y-body-axis
ψ	-	heading	w	-	velocity along z-body-axis
V	-	absolute velocity	$\dot{h}$	-	vertical geodetic velocity
V_{GND}	-	ground speed	h	-	altitude (supported by air data)
α	-	angle of attack	χ	-	ground-track angle
β^*	-	sideslip angle	ϕ	-	latitude
γ	-	flightpath angle	λ	-	longitude
RNG	-	range to destination	XTR	-	crosstrack error
BRG	-	bearing to destination	TTG	-	time to go to destination

It should be noted that:

(1) Since the attitude is also internally available as quaternions and the CCV-F104-G control system uses attitude as feedback (among other variables), the control system engineers confronted with the old attitude/singularity problem became accustomed to quaternions. Now they have converted their control laws such that quaternions are directly fed into the control laws.

(2) The state vector contains the angle of attack (α) and sideslip angle (β^*). It should be noticed, that α does not contain the gust term α_w and β^* contains also the drift angle δ.

The direct use of α_{IN} for stabilization (feedback) will be tested in the CCV-F104. No final assertion can be made about an appropriate filter etc. for the elimination of δ in the β_{IN} signal. Investigations and tests are under way.

4.2.2.5 Computer

The computer is a 16 Bit Teledyne TDY-43. This machine operates at a 3.36 MHz clock rate, executing short instructions in 2.38 μsec and multiply instructions in 5.36 μsec. Throughput, for a basic 80% ADD/20% multiply instruction mix is approximately 335 thousand operations per second (KOPS).

Each computer utilizes a 16 K by 16 Bit core memory, and is plug-in expandable to 32 K x 16 Bit capability. An additional 2 K x 16 Bit high speed semiconductor RAM memory is provided for serial DMA accumulation of IMU data and other high speed processing functions.

Three separate bidirectional DMA channels are provided for communication with companion computers in the quad-redundant configuration. DMA operations are performed on a cycle-stealing basis with I/O having priority over the CPU.

Digital I/O capability is provided for communication with the IMU and the Control and Display Unit. The digital I/O section also provides for 6 channels of 16 Bit parallel digital inputs and 6 channels of 16 Bit parallel outputs.

The analog I/O section provides capability for 128 channels of analog input and 8 channels of analog output. The extensive analog input capacity provides for "wrap-around" of the analog outputs of all four computers for redundancy management purposes.

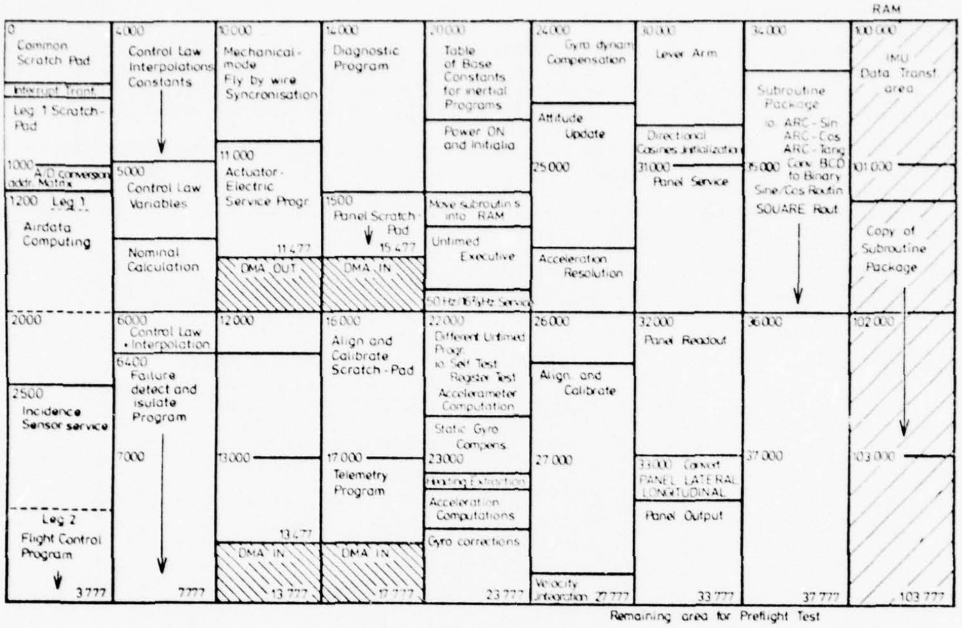

Figure 17. Computer memory occupation: 16 k + 2 k RAM.

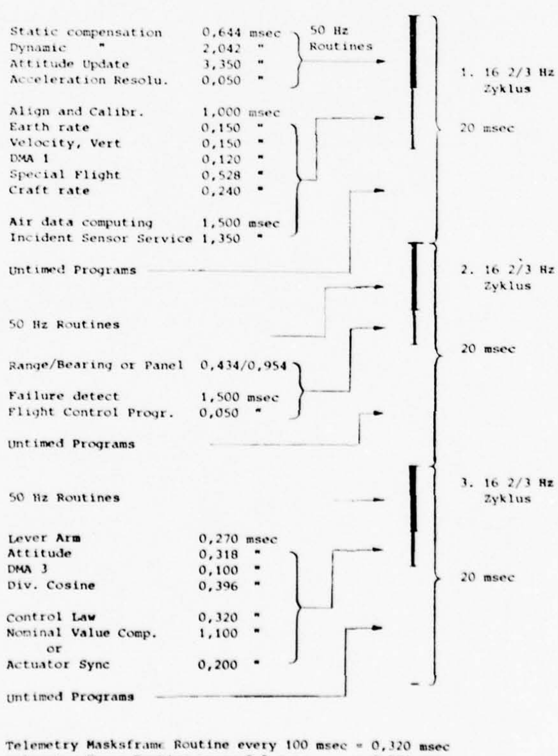

Figure 18. Time chart of CCV programm.

Figure 5. Parameters Calculated by Horizontal Path Generator.

Figure 6. Speed/Time Computational Flow Diagram.

3.2.3 Flight Safety Considerations

9-12 Any equipment used in a flight-safety critical position has to satisfy two requirements. The first is the reliability figure, which is hard to get and harder to believe in case one is working with prototypes. The second, which we are concentrating on, is the operational requirement. In case of the CCV-F104 program, double fail-op requirement was assumed.

The solution is strictly a majority-decision software logic within the computers, which act as central voters and monitors.

Since the number of new technologies involved already have been enough, principles such as skewing or self-monitoring have not yet been incorporated.

In order to accomplish the failure detection and isolation for all data, the inertial and non-inertial data have to be exchanged between computers, done here by DMA data exchange (see Figure 19). This is a very fast and software-saving method.

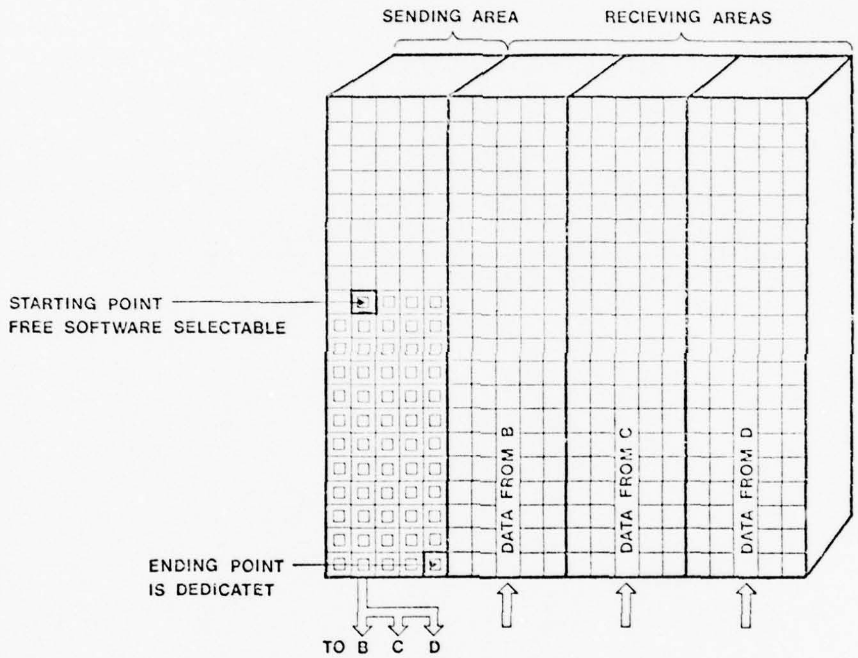

Figure 19. Principle of DMA data exchange.

Because of the DMA data-exchange feature, each computer possesses all variables stored in its own memory and in the memories of the other three computers. Thus it can vote.

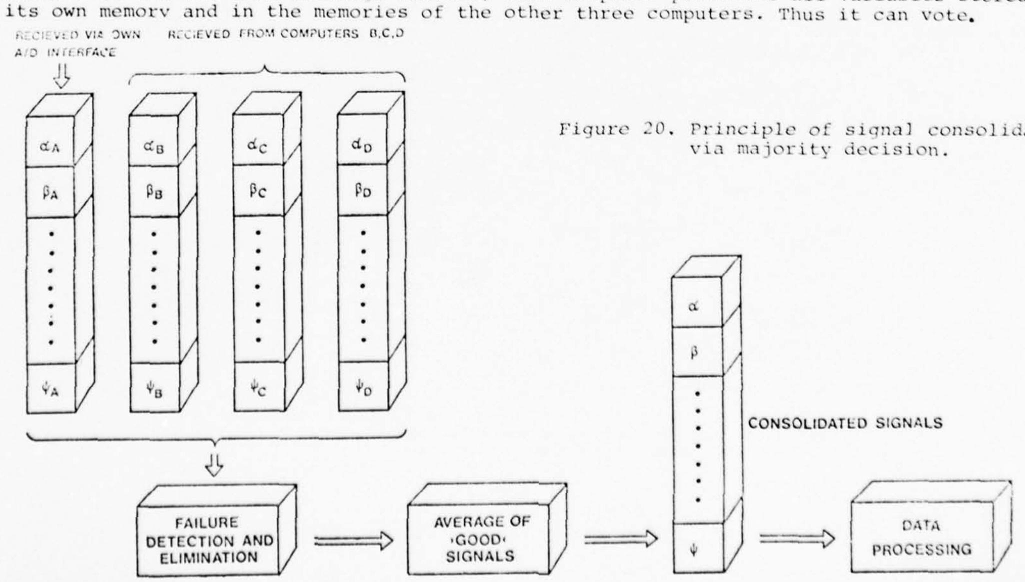

Figure 20. Principle of signal consolidation via majority decision.

As mentioned before, the computers run identical but nonsynchronized software. This feature, which is imposed by computer hardware, has resulted in an undesired effect. In general, it forces one to use higher failure thresholds. In case of the strapdown signals it did not allow us to consolidate the error-compensated raw signals as initially intended. Since the asynchronism between computers may be as big as 60 ms with a maximum angular acceleration of 1000°/s², rate thresholds of 60°/s arising out of this asynchronism are not tolerable. In practice, failure detection, therefore, is done based upon attitude and velocities. The asynchronism problem is not considered to be a major problem; however, any future system should run soft-synchronized to ease the programmers' work.

5. PROJECT STATUS

At the time when this paper was written, the CCV-F104 had completed a first test-phase, which was a pure calibration phase for the aerodynamic sensors. A total of 11 flights were performed. One IMU (part of the flights two IMUs) was installed, but was not subjected to detailed tests. However, the navigation accuracy observed was between 1 and 3 nmi/h CEP.

At the time of writing, phase II of the flight-tests with the fully equipped plane had just commenced. Twelve flights have been performed. These and a few follow-on flights are purely dedicated to aircraft open-loop performance evaluation since some external modifications may affect the aircraft aerodynamics.

More detailed information will be given at another occasion.

6. DESIGN IMPROVEMENT CONSIDERATIONS

6.1 Introduction

The guidance and control system of the CCV-F104-G is presenting a milestone with its total digital FBW, its functional integration and its strapdown system. However, some techniques known already at the beginning of the project have been excluded, because one wanted to limit the risk. Other new techniques have evolved meanwhile and a number of lessons have been learned already during the first flights and the ground integration. So has for instance skewing technique been applied only to aerodynamic sensors in the CCV-F104-G and self monitoring technique only to the actuating system.

As an outlook to future projects some of this aspects shall be presented in the following.

6.2 Sensor Configuration

The orientation of sensors plays a major role in the reduction of hardware without aggrivating the performance and operational requirements.

Without discussing the particular requirements a summary of some of today aircraft are listed in Figure 21.

AIRCRAFT TYPE	RATES (FLIGHT CONTROL)	INS, HAS/VG
727	5 RG 3 ACCEL	2 VG
747	8 RG 3 ACCEL (8 RG 7 ACCEL)	2 INS (3 INS)
7X7	DUAL REDUNDANT (PROVIDED BY INS)	2 INS (STRAPDOWN)
YC-14	9 RG 6 ACCEL	1 INS 2 VG
F-14	7 RG 3 ACCEL	1 INS 1 HAS
F-15	6 RG 4 ACCEL	1 INS 1 HAS
F-16	12 RG 8 ACCEL	1 INS 1 HAS

Figure 21. Inertial sensors in some of todays airplanes (Reference [I])

Figure 11. Interactive Control/Display.

Figure 21 may be used as a starting point for cost effectiveness considerations with the following sensor configurations.

6.2.1 Full Skewing

The subject of optimal sensor configurations for redundant strapdown systems has received considerable attention in recent years. The selection of an optimal configuration depends upon the level of redundancy to be achieved, the type of sensor and its error statistics among other considerations.

Fail-Op/Fail-Op redundacy, i.e. the ability to survive any two failures, requires a minimum of 6 single-degree-of-freedom (SDF) sensors or 4 two-degree-of-freedom (TDF) sensors. Under fairly general assumptions concerning error statistics the orientation of the sensors for optimum accuracy is symmetric. For SDF sensors this is equivalent to arranging the instruments with sensing axes normal to the six non-parallel faces of a regular dodecahedron, as shown in Figure 22.

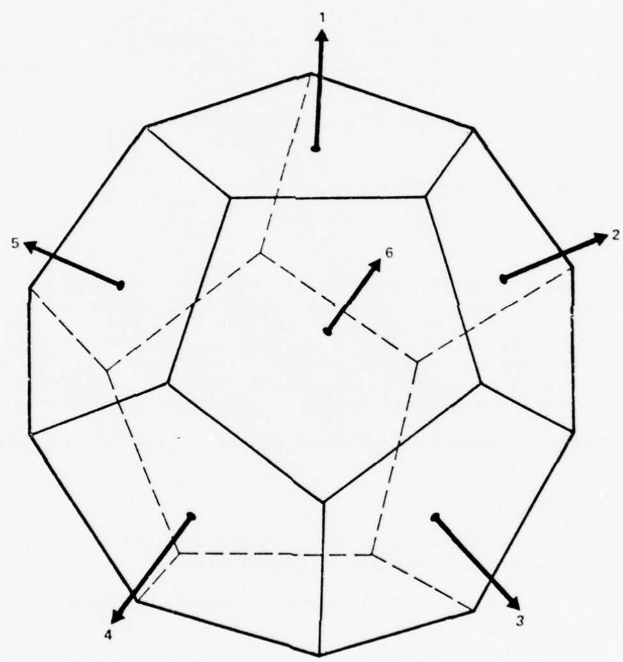

Figure 22. Dodecahedron orientation of six SDF-sensors.

The symmetric configuration for TDF gyros is equivalent to arranging the sensors with spin axes normal to the faces of a regular tetrahedron or semi-octahedron. The semi-octahedron arrangement is indicated in Figure 23 (Orientation of the sensing axes about the spin axes is arbitrary from the standpoint of accuracy, but does affect the ability to detect and isolate failures.) A photograph of a semi-octahedron instrument package produced by Teledyne is shown in Figure 24.

As the above table expresses, there are two candidates for the heart of the sensor system

Gimbaled platforms plus Rate Sensors plus eventually Accelerometers	vs	Strapdown Navigation Systems

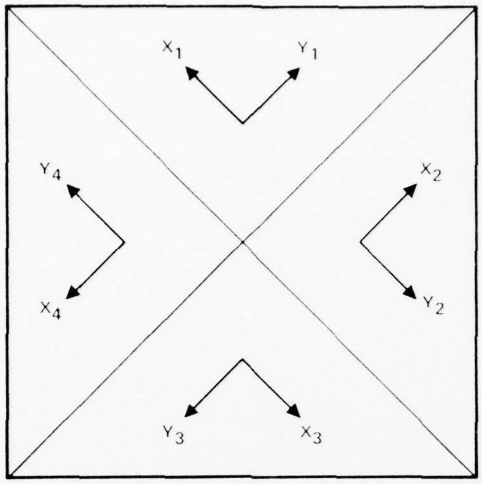

Figure 23. Semi-octahedron orientation.

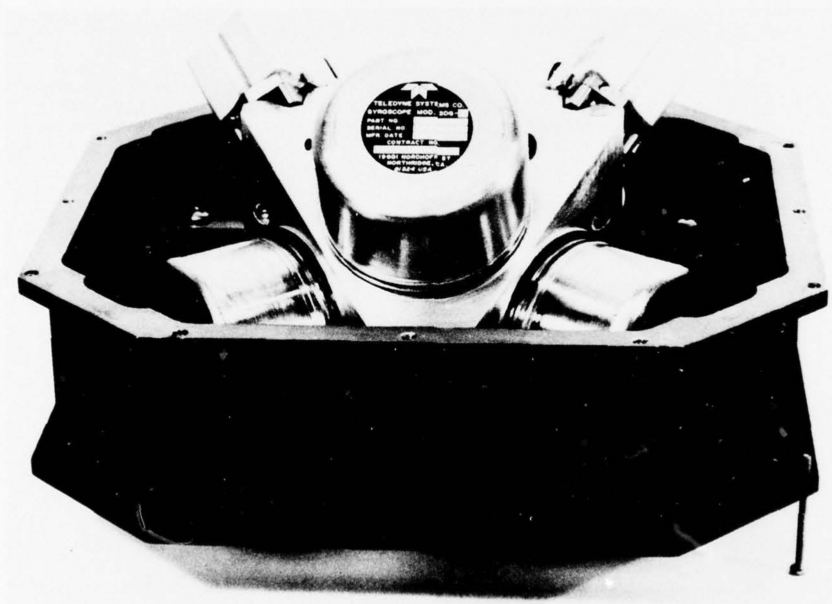

Figure 24. Semi-octahedron Instrument.

6.2.2 Semi-skewing

An alternative to the minimal sensor redundancy approaches is the "brick wall" approach, which implements redundancy at the IMU level rather than the sensor level. This approach requires more sensors but provides certain offsetting advantages, some of which are discussed in a later section of this paper.

The principal is shown in the following Figure 25.

9-16

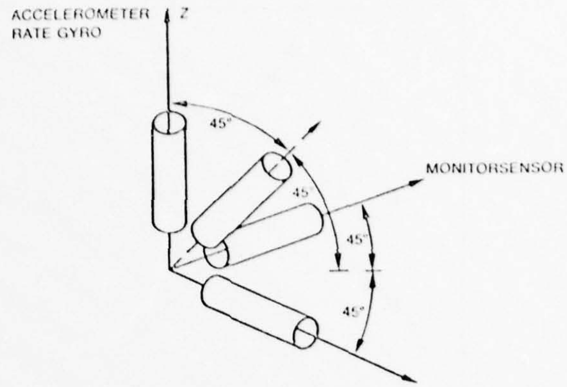

Figure 25. Semi-skewed arrangement principle of inertial sensors

The monitorsensor is used to derive a signal which represents the geometric sum of the three orthogonal sensor signals. This method provides a failure status for the whole block but does not allow a failure localization or isolation within the block.

If the sensor is a TDF sensor, the effectiveness of such an approach is far better, because in order to get three orthogonal signals two sensors would be needed, providing four signals.

A useful TDF sensor configuration utilizing this fourth signal for failure detection is indicated in Figure 26. This "semi-skewed" configuration provides failure detection capability directly by comparison of the four axis measurements. A corresponding arrangement for SDF sensors would utilize three orthogonal sensors with a fourth sensor at equal angles to the three as Figure 25 depicts.

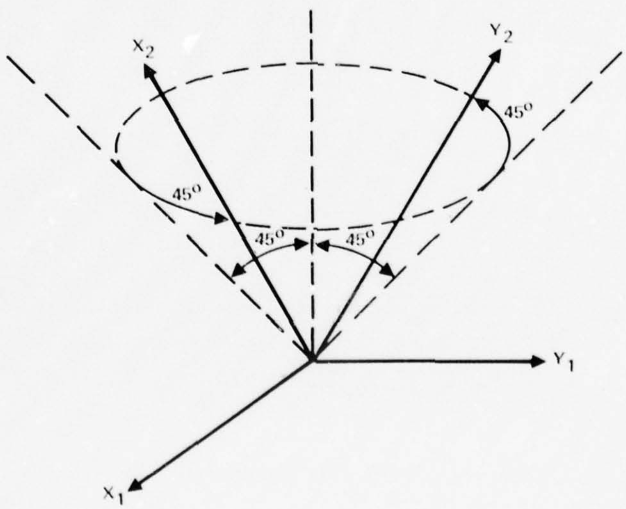

Figure 26. Semi-skewed arrangement of two TDF-sensors.

6.3 Sensor Type

An important consideration in implementing a redundant inertial system is the number of sensors which are employed. The number of sensors utilized, together with their supporting electronics, obviously has a direct effect on system acquixition costs. It also has significant reliability and maintainability implications, since increased parts count implies reduced reliability.

The dodecahedron and tetrahedron (or semi-octahedron) sensor configurations provide minimum sensor count for Fail-Op/Fail-Op redundancy. A total of 12 sensors is required if SDF instruments are employed (6 gyros and 6 accelerometers). Using TDF gyros, the count is reduced to 10 since only four gyros are required.

The potential for a dramatic reduction in sensor count exists in a new sensor currently being developed by Teledyne for the U.S. Air Force. This sensor, a derivative of the SDG-5 gyro, is the Spin Coupled Accelerometer Gyro (SCAG). The SCAG provides two axes of angular rate (gyro) information as well as two axes of linear acceleration information in a single instrument which is smaller than the SDG-5 gyro. This unique capability permits the implementation of a full Fail-Op/Fail-Op inertial sensor package with a total of only four SCAG sensors, as no accelerometers are required.

The following Figure 27 compares different approaches and lists some advantages and disadvantages.

Fail-Op² Requirement	CCV-F104-G	FULL-SKEWED SYSTEM	SEMI-SKEWED SYSTEM
SDF-Gyros	N.A.	6	3 x 4 = 12
SDF-Accelerometers		6	3 x 4 = 12
TDF-Gyros	8	4	3 x 2 = 6
TDF-Accelerometers	12	6	3 x 4 = 12
SCAGS	N.A.	4	3 x 2 = 6
ADVANTAGES	REFERENCE ONLY	MINIMIZES HARDWARE SAVES SPACE REDUCES POWER CONSUMPTION	EASES REDUNDANCY MANAGEMENT ALLOWES DISTRIBUTION THROUGHOUT THE A/C SIMPLER POWERSUPPLY EASY TO REPLACE
DISADVANTAGES		POWER REDUNDANCY DIFFICULT SYSTEM REPLACEMENT IN CASE OF FAILURE SENSITIVITY REDUCTION IF SENSOR FAILES	USES MORE SENSORS USES MORE SPACE

Figure 27. Full-skewed sensors vs semi-skewed sensors.

Figure 27 is selfexplaining, but going back to Figure 21 it is clear, that strapdown are strong contenders for a central inertial reference system especially in combat aircrfaft.

7. CONCLUSIONS

Based upon our experience with a quadrupally redundant strapdown system, and on the growth potential observed here and on other occasions, we find strapdown inertial technology to be an excellent innovative technique. Especially in redundant and integrated guidance and control systems strapdown sensors are the prime candidates for new projects.

Strapdown offers a significant reduction of hardware and cost and considerably reduces the effort in those applications where not only redundant rates and accelerations but also redundant altitudes and other higher information are needed.

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ACKNOWLEDGEMENT

The basic work described herein has been carried out under contract of the German Ministry of Defence (GMOD).

Program : KEL

Contract: CCV-F104-G

PRELIMINARY FEASIBILITY ASSESSMENT OF MULTI-FUNCTION INERTIAL REFERENCE ASSEMBLY (MIRA)

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SUMMARY

The multiplicity of inertial and air data sensors on advanced Air Force fighters and transport aircraft is contributing significantly to sharply increasing avionic costs. A potential solution is the development of a Multi-function Inertial Reference Assembly (MIRA) subsystem which satisfies all on-board inertial and air data reference data requirements for flight control, navigation and weapon delivery.

This paper discusses mission and performance goals established for MIRA feasibility studies covering flight control, navigation and weapon/cargo delivery as applied to the fighter aircraft (F-15) and a transport aircraft. The relationship between the key technical issues of concern and the feasibility criteria and the methodology to perform the trade-offs which impact life cycle costs are described. Functional performance and reliability requirements are shown. Computational requirements for a representative MIRA system is summarized. Computer programs were used to evaluate time histories of sensor and system error propagation and to assess the impact on flight control system control laws as MIRA sensors are installed at various aircraft installation locations. The criteria defined to perform the preliminary feasibility assessment is discussed. Comparative studies of life cycle costs show a saving estimate in excess of 69 million dollars for MIRA application to a quantity of 144 fighters over a 15 year operational life. Cost savings for transport applications are qualitatively significant, particularly for the operations and support cost element. The results of ring laser gyro (RLG) and tuned rotor gyro (TRG) studies of performance and reliability improvements required are summarized. The laboratory demonstrations performed by three subcontractors with operating redundant equipment, which shows software capability to provide fault coverage, is discussed.

Technology projections indicate that by 1980 performance and producibility will be nearing maturity for sensors and microprocessors, and that software techniques will be developed which provide adequate fault coverage and redundancy management for both skewed and multi-unit sensor system architectures.

The results to date have shown that questions relating to key issues have been satisfactorily answered with the conclusion that the MIRA program should proceed into further detailed system configuration studies, which specifically address commonality, test and maintenance, standardization trade-offs, and the preparation of a technical exhibit for a selected best MIRA candidate configuration.

I. INTRODUCTION

Current military fighter and transport aircraft use avionic equipment which generally follows a federated or a consolidated system architecture. This has resulted in various inertial and air data functions being replicated and sensors being tailored and dedicated to specific subsystem tasks, for example, navigation: Inertial Navigation System (INS), Attitude and Heading Reference Systems (AHRS) and Air Data Computer; flight control: rate gyros, accelerometers and dynamic pressure sensors; weapon delivery: lead computing gyros and accelerometers.

With the progress being made with inertial sensors, such as strapped down tuned rotor gyros and ring laser gyros, and large scale integrated (LSI) circuit microprocessors, the question has been posed which asks, "Is it possible to configure a multi-function inertial reference assembly subsystem which (a) provides functional outputs adequate for navigation, flight control and weapon delivery, (b) achieves the required mission and safety-of-flight reliability, and (c) results in significant life cycle cost savings when compared with current approaches?"

In response to this question, the Air Force, under joint sponsorship of Flight Dynamics Laboratory, Avionics Laboratory, and Aeronautical Systems Division, created the MIRA program to determine the eventual payoff when MIRA is applied to advanced Air Force fighters, transports and remotely piloted vehicles (RPVs) of the 1980-1990 time period (Figure 1).

The specific objective and approach is shown in Figure 2. In order to implement the program, the Air Force partitioned the MIRA program into two phases: Phase I, Feasibility, which was awarded to McDonnell Douglas Corp. (MDC), St. Louis, Missouri, in June 1976 and will end in September 1978, and Phase II, Verification, which is currently scheduled to be started in Air Force fiscal year 1979 as shown in Figure 3. MIRA application would take place starting in the mid 1980s. MDC has put together a MIRA team for Phase I being led by McDonnell Aircraft (MCAIR) in St. Louis where fighter studies are also being concentrated, together with Douglas Aircraft (DAC) for transport studies and subcontractors

Honeywell, Minneapolis, Minnesota; Singer-Kearfott, Little Falls, New Jersey; and Litton, Woodland Hills, California, for support studies. Subcontractor studies emphasized strapped down system, ring laser gyro (RLG) and tuned rotor gyro (TRG) implementation technology, analytical trade-off studies and hardware demonstrations. The activities and studies described in the following sections of this paper were performed as part of Task 1, Phase 1. Task 1 was aimed at making a preliminary assessment of feasibility of the MIRA concept for both fighter and transport aircraft through analytical studies performing comparative analyses, and laboratory demonstrations with available hardware.

Detailed system configuration studies and the selection of a recommended MIRA system have been completed under Task 2. During Task 3 the MIRA technical exhibit will be written for use in the MIRA Phase 2 verification activity.

Figure 4 illustrates conceptually how the MIRA is envisioned to functionally replace the inertial and air data references on current aircraft. MIRA, as a subsystem which has adequate performance and reliability (including redundancy), feeds the using subsystems through the data bus network, MIL-STD-1553A.

Key issues to be resolved in determining the feasibility of the MIRA concept to effect cost savings are shown in Figure 5. Some multi-function requirements needing definition are: sensor accuracy required for long duration flights and the wide dynamic range for fighter application; and sensor location effects, which include the vibration effects on redundant and/or separate sensors when operated in a flexible airframe. Failure detection and isolation requirements place demands on microprocessors to perform within real time in order to keep the fault detection and isolation at high confidence levels, the nuisance alarms at low levels, and to accomplish the necessary redundancy management for system reconfiguration. Adequate reliability must be achievable in a practical sense and modularity, which will permit flexibility of application, is essential.

The MIRA road map of activity is shown in Figure 6. This paper will cover the sign posts leading up to crossing over the bridge. Because it is believed that the fighter aircraft application will impose the more demanding requirements of the MIRA, most of the ensuing discussion is addressed to fighter analysis. A MIRA that is feasible for the fighter should be capable of adaptation to meet the normal and unique performance requirements of a transport aircraft.

II. SYSTEM REQUIREMENTS

The latest Air Force operational fighter, F-15A, was selected as the principal data base source for comparative fighter studies, supplemented with F-4 and F-18 data (Figures 7 and 8). A representative aircraft was selected for comparative transport studies. Each aircraft has a high authority fail-safe control augmentation (CAS) flight control system. Inertial navigation is required on each aircraft. The F-15 specializes in air superiority weapon delivery and has excellent air-to-ground capability. The transport performs cargo and troop drop missions.

Missions were defined for fighters and transports which were representative of combat conditions. The fighter mission lasting about two hours involving high dynamics, and the transport mission being approximately 30 hours, involves numerous mission segments where landings are made and drops occur. Mission performance goals were defined to be 1 NM/hr for inertial navigation, 8 milliradian for air-to-surface weapon delivery, and Level 1, MIL-F-8785 flying qualities. Air-to-air weapon delivery performance is classified and for the purposes of this paper has not been included.

Output signal requirements were determined based upon allocations to functions for various current avionic equipments. Figure 9 lists some of the key parameters which were taken from a detailed list which showed a total of 117 outputs.

III. ANALYSIS

Analysis of computational requirements are summarized in Figure 10. A preliminary analysis of function commonality showed that F-15 air inlet controller requirements were so aircraft/engine oriented that the air inlet requirements should be removed from the main MIRA modules. It should be noted that the required processing speed (Kops) and refresh rate to implement MIRA would be decreased (approximately by a factor of four) for the transport, but that memory word storage requirements would stay about the same.

Flight control key issues, structural modes frequency and amplitude, gain and phase margins, fuselage stations for MIRA installation, and lever arm effects, were analyzed. Results, illustrated in Figure 11, showed that when sensors are combined the best location for flight control is in a zone around fuselage station (F.S.) 425 but that the sensor Line Replaceable Unit (LRU) could be located as far forward as F.S. 225 before the CAS would go unstable without change in control laws. The analysis also showed the methodology of how control laws changes could be made to permit LRU location to be moved up to the most available forward location (just aft of the radar) and still provide Level 1 handling qualities.

Figure 12 reports the analysis results for the fighter hardware Mean Time Between Failures (MTBF) reliability apportionment. The MTBF is based upon parts count. For comparative purposes, the safety-of-flight reliability is taken care of by having mechanical back-up for flight control. Figure 13 shows transport reliability results.

The 150 hour MTBF is a minimum requirement and should not be interpreted as being the actual MTBF. In reality the actual MTBF should be significantly higher. The transport mission reliability of 0.99929 is required for each of three different types of missions (training, deployment augmentation, intratheater support) defined for reliability purposes.

IV. FEASIBILITY CRITERIA

All of the previous discussion has been oriented to establishing the MIRA requirements and performing associated analysis in preparation for being able to respond to the preliminary feasibility assessment criteria which consists of the six categories, all interrelated, shown in Figure 14.

Producibility data on RLG and TRG sensors, which covered fabrication, assembly and test, and the methodology associated with each sensor type, was supplied by each equipment subcontractor. Because producibility is related so closely with proprietary data, the review of the data was done at each subcontractor facility with only MDC and government personnel in attendance. The conclusion being that TRG technology is very much like the present gimballed TRG technology and therefore, represents minimum risk. However, more experience is required with the RLG technology, because there are still size and performance developments in work. Within one to three years the RLGs should have proven producibility based upon current development trends.

The cost savings potential was handled on a total life cycle cost (LCC) basis (Figure 15). MCAIR, DAC and each subcontractor postulated a system and a proposed mechanization based upon the MIRA performance goals and outputs defined by MDC documents. Equipment costs were supplied to MDC which, in turn, were input into a LCC model at MDC. The MDC LCC model was a combination of the RCA PRICE model and a previously used MCAIR advanced concepts cost model (ACCM). LCC was broken into three functional elements: (1) RDT&E (Research Development, Test and Evaluation), (2) Investment, and (3) O&S (Operations and Support). For comparative studies Figure 16 shows the F-15A avionics impacted by MIRA which would be replaced entirely or partially. The uninstalled weight of this equipment is approximately 187 pounds (85 kgs). The 187 pounds (85 kgs) includes 28 pounds (13 kgs) of equipment which would not be replaced by MIRA, e.g., the Signal Data Recording Set tape recorder, the Air Inlet Controller valve drive circuitry and the Navigation Control Indicator. Therefore, only 159 pounds (72 kgs) of F-15 avionics would be in actuality replaced by MIRA. A MIRA configuration which functionally replaces the 159 pounds (72 kgs) of F-15 equipment is shown in Figure 17. (Figure 17 is used for cost comparative purposes only, i.e., its performance is equivalent, as much as possible, to the performance of the equipment replaced.) The equipment weight estimates for Figure 17 is projected to range from 50 to 60 pounds (23-27 kgs). It should be noted that survivability and mission reliability enhancement is provided by separate LRUs. Each Multifunction Reference Unit is capable of autonomous functional operation.

Fighter cost savings, Figure 18, are shown to range from 69 million to 80 million dollars. This range (a) is based upon subcontractor data, (b) is independent of sensor mechanization, and (c) includes the impact of technology improvements projected to take place by 1980 with sensors and LSI microprocessors.

In performing the cost comparison for transports, the situation is slightly different from the fighter since there is no weapon delivery/lead computing gyro requirements. The transport currently has a dual inertial sensor system (ISS) and a dual air data system (Figure 19). By integrating LRUs for the MIRA configuration, improved mission reliability results due to dual processing capability, and slightly improved production costs results (Figure 20). However, due to potential commonality and standardization of LRUs, qualitative significant savings are expected in the O&S cost element.

Assessment of functional requirements satisfied and identification of performance improvements required were made with the use of subcontractor data and MCAIR math modeling studies using a computer program called SIMSIN, which can evaluate strapped down sensor and system performance over arbitrary flight profiles and provides a time history of error propagation. The results showed that transport functional position and velocity requirements can be met with current TRG and RLG systems but that product enhancement improvements for production are needed for fighters due to the high dynamic environment (Figure 21). Each MIRA subcontractor is at a different level in verifying his equipment capability, which is proprietary, and therefore the "present capability," Figure 21, is not meant to be used to single out a particular subcontractor, but rather is to be considered as an industry-wide combined average. By the end of 1980 it is expected that the improved sensor performance required for fighter applications will be demonstrated, by several equipment suppliers, to at least the brassboard level.

The impact of reliability on life cycle costs is significant. Studies showed that MIRA needs an improvement goal of 11.3 over current operational experience (Figure 22). To achieve this goal, an approach to reliability improvement in both design and management, is believed achievable for MIRA (Figure 23). This type of methodology is currently being used on the F-18 and should start showing payoff benefits within the next two years.

V. LABORATORY DEMONSTRATIONS

In addition to analytical studies, hardware laboratory demonstrations were performed by each MIRA subcontractor.

The hardware demonstrations were aimed at demonstrating key technical issues such as software redundancy management, fault detection and isolation, sensor skewing, cluster skewing, cluster separation, frequency response, resolution, noise content, quantization, failure mode effects, and open and closed loop operation.

During each hardware demonstration, the operation was first demonstrated in a full-up configuration. Simulated failures (hard over, slow over, null) were entered into the system by the operator and the signal outputs recorded and compared with the full-up performance. The impact upon flight control, navigation and weapon delivery outputs were each assessed with appropriate criteria for fault coverage. Honeywell's RLG demonstration set-up is shown in Figure 24; Singer-Kearfott TRG set-up is shown in Figure 25; Litton TRG set-up is shown in Figure 26. It should be noted that demonstration equipments were made available at no cost to MDC or the Air Force.

Each subcontractor's demonstration plan, procedure and criteria was tailored to be supplemental to the analytical tasks performed and was based upon available equipment.

Therefore, it was not required that the equipment demonstrated be configured as a MIRA system, partitioned as a MIRA system, or even of flight worthy construction, even though some of it was.

VI. RESULTS, CONCLUSIONS AND RECOMMENDATIONS

The results and conclusions of key issues discussed in Figure 5 are shown in Figure 27. Overall conclusions for Task 1, shown in Figure 28, range from MIRA feasibility established to the need for a supplemental effort to take the hardware demonstrated in the laboratory and evaluate the software redundancy management and/or fault detection and isolation capability under dynamic moving vehicle conditions, thus providing added feasibility in an environment which provides translation inputs to the system.

Recommendations coming out of Task 1 studies are shown in Figure 29. As a follow-up to the second recommendation, the Litton system was tested in September 1977 in the Air Force Avionics Laboratory mobile evaluation laboratory.

The Task 1 studies, analysis, results, conclusions and recommendations reported here were formulated in a milestone technical report and approved by the Air Force in September 1977 (Reference 2). The MIRA activities currently completed include refining the configuration candidate studies and selecting a final MIRA configuration. Detailed analysis has been made of the selected configuration. The final report currently in work will include the preliminary MIRA technical exhibit, life cycle cost criteria, application to advanced aircraft, and supporting rationale.

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ACKNOWLEDGEMENTS

A key factor in the success of the program has been due to the establishing of feasibility criteria early, identifying key technical issues, keeping the program objectives constantly in front and having a very closely knit and working MIRA team. The authors would like to acknowledge the contribution of each member of the MIRA team: the Air Force Flight Dynamics Laboratory, Air Force Avionics Laboratory, and Air Force Aeronautical Systems Division, and from each industry member, represented by Mr. G. C. Jackson, McDonnell Aircraft Company; Mr. E. Rodriguez, Douglas Aircraft Company; Mr. C. Senechal, Honeywell; Mr. M. Goldstein Singer-Kearfott; and Mr. H. Daubert, Litton.

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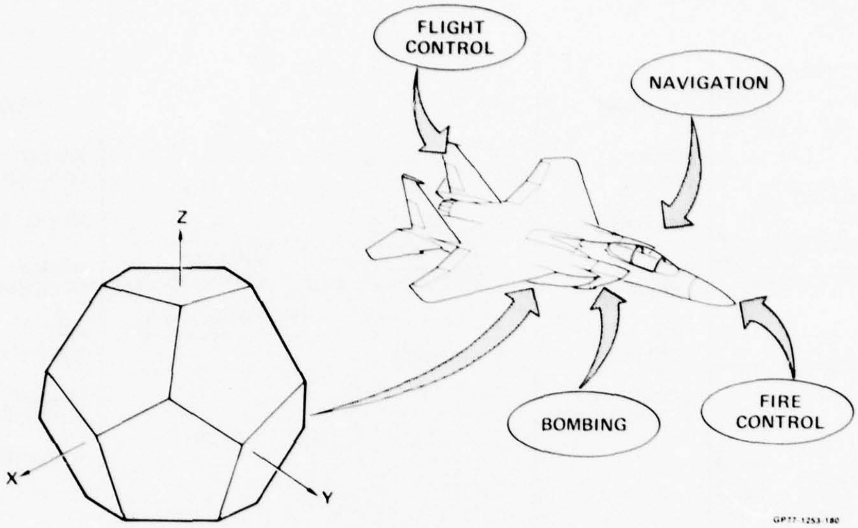

FIGURE 1
MULTI-FUNCTION INERTIAL REFERENCE ASSEMBLY (MIRA)
(An Avionics Subsystem Concept for Application
to Advanced Air Force Aircraft of the 1980-1990s)

- REDUCE LIFE CYCLE COST OF INERTIAL EQUIPMENT
- SYSTEMS ANALYSIS TO DETERMINE BEST CONFIGURATION OF INERTIAL/AIR DATA SENSORS FOR THE FUNCTIONS OF:
  - FLIGHT CONTROL
  - WEAPON DELIVERY
  - NAVIGATION

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FIGURE 2
OBJECTIVE AND APPROACH

PHASE	TASK	ACTIVITY	1976	1977	1978	1979	1980	1981
I FEASIBILITY	I	AIRCRAFT MISSIONS, GOALS, AND ENVIRONMENT						
		MIRA OUTPUT, PERF., AND RELIAB. REQMTS						
		PRELIMINARY DESIGN AND ANALYSIS						
		LABORATORY DEMONSTRATIONS						
	II	CONFIGURATION TRADE STUDIES						
		DETAIL ANALYSIS AND MIRA DEFINITION						
	III	LCC CRITERIA						
		PRELIMINARY SPECIFICATION						
		ADV. A/C APPLICATION STUDY						
II VERIFICATION		MIRA EQUIP. DESIGN, DEV. AND TEST						
		OPEN AND CLOSED LOOP FLIGHT TESTS						
III APPLICATION		APPLICATIONS TO FIGHTERS, TRANSPORTS, BOMBERS, RPV						

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FIGURE 3
MIRA PROGRAM SCHEDULE OVERVIEW

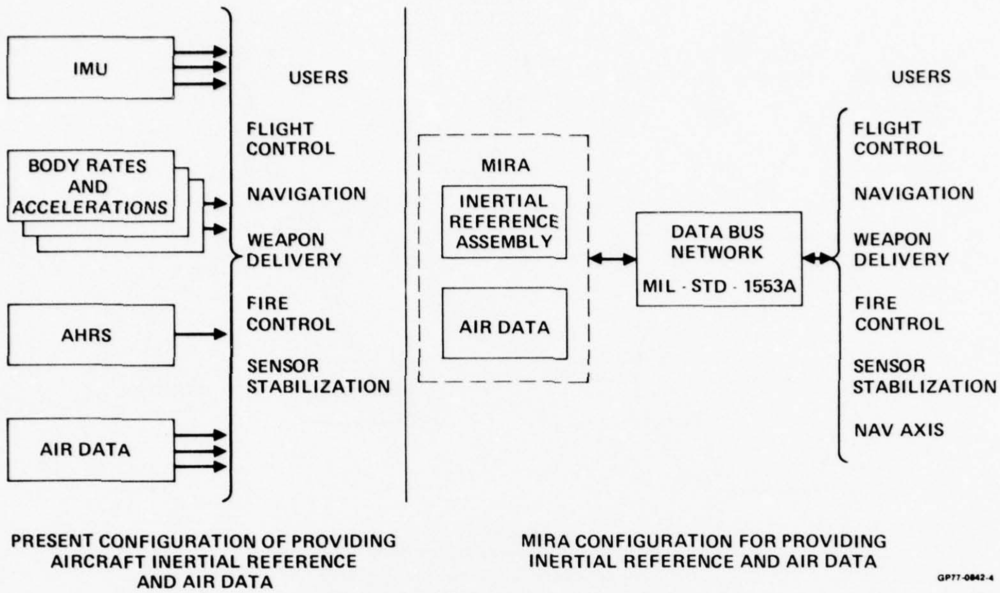

FIGURE 4
MIRA INTERFACE CONCEPT COMPARISON

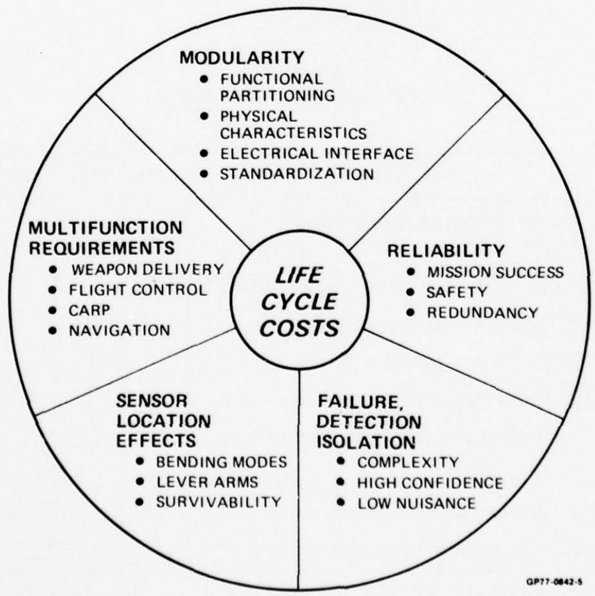

FIGURE 5
MIRA KEY ISSUES

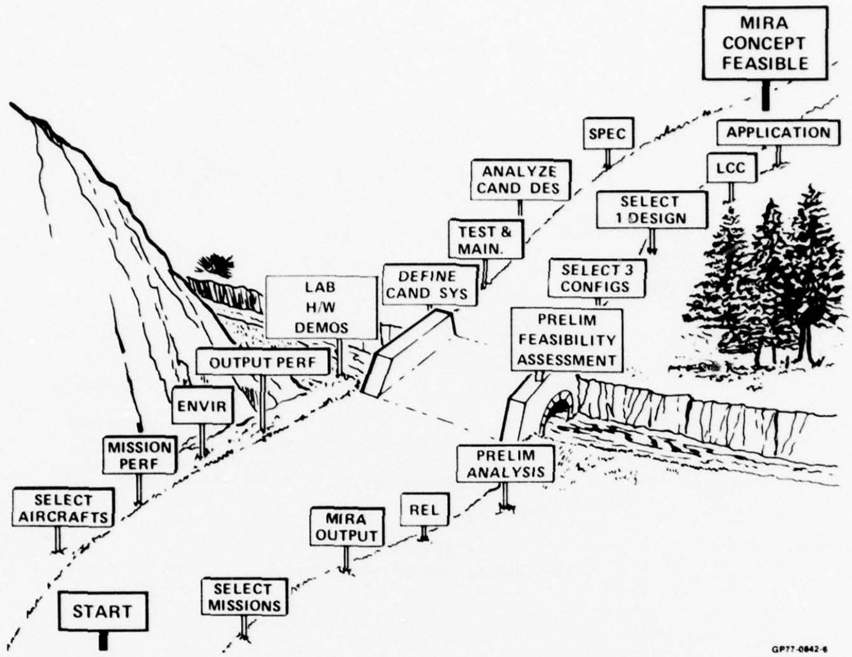

FIGURE 6
PHASE I ACTIVITY - ROAD MAP

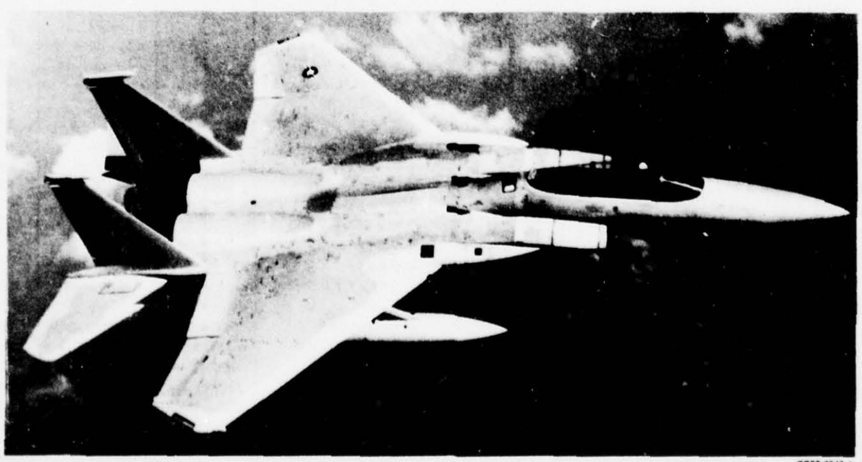

FIGURE 7
F-15A AIRCRAFT

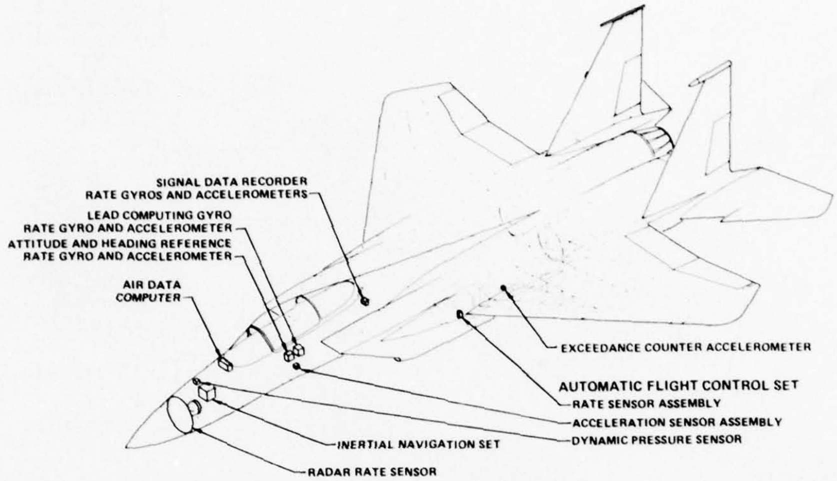

FIGURE 8
F-15 INERTIAL/AIR DATA SENSORS

	OUTPUT PARAMETER	RANGE	ACCURACY STANDARD DEVIATION	FORM	ITERATIONS PER SECOND	MTBF
1.	LATITUDE	±90°	1 NM/HR	1553A MUX	25	3,000
2.	LONGITUDE	±180°	1 NM/HR		25	3,000
3.	INERTIAL ALTITUDE	−1,000 TO 80,000 FT	250 FT		25	3,000
4.	NORTH/EAST VELOCITY (2)	±4,100 FPS	2.5 FPS RMS		80	15,000
5.	VERTICAL VELOCITY	±2,050 FPS	2 FPS RMS		80	15,000
6.	TRUE HEADING	±180°	6 ARC MIN		80	7,500
7.	DRIFT ANGLE	±35°	0.1° TO 1°		25	9,000
8.	AZIMUTH ANGLE	±180°	6 ARC MIN		80	10^7
9.	ELEVATION ANGLE	±90°	2 ARC MIN		80	10^7
10.	ROLL ANGLE	±180°	2 ARC MIN		80	10^7
11.	BODY LINEAR ACCELERATION (3)	±390 FT/SEC^2	0.33 FT/SEC^2		80	10^7
12.	BODY YAW/PITCH RATE (2)	±85°/SEC	0.02°/SEC		80	10^7
13.	BODY ROLL RATE	±360°/SEC	0.02°/SEC		80	10^7
14.	BAROMETRIC ALTITUDE	−1,000 TO 80,000 FT	2 FT OR 0.2%		25	10^7
15.	TRUE ANGLE OF ATTACK α	−10° TO 35°F	$0.12 + 0.004\,\alpha + 10$		25	10^7
16.	INDICATED AIRSPEED V_I	14 TO 1,600 KTS	4 KTS $V_I < 100$ 2 KTS $V_I > 100$		25	10^7
17.	TRUE AIRSPEED	60 TO 1,710 KTS	5 KTS		25	10^7
18.	MACH NUMBER	0.09 TO 3 M	0.005 TO 0.01		25	7,500
19.	TOTAL TEMPERATURE	−62°F TO 233°F	0.44°		25	9,000
20.	FIRST RAMP SERVO DRIVE	±50 MA	INTO 100 Ω	LOAD	CONTINUOUS	10^7
21.	DIFFUSER RAMPS SERVO DRIVE	±50 MA	INTO 100 Ω	LOAD		10^7
22.	BYPASS SERVO DRIVE	±50 MA	INTO 100 Ω	LOAD		10^7
23.	NON MUX BUS DISCRETES (10)	ON-OFF	N/A	WIRES	RANDOM	30,000
24.	MUX BUS DISCRETES (10)	ON-OFF	N/A	1553A MUX		30,000
25.	REACTION TIME	10 MINUTE (MAX) FOR GYRO COMPASS ALIGN: 3 MINUTE (MAX) FOR STORED HEADING ALIGN.				

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FIGURE 9
KEY OUTPUT REQUIREMENTS

10-9

	KOPS	WORDS	REFRESH/SEC
EXECUTIVE	16	1,000	128
GYRO COMPENSATION	41	538	64
ACCEL COMPENSATION	10	126	64
FDI/RM	15	312	64
DESIGN EQUATIONS	10	700	64
ALIGNMENT MATRIX	8	175	64
QUATERNION	39	900	64
VELOCITY/POSITION	45	1,200	32
ATTITUDE/RATES/ACCEL, ETC	4	500	128
AIR DATA	364	3,500	20
OUTPUT	5	350	A/R
DATA BASE	–	1,500	–
SUBROUTINES	–	600	–
BITE	40	600	–
TOTALS	597	12,001	

Note: Air inlet requires 1,000 additional words, 7 KOPS

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FIGURE 10
COMPUTATIONAL REQUIREMENTS ESTIMATE
(Representative MIRA System - Fighter)

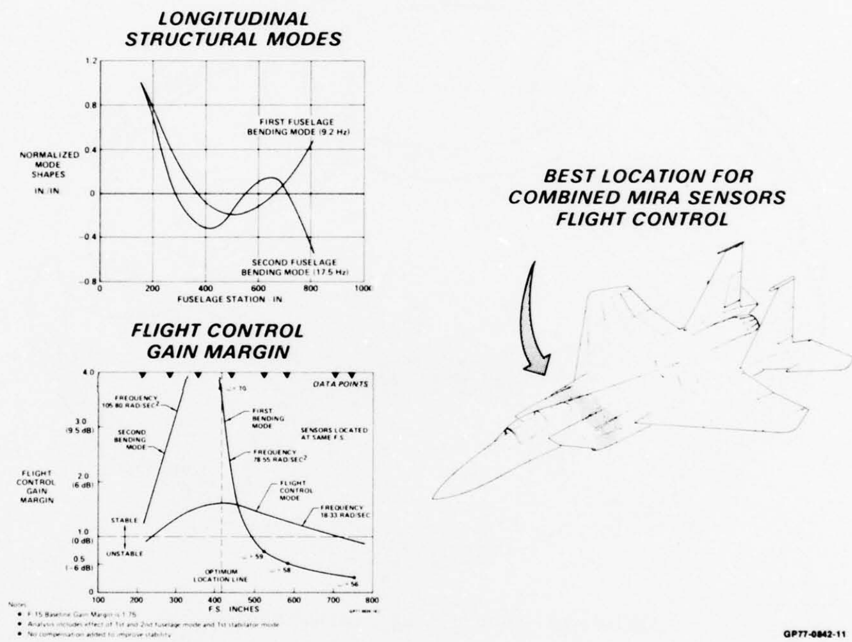

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FIGURE 11
FIGHTER ENVIRONMENT

MIRA FUNCTION	POSITION	HEADING	ATTITUDE	ANLR BODY RATE	ACCEL	VELOCITY	AIR DATA	FIGHTER APPORTIONED HARDWARE SERIES MTBF (HR)
NAVIGATION	✓	✓				✓	✓	1071
FLIGHT CONTROL			✓	✓	✓	✓	✓	4970
WEAPON DELIVERY		✓	✓	✓	✓	✓	✓	5596

- ALL FLIGHT CONTROL OUTPUTS NEED TO BE AT LEAST FAIL-SAFE
- MISSION EFFECTIVENESS REQUIREMENTS ARE MET USING THE ABOVE HARDWARE MTBF APPORTIONMENT

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FIGURE 12
RELIABILITY - FIGHTER

10-10

	HARDWARE SERIES MTBF (HOURS)	MISSION RELIABILITY
• C-15A MIRA RELIABILITY REQUIREMENTS	150	0.99929

- LOSS OF AIR DATA AND INERTIAL OUTPUTS ARE CRITICAL TO FLIGHT SAFETY UNDER IFR CONDITIONS
- THE DEGREE AND CRITICALITY OF MIRA FAILURE WILL BE ESTABLISHED WHEN DETAILED SAFETY AND HAZARD ANALYSIS ARE PERFORMED IN TASK II

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FIGURE 13
RELIABILITY - TRANSPORT

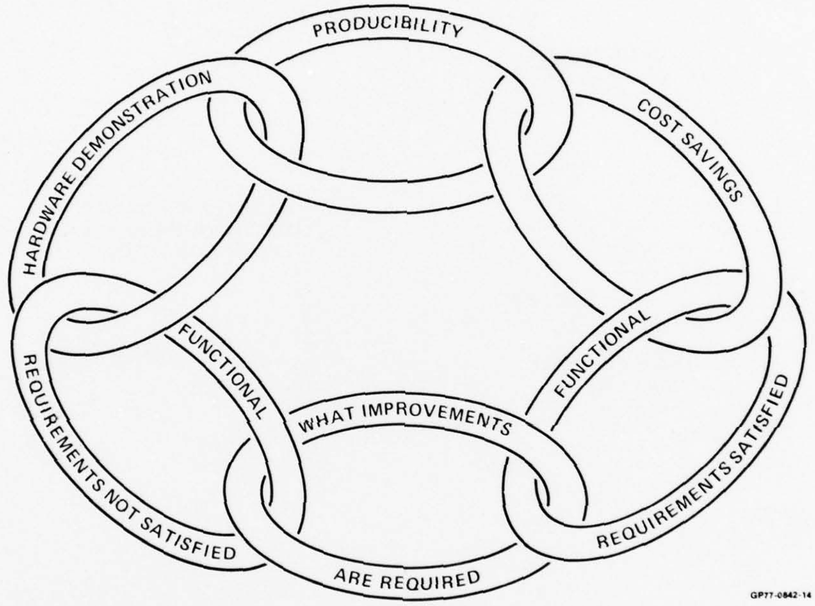

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FIGURE 14
PRELIMINARY FEASIBILITY ASSESSMENT CRITERIA

- DETERMINE CONVENTIONAL AVIONICS TO BE REPLACED OR AFFECTED BY MIRA
- SELECT A REPRESENTATIVE BASELINE MIRA CONFIGURATION
- SELECT A COST MODEL FOR EACH ELEMENT OF COSTS
  - R&D (DEVELOPMENT)
  - INVESTMENT (PRODUCTION)
  - O&S (DEPLOYMENT)
- PERFORM COST PREDICTIONS FOR THE CONVENTIONAL AND MIRA AVIONICS LIFE-CYCLE-COST AND DETERMINE THE POTENTIAL SAVINGS
- CONDUCT FIGHTER COST STUDY IN PARALLEL WITH TRANSPORT COST STUDY

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FIGURE 15
LIFE-CYCLE-COST FEASIBILITY STUDY APPROACH

10-11

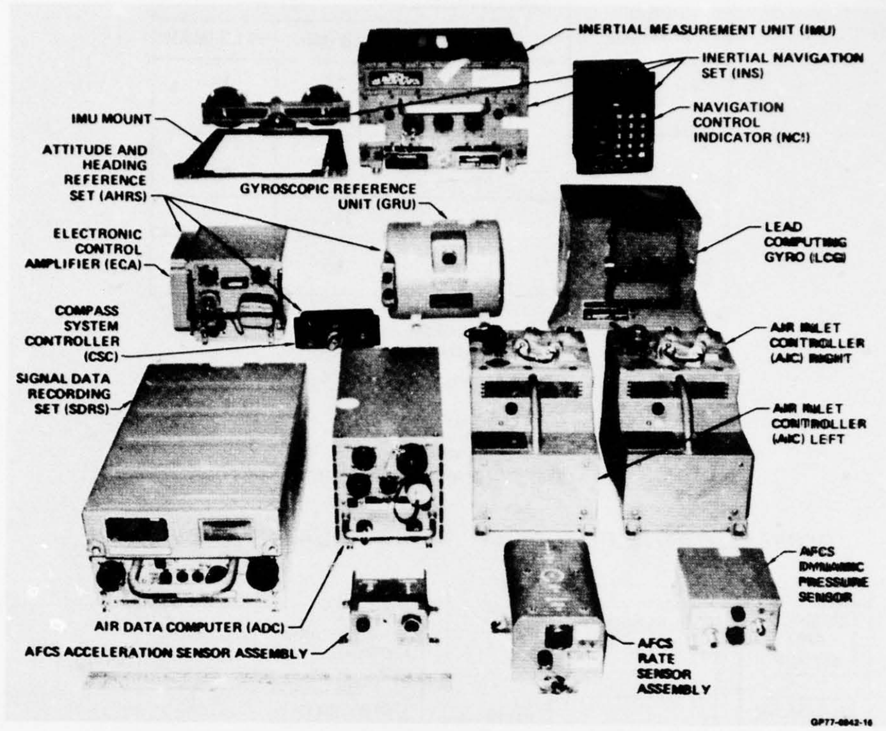

FIGURE 16
F-15 EQUIPMENT IMPACTED BY MIRA

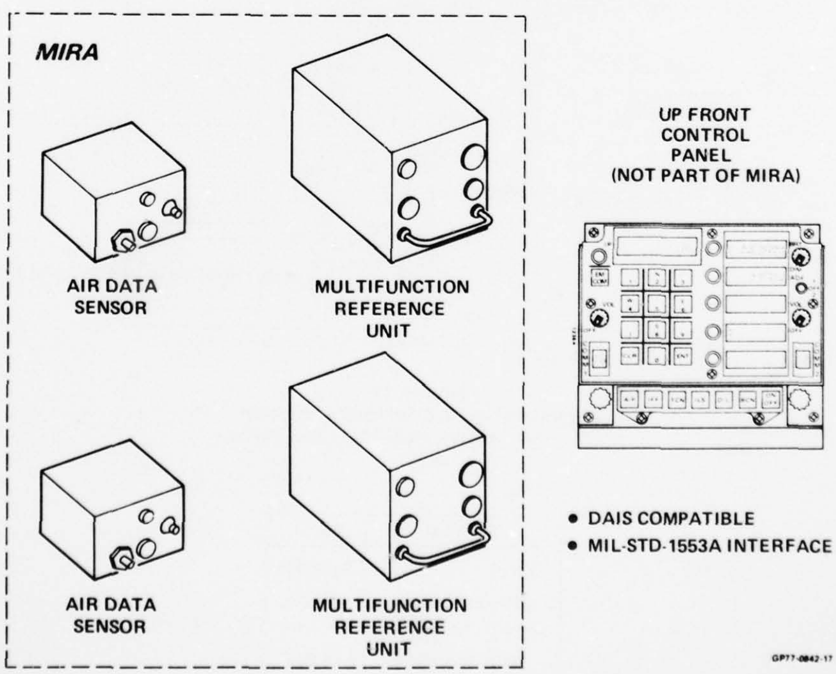

FIGURE 17
REPRESENTATIVE SYSTEM CONFIGURATION FOR
COST COMPARATIVE ANALYSIS - FIGHTER

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THE IMPACT OF INTEGRATED GUIDANCE AND CONTROL TECHNOLOGY ON WEA--ETC(U)
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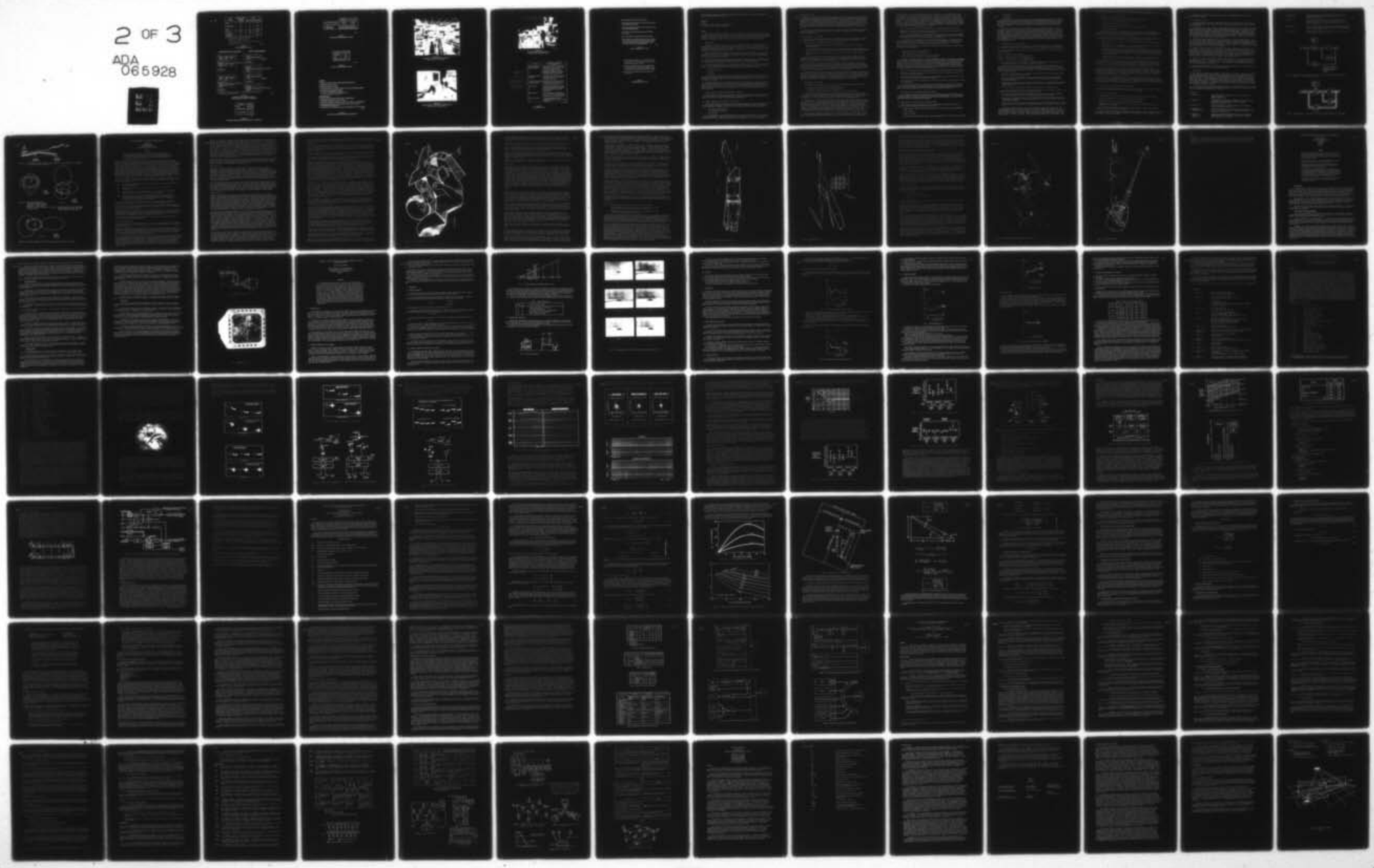

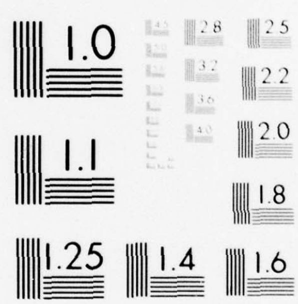

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

10-12

COST ELEMENT	CONVENTIONAL (159.3 LB) AVIONICS	MIRA	
		50 LB (MIN)	60 LB (MAX)
R&D	30	17	18.5
INVESTMENT	60	29	32.0
O&S*	95	59	65.5**
TOTAL	185	105	116.0
MIRA COST SAVINGS	N/A	80	69.0

Note (1) All numbers are in millions of 1 Jan. 1977 dollars.
(2) 200 systems, 144 aircraft, 15 year operational life
(3) Technology based on 1985 aircraft go-ahead

*O&S includes investment support costs because PRICE investment does not include investment support

**Approximated from ratio of investment costs

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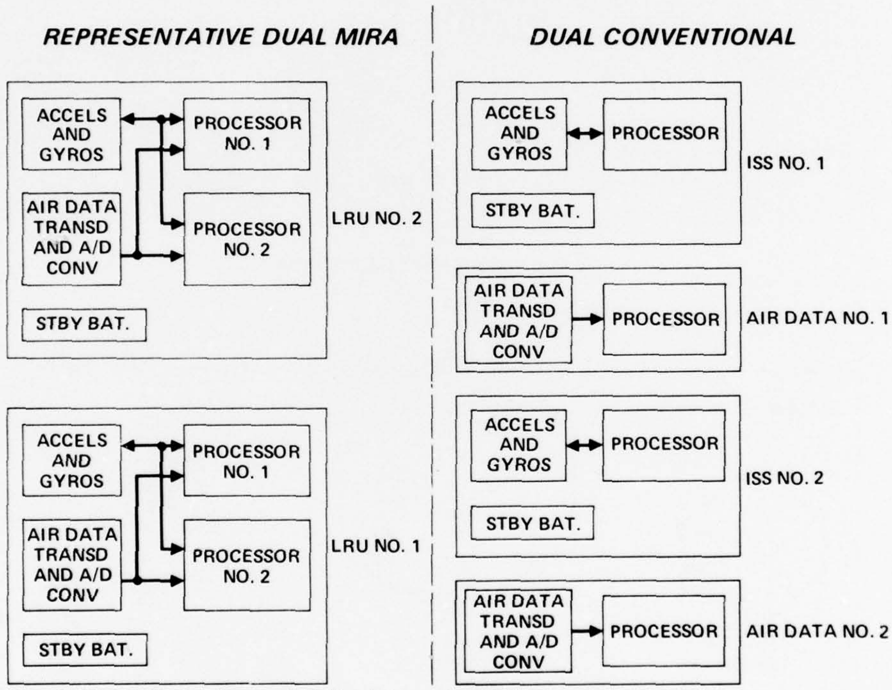
FIGURE 18
COST COMPARISON - FIGHTER



Note: Dual MIRA is functionally more reliable than dual conventional MIRA due to dual processing of inertial and air data in each MIRA processor.

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FIGURE 19
REPRESENTATIVE CONFIGURATION FOR
COST COMPARATIVE ANALYSIS - TRANSPORT

SYSTEM	COST
MIRA	$223,000
CONVENTIONAL	$224,000
COST SAVINGS	$1,000

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FIGURE 20
EQUIPMENT PRODUCTION COST COMPARISON - TRANSPORTS

	PRESENT CAPABILITY	REQUIREMENT
• POSITION ACCURACY	2-3 NM/HR	1 NM/HR
• VELOCITY ACCURACY	8-10 FPS	2-3 FPS
• ASSOCIATED REDUNDANCY	FURTHER INVESTIGATION UNDER DYNAMIC CONDITIONS	

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FIGURE 21
PERFORMANCE IMPROVEMENT REQUIRED

EQUIPMENT	MTBF (HRS)
CURRENT	67.0
MIRA	760.0
IMPROVEMENT GOAL = 11.3	

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FIGURE 22
RELIABILITY IMPROVEMENT REQUIRED

DESIGN

- CONTRACTUAL LIMITATION OF COMPONENT STRESS LEVELS
- IMPROVED THERMAL DESIGN
- IMPROVED PART APPLICATION
- FEWER COMPONENTS - ELIMINATE GIMBALS, USE OF LSI MICROPROCESSORS
- RELIABILITY DEVELOPMENT TESTING
  - DESIGN-LIMIT ENVIRONMENTAL EXTREMES
  - REALISTIC DUTY CYCLES TO MATCH MISSION PROFILES
  - ACCUMULATE EQUIVALENT SERVICE LIFE

MANAGEMENT

- IMPROVED PARTICIPATION DURING EARLY DESIGN PHASE
- INCREASED FUNDING FOR EARLY TESTING
- EQUIPMENT DESIGNERS TASKED WITH SPECIFIC RELIABILITY REQUIREMENTS
- IMPROVED CORRELATION OF LABORATORY AND FIELD RELIABILITY MEASUREMENT GROUND RULES
- CONTINUOUS AND AGGRESSIVE FOLLOW-UP THROUGHOUT LIFE OF PROGRAM

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FIGURE 23
METHODS OF ACHIEVING RELIABILITY IMPROVEMENT

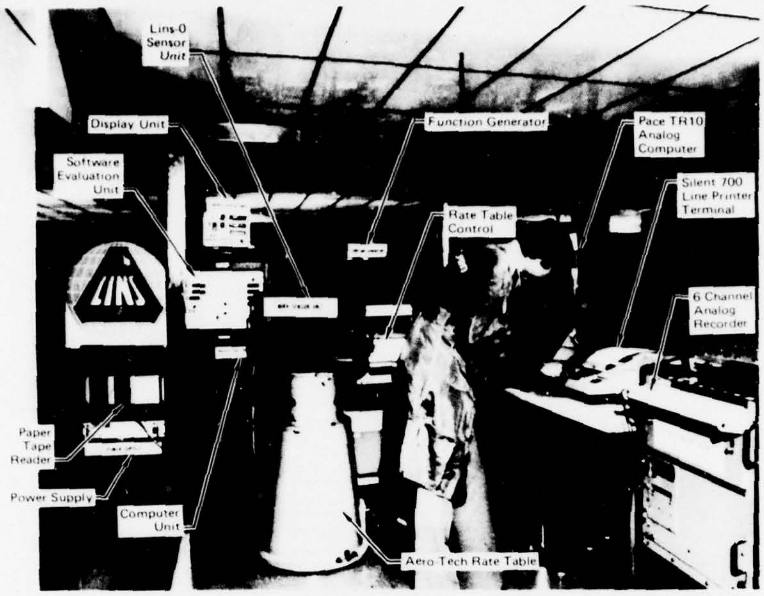

FIGURE 24
HONEYWELL LABORATORY DEMONSTRATION
(Ring Laser Gyro - 1 Tetrad Cluster)

FIGURE 25
SINGER-KEARFOTT LABORATORY DEMONSTRATION
(Tuned Rotor Gyro - 2 Skewed Clusters)

GP77-0042-26

FIGURE 26
LITTON LABORATORY DEMONSTRATION
(Tuned Rotor Gyro - 2 Skewed Clusters)

MIRA KEY ISSUES

KEY ISSUE	PRELIMINARY ASSESSMENT OF FEASIBILITY
MULTI-FUNCTION REQUIREMENTS	FUNCTIONAL OUTPUTS OF INERTIAL POSITION, VELOCITY, ACCELERATION, RATES AND AIR DATA CAN BE OBTAINED. TECHNOLOGY IMPROVEMENTS WILL IMPROVE POS AND VEL ACCURACY IN THE NEAR FUTURE
SENSOR LOCATION EFFECTS	CONTROL LAWS CAN BE MODIFIED TO ACCOMMODATE USE OF PHYSICALLY SEPARATED MRU LRUs
FAILURE DETECTION AND ISOLATION	DEMONSTRATED IN LAB FOR HARD OVER, SLOW OVER AND SOFT OVER FAILURES WITH RLG AND TRG
RELIABILITY	REDUNDANT AND SKEWED SENSORS AND CLUSTERS PROVIDED ADEQUATE MISSION AND SAFETY OF FLIGHT RELIABILITY
MODULARITY	MIL-STD-1553A MUX CAN BE USED. INERTIAL/AIR DATA FUNCTIONS CAN BE INTEGRATED AND STANDARDIZED
LIFE-CYCLE-COST	FIGHTER SAVINGS: $69 - $80 M TRANSPORT SAVINGS: QUALITATIVELY SIGNIFICANT

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FIGURE 27
KEY ISSUES RESULTS

- MIRA FEASIBILITY HAS BEEN ESTABLISHED
- MIRA PROVIDES SIGNIFICANT COST SAVINGS AND MERITS CONTINUED DEVELOPMENT
- MIRA PERFORMANCE CAPABILITY WILL BE ADEQUATE FOR FIGHTER AND TRANSPORT
- RELIABILITY AND MAINTAINABILITY REQUIREMENTS WILL BE MET
- AIR DATA COMPUTATION SHOULD BE PART OF MIRA
- SUPPLEMENTAL DEMONSTRATION EFFORT TO EVALUATE MIRA VELOCITY, ACCURACY AND REDUNDANCY UNDER DYNAMIC TRANSLATION CONDITIONS DURING PHASE I WILL PROVIDE ADDED KEY INFORMATION

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FIGURE 28
OVERALL CONCLUSIONS - TASK I

- CONTINUE MIRA FEASIBILITY STUDY (PHASE I) FOR DEVELOPMENT OF FINAL MIRA CONFIGURATION AND PREPARATION OF SPECIFICATION
- INITIATE A SUPPLEMENTAL TEST TO DEMONSTRATE PERFORMANCE ON SOFTWARE REDUNDANCY USING LAB HARDWARE ON A MOVING VEHICULAR TEST BED
- BEGIN DETAIL PLANNING FOR MIRA PHASE II ADP FLIGHT TEST

GP77-0842-29

FIGURE 29
RECOMMENDATIONS - TASK I

APPLICATION OF PARALLEL FILTERS FOR MALFUNCTION DETECTION AND ALTERNATE MODE CAPABILITY IN AN INTEGRATED NAVIGATION SYSTEM

11-1

by

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SUMMARY

This paper presents a software method for malfunction detection and alternate mode capability in dynamic systems where slowly increasing sensor errors may occur. The method is based on parallel Kalman filters and tests on the filter outputs. The paper describes how this method will be implemented in the integrated navigation system for the new Norwegian coastguard vessels. Some preliminary simulation results are presented.

1 INTRODUCTION

Malfunction detection has proven to be an important feature of automatic and complicated systems. This paper presents a software method for detection and isolation of "difficult" malfunctions in dynamic systems. Despite the specific application at hand, the method focused in this paper is generally applicable to sensor data integration where gradually sensor-deterioration may occur.

The method presented is neither supported by practical experience nor by extensive simulation results. The contribution of this paper is rather to suggest a somewhat new approach to a difficult and important problem when trying to obtain confident estimates from an unreliable dynamic system.

The ideas described in this paper stem from the development of an integrated navigation system for the new Norwegian coastguard vessels. The development is being carried out at Kongsberg Våpenfabrikk and Norwegian Defence Research Establishment.

The navigation information from the different sensors are to be fed into a minicomputer. The information is processed to obtain the "best" estimates of the vessels position, velocity, and orientation. The processing will be designed to produce:

- reliable estimates
- accurate "
- confident "

The expression "reliable estimates" means that the estimates are available most of the time. Accurate estimates means that the estimates are close to the true state. The expression "confident estimates" denotes estimates possessing a true accuracy close to the expected accuracy.

Simulations showed that the common malfunction detection methods failed to detect and identify gradually increasing large biases in the radionavigation systems Decca and Loran C which was to be used. These errors, caused by athmospherical disturbances, drasticly deteriorated the position accuracy of the integrated navigation system if they were not detected. The method described here is a result of the investigations which were preformed.

2 DESCRIPTION OF A METHOD FOR MALFUNCTION DETECTION

This chapter describes and discusses a somewhat new approach to malfunction detection. The first part of the chapter briefly lists the usual methods for malfunction detection in order to support the following presentation.

2.1 Common approaches to malfunction detection

Before the method is presented, it may be appropriate with a short recapitulation of the most common approaches to malfunction detection in dynamic systems where Kalman filter technique is applied. See ref (1). The usual tests focus on (see figure 1):

- the received measurements
- the measurement residuals
- the estimates

If the redundancy is sufficiently high, one may perform majority voting on the received measurements. Other sorts of reasonableness checks are also used. Knowing the normal range of the variables being estimated, alarms may be given if the estimates exceed these limits.

11-2

Most of the malfunction detection methods perform tests on the measurement residuals (the differences between the received measurements and the corresponding expected ones; the expected measurements being calculated from the current estimates). The statistical properties of the measurement residuals are easily calculated for normal situations, and the different malfunction detection methods try to detect violations from the normal statistics. Some of the methods also take into account the resulting statistical properties of the measurement residuals when spesific malfunctions occur; see ref (2). The tests may be made more efficient by designing so-called failure sensitive filters, see ref (3).

Reference (4) describes an application of parallel filters for malfunction detection where the measurement residuals are used to calculate the probabilities of different specified malfunctions.

2.2 Description of the method

The method to be presented inherits a lot from known malfunction detection methods. The ideas presented in this paper are partly inspired by the method presented in (5).

The method may be summarized as follows (see figure 2):

- Apply independent Kalman filters, each filter taking input from a subset of available sensors.

- Check the differences between the sets of filter estimates and apply hypothesis testing based on the covariance matrices calculated in the parallel Kalman filters.

- When a malfunction is identified, switch to a filter which excludes the bad sensor.

The filter inputs are generally subsets of the available sensor output, but one Kalman filter may use data from all the sensors. The system estimates may either be a combination of estimates from some of the filters or they may be the estimates from one single filter, preferably the one with input from all sensors.

When a malfunction occurs, the sets of estimates from the different filters are influenced by the erroneous data in different ways, and some sets may not be influenced at all. The differences between the sets of estimates is accordingly a means to identify and isolate the malfunction. In normal operation the sets of estimates will differ due to the normal noise and errors in the system. This is the reason why hypothesis testing based on the covariance matrices of the Kalman filters is applied.

This parallel filter approach offers a lot with respect to alternate mode capability. When one of the sensors is declared malfunctioning and its data rejected, the sensor may have deteriorated the accuracy of the system estimates prior to the identification. Among the sets of estimates there may now excist one (or more) that is not influenced by the data from the malfunctioning sensor. These estimates ought to be regarded as the "best" estimates, and should be defined as the systems estimates. (See figure 3).

2.3 Discussion of the method

This section will discuss the following topics concerning the presented method:

- relations to other malfunction detection methods
- some general aspects concerning implementation
- hypothesis testing based on filter outputs and covariance matrices

We have not performed any detailed investigation to compare the presented method with other malfunction detection methods; some simulation results are, however, mentioned in section 4.2. Still we would like to make a few comments relating to the methods presented in (5) and (4); both being fairly similar to the presented method.

The method in (5) also tests for malfunctions by monitoring the differences between sets of estimates. The method presented here is probably more generally applicable than the method in (5) in that it applies Kalman filters taking into account the noise in the sensors during normal operation, rather than using Luenberger observers. The method presented also allows more freedom in combining the inputs to the estimators because there is no requirement that the basic state variables has to be observable by the single sensors. Taking into account the calculated covariance matrices in the alarm- and identification scheme seems more theoreticly sound than the approach in (5). However, the method presented seems to demand far more computer capacity than the method in (5).

The method decribed in (4) has several common features with the presented method. Both methods apply independent Kalman filters, and both methods have, at least in principle, the same alternate mode capability. The basic difference is that the method in

(4) applies the sets of measurement residuals to identify occuring malfunctions rather than monitoring the differences between the sets of estimates. Theoreticly the method in (4) seems to be optimal in the Bayesian sense. However, simulations indicate that the success of this method is strongly tied to the spesific applications. Reference (6) questions the "identifiability" of the method and mentions the lack of general convergence proofs. (A simple performance statement of the presented method is found later in this section.) As for the number of Kalman filters to implement, the presented method demands a limited number, while the method in (4) in principle demands an infinite number. However, the presented method also may have to reduce the number of parallel Kalman filters because of restricted computer capacity available.

One interesting qualitative statement about the presented method may be noted:

Assume that a system contains N sensors, and that the estimates of the system variables have to meet certain accuracy requirements.

If all N Kalman filters with inputs from N-1 sensors meet the accuracy requirements, then the method described is able to detect all sensor malfunctions that will cause the system estimates to violate the accuracy requirements.

This statement is easily realized. Say that the sensor number i is malfunctioning. Since at least one set of estimates is not influenced, the difference between this set and the system estimates will become too large to be accepted from the corresponding covariance matrices, before the accuracy requirements are violated.

Several important questions are arising when a system with fault detection capability as decribed is to be designed:

- How many parallel filters to design?
- How to design the filter models?
- What sensor inputs should be fed to each filter?
- How to design the malfunction detection scheme?

The answer to these questions are stronly dependent on the characteristics and number of the sensors available and on the computer capacity available. It could be desirable to obtain sets of estimates based on single sensors and to perform the identification by majority voting on the estimates (5). However, if estimates may be obtained this way, they are often too uncertain to be useful. It could also be desirable to design the parallel filters as failure sensitive filters (3). This approach may lead to a better malfunction detection capability, but the alternate mode capability will suffer because the filter estimates is not of minimum variance type.

One way of implementing the malfunction detection and identification scheme may be:

1) The sets of estimates from the parallel Kalman filters are compared in pairs, and the differences are obtained.

2) An alarm is activated if at least one of the differences is too large to be consistent with the corresponding covariance matrices.

3) The identification of the malfunction is based on the pattern constituted by the set of differences.

The identification scheme may involve the use of decision tables. The conditions to test on can be determined by simulating the different kinds of malfunctions in the normal noisy environment. The hypothesis testing in 2) may be performed in different ways. One method is described in (7), another is described in section 3.2.

3 APPLICATION IN AN INTEGRATED NAVIGATION SYSTEM

This chapter describes the application of the malfunction detection method in the integrated navigation system for the new Norwegian coastguard vessels. The integrated navigation system is briefly described, and the specific implementation of the failure detection method is described and discussed.

3.1 Description of the integrated navigation system

The integrated navigation system for the new Norwegian coastguard vessels will consist of the following navigation aids (nav aids):

- NNSS receiver (NNSS: US Navy Navigation Satellite System, also called TRANSIT)
- Decca receiver
- Loran C receiver
- Inertial Navigation System (INS, accuracy approximately 1 nautical mile per hour)

- Relative log

11-4 - Gyrocompass

The information from the nav aids is fed into a minicomputer through a bus system. The minicomputer will process the information using Kalman filter technique. A recent formulation of the Kalman filter algorithm called UD-factorization will be applied (8), (9). The principal quantities to be estimated are ships position, velocity, heading, roll and pitch-angles. In addition sea currents and offsets in the nav aids will be estimated.

The NNSS contributes with a high position accuracy to the system when a fix is obtained. The fixes are obtained at intervals varying from about ½-2 hours. The Decca and Loran C systems contribute with their relative high signal stability. Bias erros in these systems may be estimated when the NNSS receiver gets a position fix. The INS contributes with a high short term stability in velocity and position indications, in addition to supplying heading roll, and pitch angles. The relative log may be regarded as a redundant speed sensor when the INS operates together with Decca and Loran C, but it contributes with velocity stabilization when some of these nav aids are not available. The gyrocompass also feeds the Kalman filter with inputs, but its main function is a back-up heading sensor.

3.2 The implementation of the method

This section describes how the malfunction detection method will be implemented in the navigation system at hand. The interactions with the operator is also mentioned.

The implementation will consist of four parallel Kalman filters with the following characteristics:

Filter 1: Input from all nav aids

Filter 2: Input from all nav aids except INS

Filter 3: Input from all nav aids axcept Decca

Filter 4: Input from all nav aids except Loran C

The filter models are all designed to produce minimum variance type estimates from the received measurements (i e not failure sensitive filters).

Only the position estimates from the parallel filters will be compared here. The main steps of the malfunction detection scheme is described in section 2.3. One key question is whether or not a pair of position indications are too far apart to be consistent with the associated covariance matrices. This question will be answered accoring to the following scheme:

1) Construct a grid network in the position plane containing the position indications from the four parallel Kalman filters.

2) At each grid point (representing a certain position) calculate the probability density associated with each of the four position indications and corresponding covariance matrices.

3) When stepping through all grid points, decide for each of the six pairs of positions indications the most probable grid point, and calculate the confidence of each of these six points.

4) If one of the six confidences are below a cerain limit, an alarm is given.

When the system has identified a malfunctioning nav aid, the operator is warned. The operator then has to decide to take one of three actions:

1) He decides to exclude the declared malfunctioning nav aid. The system estimates will then be based on the filter excluding the malfunctioning nav aid.

2) He decides to ignore the warning. The computations in the system will continue as if nothing had happend. The system estimates are those from the Kalman filter with input from all sensors.

3) He decides to trust the declared malfunctioning nav aid more than the other nav aids. In this case he reinitializes the integrated system with information from this nav aid.

It should be noted that only one of the filters may physically control the INS. In normal operation this is done by the Kalman filter with input from all nav aids. The misalignement estimates from this filter is then used to torque the gyroes in order to align the INS platform to north and level. The other filters taking inputs from the INS, each "controls" a model of the INS. These models derive misalignment estimates of the INS platform. If Decca or Loran C is declared malfunctioning, the filter without this nav aid takes the control. This filter then aligns the platform according to its own misalignment estimates, and thereby remove the influence from the malfunction.

3.3 Discussion of the implementation

This section discusses the following aspects of the implementation:

- the reasons for the implementation of this method
- the reasons for the choice of filters
- the expected capability of the method under various conditions
- computer requirements

The presented method will be implemented in the navigation system at hand to cope with slowly increasing bias errors in Decca and Loran C; biases due to atmospherical disturbances. The more common malfunction tests seem to fail in the detection and identification of these kinds of errors (see section 4.2).

One may ask why we decided to use just four Kalman filters, and why just these subsets of sensors were selected as inputs. The main reason are listed below:

- Restricted computer capacity limited the number of parallel Kalman filters to just a few.
- This method is to be implemented in order to detect drifts in Decca and Loran C. Therefore some of the subsets had to exclude these nav aids.
- One Kalman filter without the INS had to be designed anyway in order to handle the filtering when the INS is down because of the specific design of this system. If one runs this filter in parallel when the INS is operating, one may be able to identify failures in INS in addition to support the identification of drifts in Decca and Loran C.
- Common malfunction detection methods (which also will be implemented) seem to be equally capable of detecting malfunctions in the rest of the nav aids.

We expect that this method is able to both detect and identify all severe errors in Decca and Loran C, provided that all four filters operate and that only one error is present at any time. We also expect that severe errors in the INS may be detected by this method. If one or more of the four filters are not running due to lack of inputs from certain nav aids, the identification capability will drasticly suffer.

The method presented here put certain requirements on the computer capacity. The computer memory needed will increase with the number of parallel filters implemented. However, the increase will not be proportional to the number of filters. This is because the same code to a very large extent may be used by the different filters (the filters will be executed sequentially). In addition many of the quantities in the different filters may use the same storage locations.

The increase in the computation time required may in many cases be a serious restriction on the use of parallel filters. In the application at hand however, the dynamics of the error states used in the modelling of the system are low. Therefore a sampling time of 30 seconds may be used, whereas four parallel filters requires approximately 5 seconds execution time (about 20 states in each filter).

4 EVALUATION OF THE MALFUNCTION DETECTION METHOD

This chapter describes the planned steps in the evaluation of the failure detection method. Some preliminary simulation results are also presented.

4.1 Steps in the evaluation

The evaluation of a failure detection method may be divided into three steps:

1) Simplified simulation of the system and the error conditions.
2) Detailed simulation of the system and the error conditions.
3) Simulations of the system using recorded measurements contaminated by the error conditions to be detected.

Step 1 involves covariance calculations and observing the response on principal estimates under different error conditions.

At step 2 a detailed simulation program should be made and used to simulate different algorithms for the failure detection method to be implemented. The algorithms of the final system should be found here.

At step 3 it should be possible to arrive at final parameters in the failure detection algorithms.

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4.2 Preliminary results

As stated earlier, extensive simulation results are not available. Before the idea of the presented method was conceived, some of the known malfunction detection methods was investigated.

It turned out that the so-called three-sigma test (rejection of too large measurement residuals) would fail to detect slowly increasing bias errors in Decca and Loran C, even when prefiltering was applied to reduce the random noise component. Simplified simulations of the parallel filter approach taken in (4) did not produce encouraging results, and the method was dropped. It was demonstrated that failure sensitive filters increased the probability of identifying these slowly increasing errors by means of three-sigma-tests and whiteness-tests of the measurement residual sequence. However the results were not satisfying and this method was also dropped.

Only preliminary simulation results are available for the presented method. These are of the first type described in the previous section. Errors were introduced into the nav aids Decca, Loran C, and the INS, and the resulting influence on the estimates from the four parallel Kalman filters were observed. A normal situation is shown in figure 4 while two examples of malfunctions are shown in figures 5 and 6. The identification scheme described in section 3.2 is not yet implemented.

Figure 5 shows a typical situation when the position information from Decca or Loran C has drifted. The filter which relies most on the malfunctioning nav aid exhibits the largest position errors while the one without this nav aid is not affected at all. Because of the unsymetrical position indications of the four filters, the malfunction may be identified. In this example a 400m bias error in Decca is softly introduced (the first derivate is continuous) during one hour. This figure shows the situation after 40 minutes, when the Decca error has increased to approximately 300m.

Figure 6 shows the situation 30 minutes after an abrupt bias shift in the drift of the east gyro. The bias introduced is ten times larger than the standard deviation of the normal gyro drift. As expected, the filter estimates are less sensitive to malfunctions in the INS than in Decca and Loran C. The pattern of the position indications is not as clear-out here as i figure 5.

A better understanding of the malfunction detection capability of this method will probably be gained during the comming investigations.

5 CONCLUSION

This paper has presented a somewhat new approach to malfunction detection in dynamic systems. The method also offers alternate mode capability. It will be implemented in the integrated navigation system for the new Norwegian coastguard vessels in order to detect and identify slowly increasing large bias errors in the radionavigation systems Decca and Loran C. Four independent Kalman filters will run in parallel. One filter takes input from all nav aids while the three additional filters ommit one nav aid each of INS, Decca, and Loran C. The malfunction detection scheme will work with the differences between the position indications from the four Kalman filters taking into account the calculated covariance matrices.

Preliminary simulations show encouraging results. Also some kinds of errors in INS seem to be detected. The method will soon be evaluated by a simulation program, simulating the integrated navigation system in detail.

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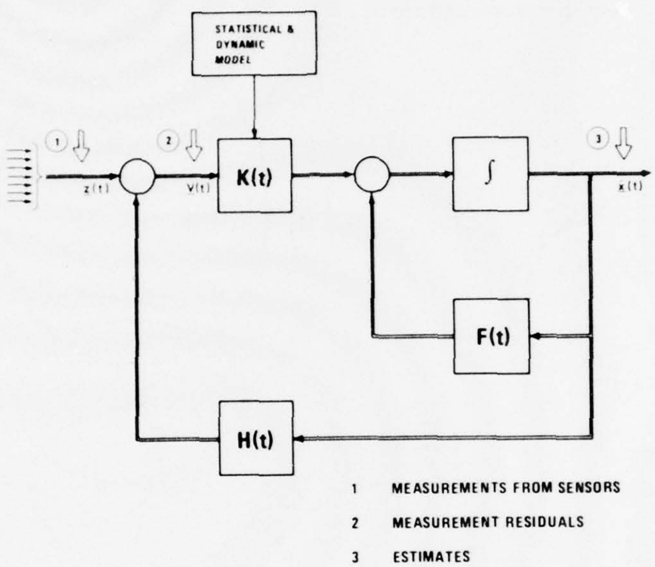

Figure 1 A Kalman filter implementation showing the most common quantities being subject to failure detection tests

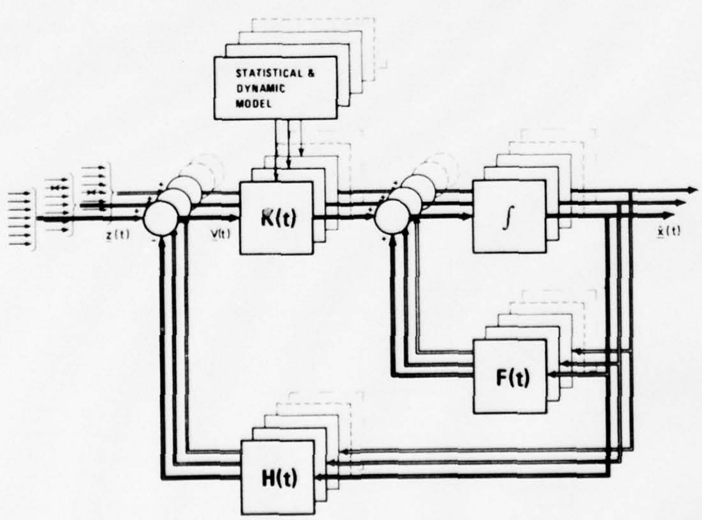

Figure 2 The prinsiple of the failure detection method presented in this paper

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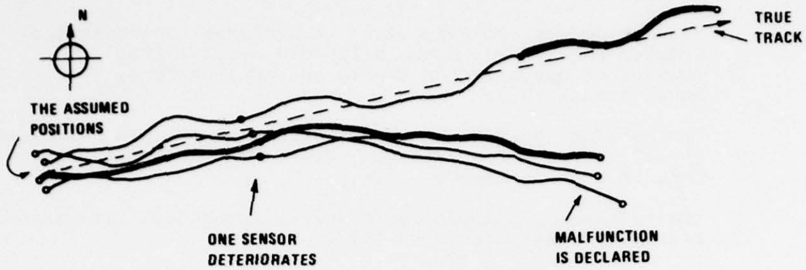

Figure 3 Illustration of the alternate mode capability. The broad line is defined as the best position estimate

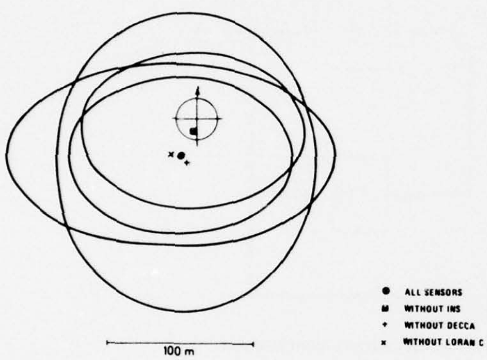

Figure 4 Position estimates and one-sigma-ellipses from the parallel filters in normal operation. (The cross in figure 4, 5 and 6 represents true position)

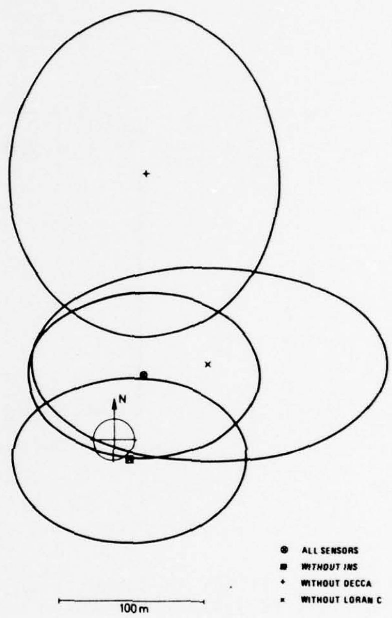

Figure 6 Position estimates and one-sigma-ellipses after a large bias shift in the east gyro of the INS

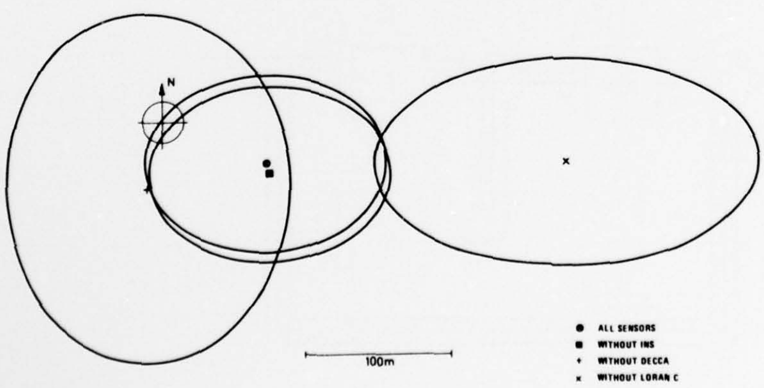

Figure 5 Position estimates and one-sigma-ellipses when Decca has drifted

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CONTROL AND DISPLAY CONCEPTS FOR COMBAT AIRCRAFT

by

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SUMMARY

This paper discusses the need for low pilot workload in future combat aircraft equipped with electronic displays and outlines means by which this may be achieved through optimization of display functions and rationalization of controls. Brief descriptions of current work on head up displays and helmet sighting systems are included.

The combat aircraft of the future will offer the pilot an agile airframe with advanced sensors and displays, a combination which adds up to a greater combat potential than that of current aircraft. However, many current aircraft demonstrably impose very high work loads on the pilot and if the advantages of future advanced sensors and displays are to be realized it is essential that they are less demanding on the pilot's time than are current avionics. The effectiveness of the aircraft will depend on the pilot - machine interface, on the ease with which he can communicate with the avionics through the controls and the efficiency with which the avionics can supply him with the data he needs through the displays. Whereas on previous aircraft it was airframe performance or weapons capabilities that made it a good fighter, future aircraft will probably have only slightly better performance and slightly better weapons, and the real scope for improvement lies in the increase in combat effectiveness which can be brought about by reducing the pilot's workload in terms of systems management thereby increasing his ability to keep his eyes out of the cockpit. Experience and studies show that air combat is essentially a visual, head-up activity and the cockpit should be designed with this upper-most in mind.

Over the past year Marconi Avionics have been engaged in studies into the cockpit and avionics design of the next British fighter. Our studies have shown that the four most significant factors that will influence the cockpit design are:

- Role and mission flexibility, including compatibility with sensors more complex than those on current combat aircraft.
- Reduced instrument panel area together with restricted reach in the high 'g' configuration.
- Greater effectiveness of single crew operation by close attention to an integrated ergonomic design to simplify operation and reduce peak workloads.
- Systems integrity which reflects the differing requirements of mission and flight safety with regard to the controls and information displays. For justifiable confidence in safe operation the pilot requires adequate reliability, redundancy and reversionary sources of flight critical information.

The results of our studies into future cockpits have led us to depart radically from conventional controls and displays in order to meet these objectives, with two fundamental design rules.

Firstly displays must in general be multipurpose and not dedicated as in current cockpits, though there is still a clear need for some conventional single function instruments when it is either uneconomic or undesirable to integrate their functions.

Secondly rationalization of some of the controls of the various aircraft systems is required to reduce pilot workload and improve combat effectiveness, for example, the provision of a single control to arm a particular weapon and simultaneously select the appropriate display modes and weapon aiming program. Some controls will be multifunction some not, according to the mode selected.

These considerations dictate an integrated approach to the cockpit design involving multifunction electronic displays and rationalized controls.

Role and Mission Flexibility

Future combat aircraft are likely to need both air to air and air to ground capability. For example it is probable that the aircraft will have to fight its way in to a ground target, deliver its weapons and fight its way out again.

Although the display and control requirements for these two roles are broadly similar, they differ sufficiently in detail that for optimum effectiveness the displays format should change according to the situation. The integration with the display system of complex sensors, not all of which are required at each stage of flight, also means that display surfaces will have different functions at different times. The displays therefore will be Multi-Function displays.

An advantage of Multi-Function over conventional displays lies in the ability to reconfigure the surfaces to give the pilot the data of highest priority for each mode of operation and to display this data on the most appropriate surface. For example, during the weapon aiming and cruise modes of flight the pilot will be looking out for some 80% of the time. The remaining time will be spent on an intermittent internal scan pattern for essential flight data. This scan pattern should be facilitated by the displays so that as much time as possible can be spent looking out. Collimated displays which are discussed later can reduce the adaption time from outside to inside and effectively increase the outside visual scan time.

However, it is important to strike the right balance between the optimum display configuration for each mode and the high degree of commonality between modes necessary if the pilot is not to be faced with too many different and confusing display patterns. With today's instrument panels it is not uncommon for a pilot to look at instruments and totally mistake their functions. This is especially true when dials are of similar appearance and the pilot is inexperienced on the aircraft. There was for example a spate of altimeter misreadings by 10,000 ft some years ago at a time when new, different altimeter presentations were introduced. It is quite easy to read and misinterpret an instrument that is always in the same place on the panel. It is obviously even easier to misinterpret a display that can appear on different places. Hence it is important that a pilot should not have to work out what display he is looking at before beginning to read it. In general, display formats, e.g. VSD (Vertical Situation Display) should stay the same even when displayed on different surfaces. It may be necessary to alter the size of the display, but the aspect ratio and layout should remain the same and be instantly recognisable. To this end it is also important that different display functions e.g. VSD and HSD (Horizontal Situation Display) should be of different layout and distinguishable at a glance.

One further aspect of Multi-Function displays concerns their flexibility for future growth. New threats, for which new sensors, and new technology are developed, can be accommodated by designing in room for growth at the outset. Data highways, for example, should be able to cope with at least 100% expansion in traffic to cope with future developments and additional sensors.

Reduced Panel Area

Although it is by no means certain, it is quite probable that future aircraft, to give extra 'g' protection to the pilot, will have him seated in a much less upright posture than at present. In addition the need for maximum external vision means that the pilot will be seated higher relative to the airframe or, put another way, more of his body will be visible from outside. The effect of these seating arrangements is that less instrument panel space is available and the pilot will be unable to reach some of the display area unless he leans well forward. Under 'g' he will only be able to reach the forward edge of the side consoles hence it will be necessary to mount some controls remote from the display surfaces. Essential controls needed during combat will be mounted on the stick and throttle. These include weapon release switches, display mode override to the air to air dog-fight mode as well as switches associated with radar operation.

Fly by wire flight control systems will make the aircraft stable when no control inputs are made and the lack of stick movement with this system would allow a control column shaped for left as well as right handed flying to make left handed flying easier, permitting the pilot to twist around further to the right when searching for aircraft in that quarter and also permitting him to use his right hand to make better use of controls on the right hand console. These controls could be operated at times when the aircraft is under low 'g' loadings, for example when low flying. It might be advantageous to put the Multi-Function keyboard on the right console to take advantage of the better dexterity of the right hand.

Single Crew Operation

Tomorrow's pilot will have to undertake tasks for which today's pilot often has the aid of another crewman. Having myself flown single seat fighters for nine years and two seat fighters for seven in the interceptor, ground attack and reconnaissance roles, I have no doubt that the single seat aircraft can do as good a job as the two seater - by day and in good weather. The night all weather role on the other hand, when flown at realistic heights and speeds, currently needs two men and I personally doubt that the avionics industry can ever provide equipment which will totally replace the navigator. That is not to say that a single seat aircraft will be unable to operate by night and in all weather, indeed quite a number already do, the question is how much of an incursion into the traditionally two seater night all-weather domain will sophisticated avionics allow the single seater of the future to make. One thing is certain, the pilot will be essentially the same animal we have today and increased capability can only come about through a reduction in workload to allow him to effectively and safely perform tasks which he can at present only perform in a relatively ineffective and sometimes unsafe manner.

One major area where a reduction of workload can be effected is by the integration of functions. Over recent years an increase in weapon release options to take account of an increased weapon inventory and more modes of release has resulted in complex switching procedures for the pilot and for example, to release a bomb from the Phantom F-4, it is usually necessary to make 9 switch selections before pressing the pickle button. For some automatic release modes a further 2 selections are necessary. To get back to the air to air guns mode 2 switch position changes are needed, to fire a missile, 2 more. These switches require both right and left hand operation and in most cases the pilot has to lean forward and look in, clearly an undesirable situation in combat and under 'g'. For future aircraft the use of pre-programmed weapon packages can reduce the number of selections to one for each mode. Composite air to air guns and missile symbology on the head-up display (HUD) can allow the pilot to fire either guns or missiles as appropriate to his situation in combat without need for mode changes. Helmet sighting systems (HSS) can allow missile lock on and firing without the need to track the target with the aircraft. Integration of functions can markedly reduce the pilots workload and the flexibility of digital systems permits changes to these functions to keep match with future developments in weapons, tactics and avionics.

The guiding principle for display configuration is that because the pilot will operate for much of the time in visual conditions and in a confused and probably hostile airspace he must be able to keep his eyes outside the cockpit as much as possible. I briefly mentioned collimated displays earlier. Apart from the HUD and HSS which are collimated and to some extent complementary in that the HSS can display data such as energy and threat direction when the pilot is looking outside the field of view of the HUD, a further development lies in the use of collimated displays within the cockpit. We have, under development, a display system where a continuous display is achieved by merging the HUD with the next display surface below it - sometimes known as the Head Level Display (HLD) to distinguish it from Head

Down Display (HDD). The HLD is also collimated and, as I mentioned, the optical arrangements are such as to give a continuous display with an instantaneous field of view of some 20^{o} horizontally by 40^{o} vertically. Below this there is still room to fit a further display surface, the HDD. Figure 1 shows the general arrangement.

A future combat aircraft might have up to 7 display surfaces in all, 3 in the centre and 2 more on each side. The 3 centre displays (HUD, HLD, HDD) would be used to carry the top priority data for the mode selected. In the air to air and air to ground modes this would be primarily data associated with target acquisition and weapon aiming, in the cruise, take-off/landing and emergency modes it would be data with flight path control.

The 4 side displays would be for data of secondary importance, this is, data which the pilot needs to have available at short notice for reference purposes but which is not vital for the immediate task in hand.

To some extent display functions would move from surface to surface as modes change. Each surface could have function indicator buttons which are labelled to show the display function entered automatically on mode change (e.g. VSD, DRILLS, RADAR) and pressed to engage optional functions.

In some cases automatic reversion would occur following the failure of another display surface. For example, if the HUD fails, its display could be "dropped" to the HLD and displayed on the view of the outside world normally seen through the HUD but which is now shown on the HLD by means of the HUD videcon recorder camera or VAS (Visual Augmentation System). Because the HLD is collimated, the pilot should be able to carry out weapon aiming with initial target alignment using the HUD standby sight and final weapon aiming through the HLD using the same symbology as that normally displayed on the HUD, but now displayed on the HLD, against the displayed image of the target. With a 1:1 display/real world relationship, the pilots actions in weapon aiming through the HLD will be essentially the same as weapon aiming through the HUD and the fact that the eyes are focussed at infinity means that the peripheral cues of the real world used in spatial orientation during flight are still present and usable. Simulator experience suggest that the peripheral cues are adequate to allow flight at low altitude to continue safely. Normally it is necessary to increase altitude slightly when looking in at instruments during very low level flight, because when the eyes are focussed on the instruments very little is seen of the outside world. However with a collimated display which is adjacent to the HUD, very little of the outside world cues are lost and it should be unnecessary to gain altitude when using this display.

In the air to air and air to ground modes the HLD could display a collimated Visual Augmentation System (VAS) picture. This could be either 1:1 image: real world or magnified by, perhaps, a factor of 10. In the 1:1 case the fact that the HUD and HLD displays are both collimated and give a contiguous display means that a VAS aligned with the HLD would give the pilot a display which in effect "sees through" the nose of the aircraft. This would allow him to engage targets which otherwise would be hidden by the nose. Examples are: a high deflection air to air guns situation or an air to ground attack using very high drag munitions. In both cases the pilot would have both target and aiming symbology displayed on the HLD as a continuation of the HUD symbology. Given the high turning rates envisaged for future combat aircraft it is likely that deflection angles greater than 15^{o}, a typical over nose view, will frequently occur in air to air combat. A system as described will allow the pilot to pull lead and fire under conditions where the limits of over the nose view would normally force him to delay his firing until he had worked off some of the angle-off.

Firing opportunities are fleeting in combat and the ability to fire with day 25^{o} lead angle will not only increase the number of opportunities to fire but also allow the pilot to fire on occasions when the hostile pilot will probably not consider it possible. For combat manoeuvring, the increased over the nose field of view will allow lead pursuit tactics to be imploved as opposed to lag pursuit tactics with a consequent decrease in time to firing.

In the magnified case, a HUD symbol would show the field of view of the VAS which would now be aligned with the HUD or missile boresight rather than with the HLD boresight as in the 1:1 case. Tracking a target with the HUD symbol would produce a magnified image of the target on the HLD for target recognition purposes. The VAS could thus be a simple two position, two field of view device with a 1:1 unmagnified view of the HLD boresight area and a, say, 10:1 magnified view of the HUD boresight area. The degree of magnification would depend on such factors as resolution, screen size, fields of view and the ease with which a pilot can track a target with the HUD and still be able to glance into the HLD for long enough to recognise what he sees. At the sort of range where unaided air target recognition is impossible, crossing rates tend to be low and this plus the fact that both displays are collimated should ease the pilots tracking task.

In the non-weapon aiming modes, the HLD would normally act as the Vertical Situation Display (VSD) allowing reference to essential flight data without the need for re-focussing of the eyes. The VSD can be a more comprehensive display than is desirable for clutter reasons on a HUD; it can be a display optimised for instrument flight and could for example include engine power settings (a parameter essentially related to vertical situation and considered to be of flight-critical status).

As part of the aim to give the pilot easily monitorable attitude information, the HUD and VSD attitude displays would come from different sources. Any difference would be resolved by the reaction to pilot control inputs or by a cross check majority vote with a third independent reversionary attitude display.

The VSD would be displayed on an adjacent surface to the HLD during weapon aiming modes. The VSD format would remain unchanged despite the change in location so that the pilot does not have a contend with a different scan pattern on the display even if the inter-display scan has altered.

HSD The HSD is of a lower order of importance than the VSD hence it can be displayed on a lower priority surface. This display would give navigation data as called up on the navigation keyboard and could also be the display for that keyboard, that is, it would show data entry as it occurred. In addition to pure

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Figure 1 Display Arrangement

navigation data, way points, time to go etc., fuel quantity, flow rate and requirements for the route, being essentially horizontal data associated with distance and time, could be displayed on the HSD. Preset Bingo fuel states and steering data could, to aid the pilot's scan, be displayed on the HUD as well as the HSD.

HDD The HDD would display radar in the air to air mode and map in all other modes. However, to cater for processor/waveform generator failures, it could also display other functions when required.

DRILLS A major advantage of electronic displays lies in the ability to display tabulated drills as a routine or in emergency thereby obviating the requirement to thumb through Flight Reference Cards. This aspect is later elaborated under the discussion of the Emergency mode.

SYSTEMS In all modes one surface would display systems data. This includes armament state in the air to air and air to ground modes and control position data in the take-off/land mode. System malfunction/serviceability data would be automatically displayed in the emergency mode. The various sub-systems, for example all services operated by or in conjunction with engine bleed air could be called up on the keyboard and their serviceability listed.

ECM/ESM Due to the importance of electronic counter and support measures, a display surface could be dedicated to ECM/ESM and, possibly, communications management.

Emergency Mode

A criticism often levelled at electronic displays is that it is impossible to display as many aircraft systems parameters as are currently shown on conventional instruments. In general, this criticism is valid during the operational modes, but if an aircraft emergency occurs (other than loss of display!) and this emergency precludes continuing with the mission, then the flexibility of electronic displays allows a reconfiguration to give displays optimised to help the pilot cope with the emergency. For example, if the aircraft was in the air to air mode and an emergency occurred, a warning could occur on the HUD and VSD and the nature of the emergency would appear on the Systems display. Selecting Emergency mode would free some displays of mission dedicated data and these surfaces could then give text giving immediate and subsequent actions together with the further ramifications of the emergency and the impact of the failures on other systems. Having more than one surface for drills would allow the pilot to cope with more than one failure when the initial failure may cause others. For example, in some aircraft a hydraulic failure can affect a number of other aircraft systems and lead to emergency situations more complex than those normally associated with a simple loss of hydraulic pressure. While it is hoped that future aircraft will not have this type of cascade failure, it remains a possibility. The flexibility of the display surfaces to reconfigure in a manner optimised to cope with an emergency should help overcome the justifiable reluctance of pilots to have important parameters not continuously displayed during normal operations.

Integrity

To gain acceptability, not least by the pilots who will fly them, it is necessary to design electronic display systems to a very high level of integrity. Electronic systems do fail from time to time and it is essential to build in safeguards so that these failures do not leave the pilot with a lot of blank screens. One method employed is to keep flight critical data separate from mission critical data since the consequences of the former failure are more catastrophic than that of the latter. At the same time, it is important to have redundancy in the mission dedicated parts of the system so that mission effectiveness is not lost in the event of simple failures. The various methods of ensuring integrity range from simple to extremely complex and are outside the scope of this paper.

To cater for major failures caused, for example, by engine flame-out when it is unlikely that sufficient electrical power will remain to drive the complete display suite, it will be necessary to provide some form of stand-by, "get you home", instruments. These will help overall integrity of flight critical displays because they can be kept separate from the main displays. They could use the flight control system computers as data sources since the flight control system integrity must of necessity be higher than all other systems.

An additional consideration is that CRTs normally require 20-30 seconds from switch-on to giving a readable display. With the accent on dispersion of aircraft on the ground during hostilities, it is likely that future aircraft will require a ground start facility using internal DC power. Even if invertors can supply power to a limited number of display surfaces, the fact remains that the pilot will have to wait 20-30 seconds before having a display good enough to use during the engine start procedure when he must closely monitor revs and temperatures. Some form of reversionary, non CRT display will thus be necessary to cope with the situation since the pilot should not have to wait for this period before starting after the receipt of a scramble order.

Whether these reversionary instruments are conventional or low power electronic will depend on the progress made with low power solid state displays which at present are at an early stage of development.

Controls

The prime requirement regarding controls must be to make the pilots task as simple as possible given the fact that he may have to operate a complex system while fighting for his life. What may, in isolation, seem an easily controllable system may be virtually impossible to operate under stress when the pilot has additional simultaneous tasks and dare not take his eyes off his adversary to look in at controls.

Where possible one switch should carry out many switching actions, such as the actions needed to give the correct attack display and weapon aiming solution when the weapon package is selected. Where possible switches should be easy to operate without the necessity to look at them. If a bank of switches should always be on in flight they should be electrically or mechanically ganged so that they can be put on with one action, but may still be selectively turned off following a malfunction. The switches should be grouped such that those that need to be operated in combat should be close to hand while those that do not can be grouped in less accessible areas but still in logical and subsystem grouping. This contrasts with current cockpits where all of the switches associated with a particular total system tend to be grouped together by systems rather than by functions, apparently without regard to whether or not they are frequently operated in flight.

Some data lends itself to insertion by multifunction keyboard. Examples are navigation co-ordinates, weapon packages. Some data is best entered by discrete buttons. Examples are flight modes, display surface functions when different from those automatically entered. Some data has to be entered under very difficult conditions. Examples are radio frequency changes when flying in close formation in cloud, mode changes under high g conditions. For radio frequency changes, an electronic display of the actual frequency selected combined with discrete rotary knobs for each digit of the frequency is probably the best compromise. As the most likely mode change under g is to air to air from any other, there should be an air to air override button on the stick.

Where multifunction keyboards are concerned, single pass data entry is desirable. It should be possible to interrupt program changes necessary in flight, such as insertion of new navigation co-ordinates, in order to use the keyboard to change the display of navigation data, and then resume the program change at the point of interruption.

The emphasis must be on fitting the system to the pilot rather than vice versa. For example, from the systems point of view it may be more convenient to address many sub systems from one multi function keyboard, but this will invariably result in greater complexity from the pilot cognitive information processing standpoint than addressing the sub system from a dedicated control.

The controls, therefore should be a mix of multi-function and dedicated with the proportions of the mix decided after comprehensive functional allocation analysis of the pilot task on simulated missions including such factors as, for example, frequency of use, likelihood of simultaneous tasks and reach distance/time considerations. One method which allows switch locations to be readily changed is to incorporate them in groups as LRUs. These LRUs would communicate with the systems they control through the mission data bus. Ideally the results of the simulator analysis should be reassessed on prototype flight tests and it should be possible to alter control layout then or when squadron experience or a change of squadron role shows this to be desirable.

A further consideration is that the aircraft may have to be airborne in a very short time initiation of a scramble order. Over recent years, the inclusion of inertial systems has increased the time it takes to get aircraft off the ground. Compared with, for example, the Hunter or Lightning which could be airborne in 1 - 2 minutes, modern inertial equipped aircraft need at least 5 minutes. If the pilot of the future aircraft has to store mission data and switch on systems through a multi-function keyboard as well as aligning an inertial system his reaction time may well exceed 5 minutes. Swift reaction from ground alert is needed to meet a low level air threat or a call for support from the army, and the extra time may make all the difference between mission success or failure. To take one system, communications, it is obviously quicker to have frequencies manually preset before the mission than to have to insert them into memory before take-off. Similarly once airborne it is quicker to change channels using one knob than to address communications, insert the channel and enter the data through a multi function keyboard.

To summarize, the choice of control for a system should be made following analysis of the task with the requirement of low pilot workload uppermost in mind. This workload can occur before flight when in a scramble situation and in flight when in combat and the control system design should therefore be centred round the pilot.

Display Types

HUD. The current generation of Head-up Displays have two major limitations.

- The Instantaneous Field of View (IFOV), that is the amount of the total FOV that the pilot can see from a given eye position, is limited.

- The amount of prime instrument panel space, that is, space directly in front of the pilot, taken up by the HUD is too large.

To overcome these two limitations the multicombiner HUD was developed. This proprietary system utilises four reflective combiner surfaces set in a glass block to build up the vertical angle of the IFOV and increases the horizontal angle of the IFOV by a unique optical arrangement shown on Figure 2 which allows the reflectors to be mounted closer to the pilots eye. In addition, the optical arrangement by dispensing with a folding mirror and refracting the CRT image into the combiner block results in a display unit of very little vertical extent, and considerable consequent saving of instrument panel space.

The rays of light from the CRT pass through a truncated but otherwise conventional HUD lens system to the base of the combiner block where they are refracted upwards and then reflected to the pilots eye. By employing coatings of different reflectance, 25%, 33%, 50% and 100%, for the four reflector surfaces, a display of even brightness is seen by the pilot. Each surface reflects a quarter of the IFOV and these four portholes are contiguous with no overlap or underlap hence the pilot sees a continuous display of approximately double the vertical FOV given by a single combiner HUD. The outside world picture is also of even brightness being built-up from multiple reflections but in a somewhat different manner. Because the combiner block is a prism the lowest ray of light from the outside world to the pilots eye is refracted downwards, as shown by the $-12\frac{1}{2}^{o}$ apparent vision line on Figure 3, on entering the block and exits on the line marked X so that the display unit metalwork, to the pilot, has an apparent vertical dimension of X-Y, the portion above X being invisible. This allows the display unit to be mounted higher than otherwise possible without obscuring the over nose view. The unit can also be mounted closer to the pilot without impacting the ejection line resulting in increased horizontal FOV.

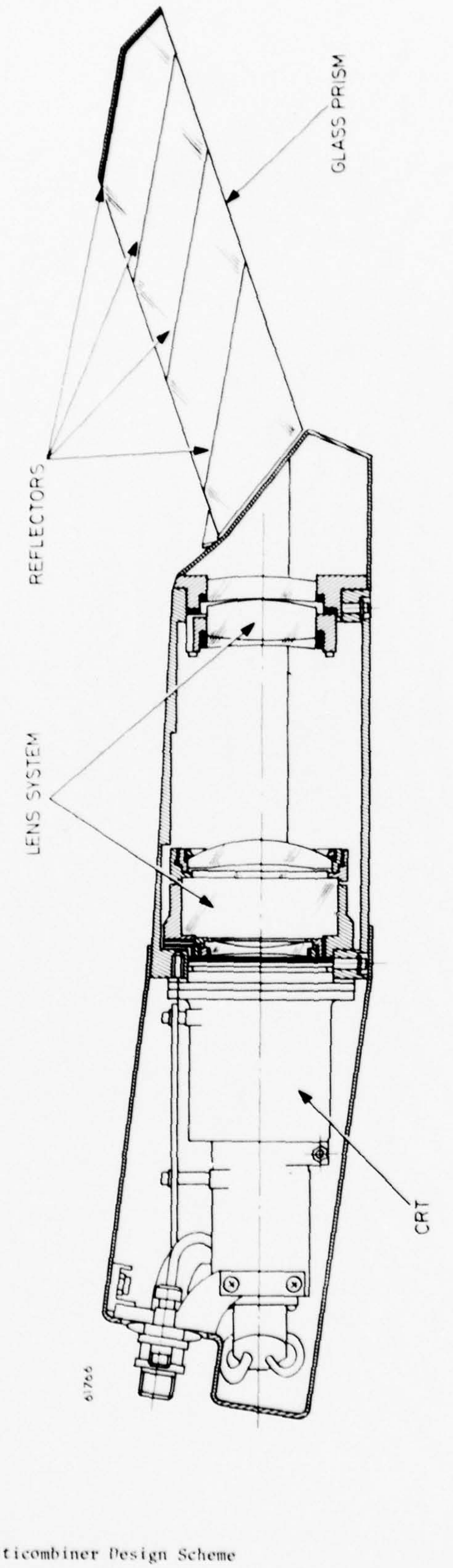

Figure 2 Typical Multicombiner Design Scheme

12-8

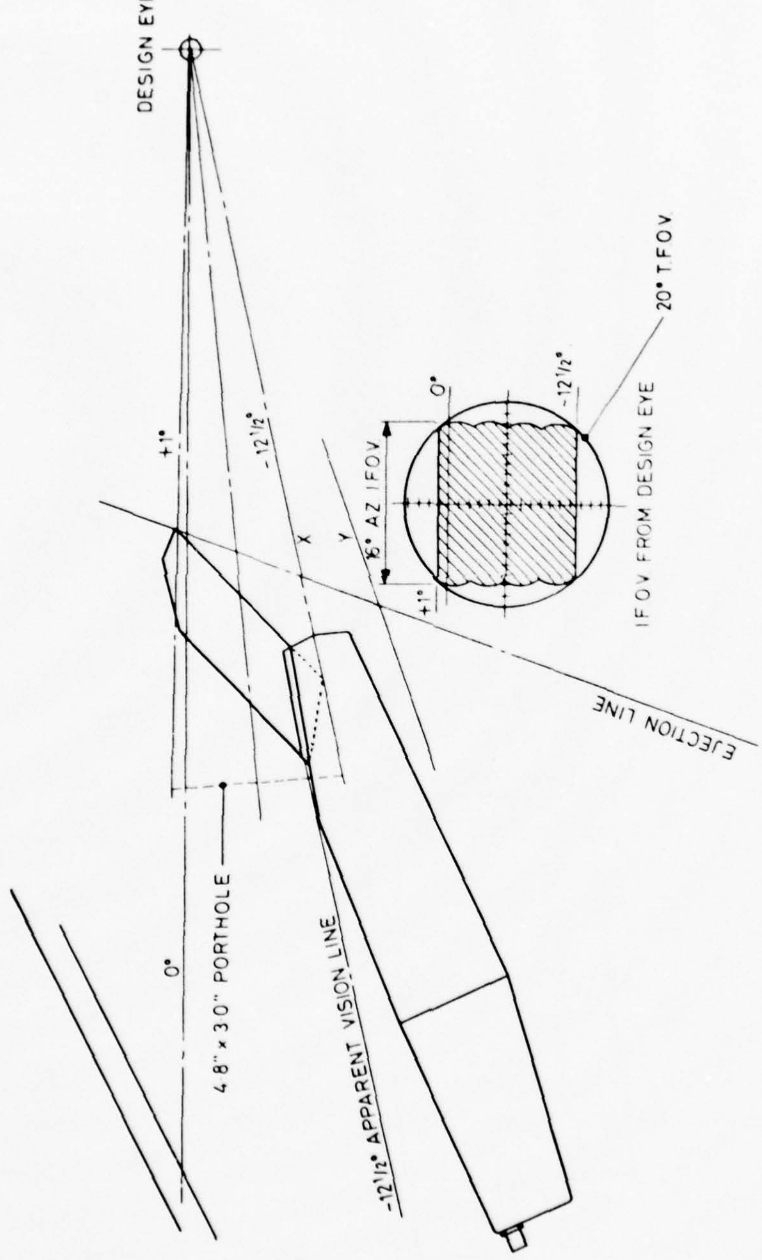

Figure 3 Multicombiner HUD

The multicombiner display unit which will be the subject of two separate flight evaluations in 1978 is considered to be the most cost effective approach to the large fields of view needed for future aircraft.

HLD. *The space thrown up below the HUD can be filled by a Head Level Display.* This can be viewed through a collimating lens system which, in addition to the advantages outlined earlier of being in focus with the HUD and outside world, also, because it is subject to binocular vision gives increased aximuth FOV. Furthermore, it is possible to build an exit optic system such that the vertical FOV is also increased and made contiguous with that of the HUD. The operational advantages of this have already been described. Combining the HUD and HLD into one unit can be of advantage from the maintenance stand-point and the units can share a common mounting tray etc.

RASTER DISPLAYS. The HLD in common with all other display surfaces except the HUD would be an all-raster display for reasons of flexibility and power dissipation.

An all-raster display unit has less complexity, less power and thermal dissipation together with higher reliability than a mixed calligraphic/raster display unit. Symbology for tabular or over-lay video formats can be generated in a suitable raster - compatible waveform generator, providing ready means of matching brightness and position of the video and symbol overlay. The format design flexibility is aided by contrast inversion, line thickness and shading facilities. This approach is made practical by the development work carried out by the Company into special techniques to eliminate staircase and flicker effects which can otherwise be present in raster display of dynamic symbology and near horizontal lines. The legibility and visual quality of raster generation symbols using these techniques are comparable with those of stroke written symbols.

These techniques are used in the MRCA TV TAB and NIMROD Acoustic Display systems. The high contrast of the displays is provided by the use of a green CRT phosphor in conjunction with a matching broad based colour filter which has an anti reflection coating on the front surface to minimise ambient light reflections. The filter attenuation is chosen as a compromise value giving the highest contrast ratio consistent with maintaining an adequate absolute brightness level through the filter.

Colour would give greatly enhanced ability for coding, maps and visual decluttting of the displays and the improvement could arguably be as high as that of a domestic colour TV compared with black/white. Red displays would assist night visual adaption, a factor long realised for HUDs. However, of the different methods currently employed for generating colour displays; multi gun shadowmask, single gun beam-indexed, and penetration phosphor, none are wholly satisfactory for military roles where ruggedness and high brightness for day time use are concerned. Nevertheless it is quite possible that colour displays will be seen on future aircraft and will realize the full potential for data display of electronic surfaces.

Helmet Display Sight System

In combat the pilot will wish to look out for most of the time so as to be as aware as possible of the actions of the other aircraft involved in the fight. A Helmet Sight System (HSS) can provide him with essential mission data while he is looking out and unable to refer to his cockpit displays. Essential mission data could include:

- Missile aiming and lock data.
- Electronic high priority threat warning.
- Energy rate.

The Marconi Avionics helmet system consists of a LED display sub-system and a helmet position sensing sub-system using LEDs and CCDs.

The display sub-system uses a 10 mm square matrix addressable 32 x 32 LED array the image of which is reflected and refracted within a prism mounted just above the upward limit of the pilots sight line on the front of his helmet. After leaving the prism the light rays are reflected to the pilots eye by a collimating combiner which may be part of the inner surface of the helmet visor or a separate component inset into the visor. The collimating display will subtend an angle of 7° at the pilots eye and a perceived brightness of 1500 ft lamberts can be achieved, which is controlled by an autobrilliance sensor.

The helmet position sub-system shown in figures 3 and 4 consists of a triangular set of LEDs on either side of the helmet which flash sequentially. The rays of light from the LEDs pass through V-slits onto multi-bit high resolution CCDs where the rays are sensed in two places. The distance apart of these two places essentially gives the elevation and the distance along essentially gives the azimuth of the line of sight from CCD to LED. The triangular set gives 3 lines of sight which are geometrically related to the direction in which the helmet, and hence helmet sight, is pointing. A microprocessor based computer performs line of sight calculations, controls the LED set and display symbology. Initial calibration is obtained by a boresight procedure in which the helmet sight and HUD boresight are used to give a datum from which the helmet sight line of sight is then calculated.

The helmet mounted equipment adds only 0.2 kg to the existing helmet weight and the system copes with an angular coverage of ± 180° in yaw and ± 70° in pitch. Such systems are at a late stage of development and have been subject to several successful flight test programmes. LEDs were used for the display in preference to a CRT because of weight, robustness and low voltage advantages. This system considerably increases the volume of sky in which air to air missile engagement is possible and by giving the missile some of the manoeuvring task that the aircraft would otherwise have had to make to achieve missile lock, earlier engagement is possible.

12-10

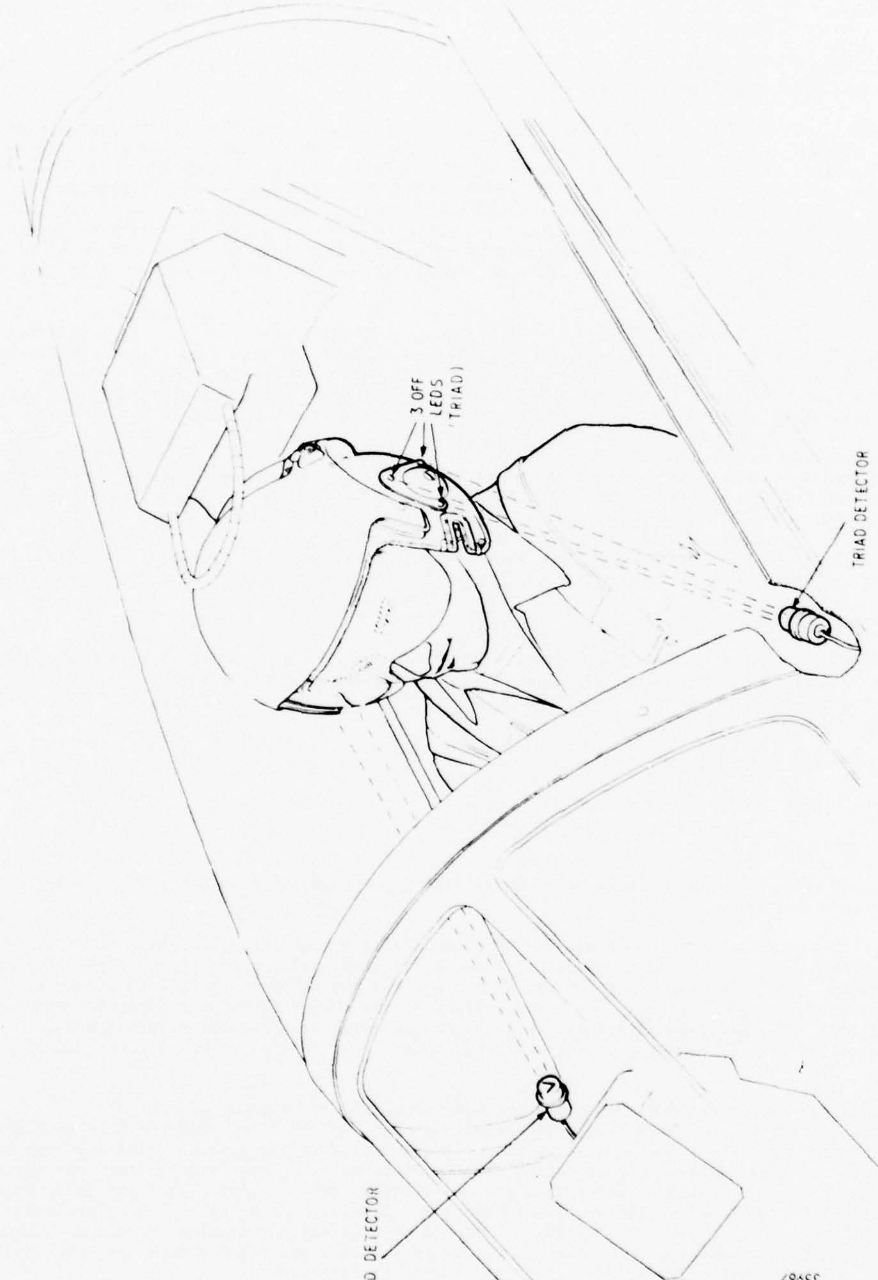

Figure 4 Typical Helmet Optical Position Sensor System (HOPS)

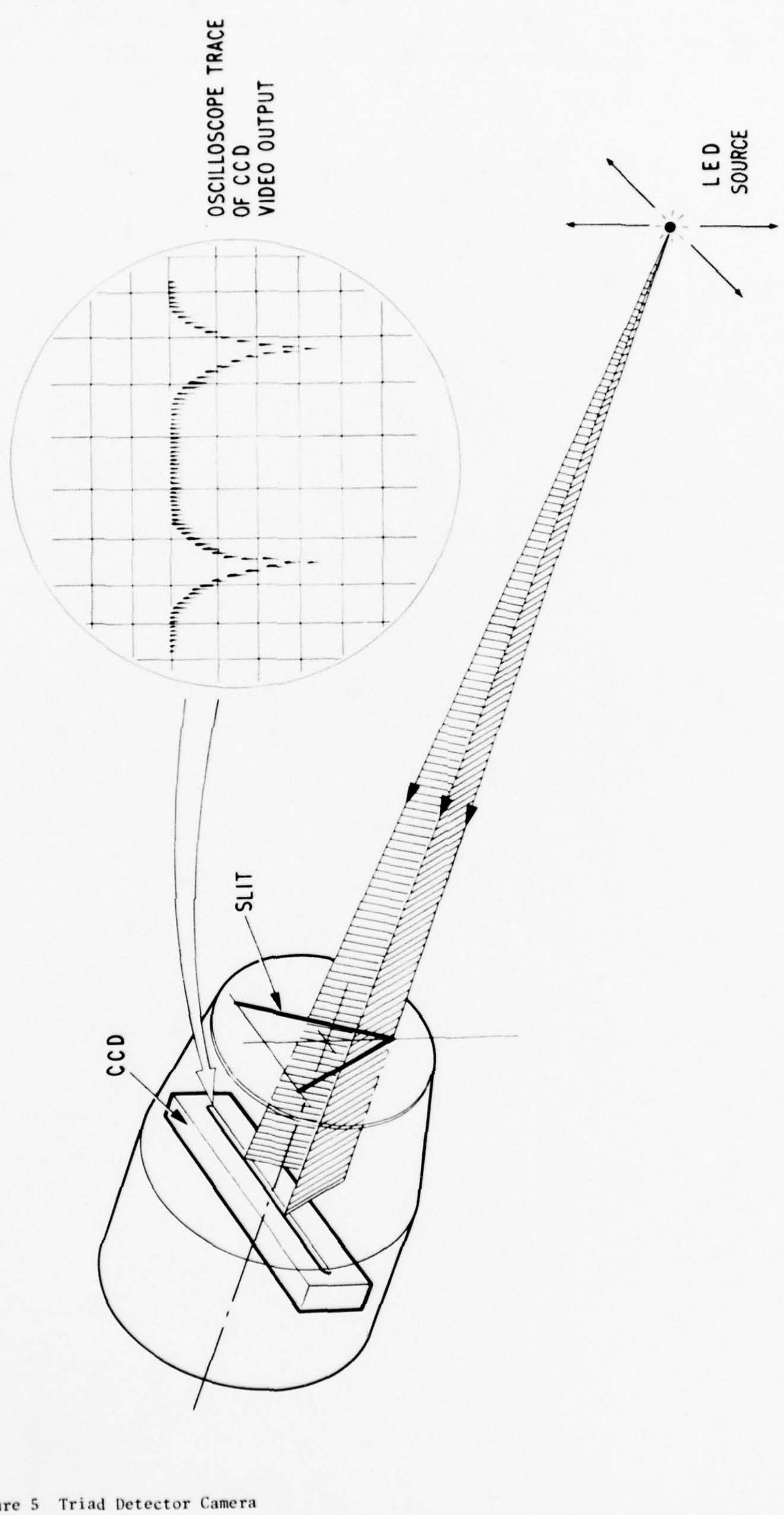

Figure 5 Triad Detector Camera

Summary

To summarize, the cockpit of future combat aircraft must be designed from a pilot's functional standpoint. Given careful design and the ability to reconfigure both displays and control layout to cater for mode and future role changes, our studies indicate that electronic displays will enhance the ability of the single seat pilot to make effective use of the advanced systems and agile airframe envisaged for future combat aircraft.

Figure 2 The principle of the failure detection method presented in this paper

AN ADVANCED NAVIGATION DISPLAY AND ITS EFFECT ON SYSTEM DESIGN

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SUMMARY

Future cockpit designs will aim at single crew operation with a high degree of display integration. The subject of the paper is an advanced navigation display which would normally be located immediately below the Head Up Display in order to show the horizontal situation.

The paper draws attention to some of the operational requirements and then describes the COMED Combined Display.

The display combines a full colour topographical map based on 35mm film with a multi-mode electronic display capable of accepting cursive or raster information.

Also described is a planning aid permitting horizontal flight profiles to be digitised automatically in a briefing room from a conventional paper map. Flight plan processing is possible and the resultant data can be loaded into a portable store which is then inserted into the aircraft system so that the planned profile can be superimposed on the topographical map.

The display concerned has been fully developed and demonstrated in flight and the planning aid has completed successful operational trials.

1. INTRODUCTION

The environment of a single-seat cockpit is particularly exacting in terms of display design for a number of reasons. There is a heavy pilot workload caused by the need to handle the aircraft, its major systems and the weapon system. The pilot of a single-seat aircraft flying at a very low altitude has to contend with turbulence, flight hazards present near the ground and threats posed by enemy defences or other aircraft. The canopy design is usually such that displays are subject to very large variations in ambient light ranging from direct sun by day through to night operations. If sensors are used to facilitate operating at night or in low visibility, their information must be displayed and assimilated.

While automation and correct system design can mitigate some of these problems, the essential need is for better displays. The most important display surfaces are the Head Up Display and the situation display immediately below it. The latter must have an electronic multi-mode capability but because the aircraft operates in close proximity to the terrain and as the pilots are accustomed to the use of topographical maps, a full colour map display is extremely attractive.

The paper describes the COMED Combined Electronic and Map Display together with its incorporation in an operational system designed to simplify navigation through all phases from pre-flight briefing through take-off to touch down.

2. OPERATIONAL AND SYSTEM REQUIREMENTS

Some of these have already been dealt with in the Introduction. It is necessary to make some assumptions about the sensors and other equipments to be carried in the aircraft since their capability to generate information determines the display requirements.

Accurate navigation is of prime importance. It can contribute to the effectiveness of the mission by enabling the pilot to select and fly a horizontal profile which avoids known defences and exploits terrain screening as much as possible. The requirements become more stringent as the target acquisition phase is entered, particularly if the pilot has to depend on the relatively narrow window afforded by a forward sensor.

Navigation is also involved in determining the protection available against collision with the ground while flying low. A sophisticated Terrain Following System can give full protection. Alternatively, as pointed out elsewhere in this Symposium, it is possible to consider simpler sensors such as a laser used as a low flying terrain avoidance aid. The dependence which can be placed on such systems is limited by the fact that they are essentially single channel and can be vulnerable to a single failure. The back-up consists of careful navigation planning and a good navigation display which can provide the pilot with information enabling him to decide how to modify his vertical profile as a compromise between additional vulnerability due to flying higher and the risks involved in continuing low flight visually, or with the aid of a visual sensor. This is an extension of the philosophy of a safety height used in simple aircraft.

Navigation/...

Navigation workload can be greatly reduced by a combination of careful planning and effective display.

13-2

The source of navigation information is likely to be an inertial navigator. There are several ways of enhancing the accuracy of such a system. There is some scope for performance improvement where it is permitted by the potential accuracy of the instruments, usually by data processing and more sophisticated methods of pre-flight alignment. In the future, the inertial data in some systems will be mixed with highly accurate position and velocity information from future aids such as Navstar. But whatever the means selected, updating navigation using known features in conjunction with forward sensors such as Ground Mapping Radar and Electro-Optics will always be attractive as a means of improving accuracy or increasing integrity. The display system should permit this.

It is also advantageous to have a degree of redundancy to cover failures and the combination of separate moving map and multi-mode electronic display elements permits this.

3. DISPLAY DESCRIPTION

The most important properties of a highly readable display are image brightness and contrast. When flying head up, the pilot may be scanning a bright background such as cloud and the display must then be bright enough to be legible when he transfers his attention to it, even if his eyes are slow to adapt to the new light level.

In some cases, direct bright sunlight can shine onto the front face of a display, and the contrast of the displayed image must be sufficient to maintain legibility. This is particularly true when full advantage is taken of the colour and content of a topographical map. The resolution of the image, the viewing distance, the effects of vibration and the basic suitability of the maps or other data displayed are also important factors.

Figure 1 shows the basic optical principles adopted in the COMED Display. A projection lamp, together with a condenser and a reflector, is used to generate a full coloured map image from data stored on 35mm colour film. The image-forming light rays from the projection lens are reflected by a mirror onto the image-forming screen.

A combination of CRT and Map images is presented to the operator via a viewing system consisting of a combining mirror, transfer lens and field lens. As a result, the map and CRT images appear superimposed on the same plane.

An important feature of the design is that it generates an exit pupil of a size only sufficient to cater for pilot head movement.

This method of optical combining and a field lens was selected against other techniques such as the rear port tube because of a number of significant advantages. Ambient light cannot fall on the CRT or map images because to do so, it must pass through the exit pupil, which is almost completely filled by the pilot's head. The brightness of the primary image approaches the brightness of the full size image and is almost five times brighter than if a full size image screen were used. For an equivalent drive power, twice the brightness of a full size (i.e. six inch diameter) tube is achieved and four times the brightness of a rear port tube.

The use of a smaller CRT makes the electron geometry easier, resulting in a smaller spot size. The tube and its associated drive circuits can be made compatible with cursive or raster writing techniques and raster information from sensors so that the display can be integrated into any probable configuration.

The manner in which the film is handled has resulted from long term development. A relatively low powered 50 watt projection lamp is used and three are carried together with an automatic change mechanism. The film transport is highly accurate and has a high slewing rate so that changes taking place when a film strip boundary is approached appear near instantaneous to the pilot. A glassless film gate deals with the problems of film damage experienced with some earlier displays. The display is aimed at high maintainability and is completely modular.

35mm film is a most cost effective means of storing full colour map information. A typical unit can accommodate 57 feet of film at a map scale of ½M and a reduction factor of 15. This is equivalent to an area of 2000 by 2000 nautical miles, in excess of the whole of North America. Since the mean film speed is 5.7 feet per second, the maximum access time is 10 seconds.

The earliest combined displays were relatively bulky and were tailored individually to aircraft cockpits. The latest version shown in Figure 2 has improved optics giving a near rectangular screen and an installation outline aimed at single-seat cockpit.

4. PLANNING SYSTEM

A comprehensive mission planning system has also been developed. It is described here in conjunction with the display but it is in fact suitable for use with any digital navigation system.

Autoplan consists of a table on which a paper map or chart can be placed in the briefing room. A hand-held cursor can be placed anywhere on the map and is connected by a single table to the electronics, which include a processor and a small printer.

The pilot places his map on the table and initialises the digitising system by inserting the coordinates of two known points. He then uses his tactical judgement and the best available briefing information to choose a best route, which he marks in pencil on the map as a series of straight line sectors. The cursor is then moved through the profile from one turning point to another, a button being pressed at each to cause its coordinates to be transferred to the processor. Keyboard entry of fixed parameters/...

parameters such as ground speed and fuel load is provided. As the route is entered, the printer prints out the various sectors and gradually accumulates a complete flight plan including times and fuel. The system is extremely accurate and a 20 turning point flight can be programmed in less than 5 minutes, including separate treatment of IPs and targets. Automatic conversion from the UTM grid to lat/long coordinates or vice versa is possible and additional copies of the printout can be generated on demand. An overlap profile from one map sheet to another can be handled.

This system has gained rapid pilot acceptance in its trials. Tactical choices can be made because the time to compute a single route is so short that alternatives can be compared for timing and fuel implications. Further copies of the flight plan can be printed out if a number of aircraft are to operate in a group.

The PODS (Portable Data Store) facility is a further system refinement. The small circular store contains a self-supporting memory and is plugged in to the ground equipment so that the flight plan can be recorded in it. The pilot then carries the store out to the aircraft together with his map and other briefing information where he inserts it into a special fixture so that the airborne computer is loaded automatically. Where the combined display is fitted, the pilot can call up the route in the form of a full colour topographical map with his planned tracks over-written electronically. Where he has stored optional tracks for use in a mission or in the case of a diversion, he can recall and examine these options in the cockpit before committing them to the guidance computer. Clearly, any other information which can be expressed as lines or electronic symbols can be stored as part of the briefing process and retrieved on the display. If the aircraft is fitted with forward sensors, points at which there are features particularly suitable for checking progress can be included in the plan.

The essence of the Autoplan/PODS concept is to enable more sophisticated planning of missions while reducing the workload by a system of automation which extends from briefing through to touch-down.

5. CONCLUSIONS

The optically combined CRT/Moving Map display is now a well developed technology and highly suitable for providing a Horizontal Situation or Navigation Display in either a two-seat or single-seat cockpit.

Electronically generated information including information from sensors or computer symbology can be overlaid over the map. Alternatively, the display can be used as a full electronic multi-mode display. It is therefore suitable for integration into a weapon system.

The display uses a field lens viewing system which is a highly successful means of offsetting the effects of ambient light and optimising brightness and contrast as required in this type of cockpit. It has been shown to be a satisfactory means of displaying information at very low light levels for night flying and its intrinsic qualities make it generally superior to a pure CRT display.

The addition of a planning aid can automate the complete process from briefing to touch down by using automatic digitising and data processing on the ground together with the transfer of briefing information to the aircraft using PODS. There are many advantages in more sophisticated mission planning including the ability to exploit the terrain fully and a reduction in cockpit workload in the air.

A display technology has developed on the basis of a combined map/radar display which is now in production. Further development has produced versions aimed at advanced single-seat applications in which the electronic component of the display is used multi-mode and forms part of an integrated display system.

The Author wishes to thank those colleagues who have assisted in the preparation of this paper, particularly Mr. W.M. Aspin, who is the Chief Engineer of the Group responsible for these developments. He wishes to note the part played in these developments by officials of the UK Ministry of Defence and the Royal Aircraft Establishment. Thanks are due to the Management of Ferranti Limited for permission to publish this paper.

13-4

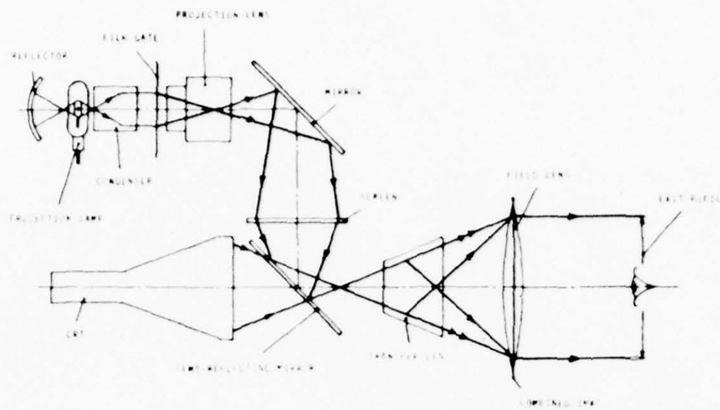

Figure 1
COMBINED DISPLAY OPTICAL SYSTEM

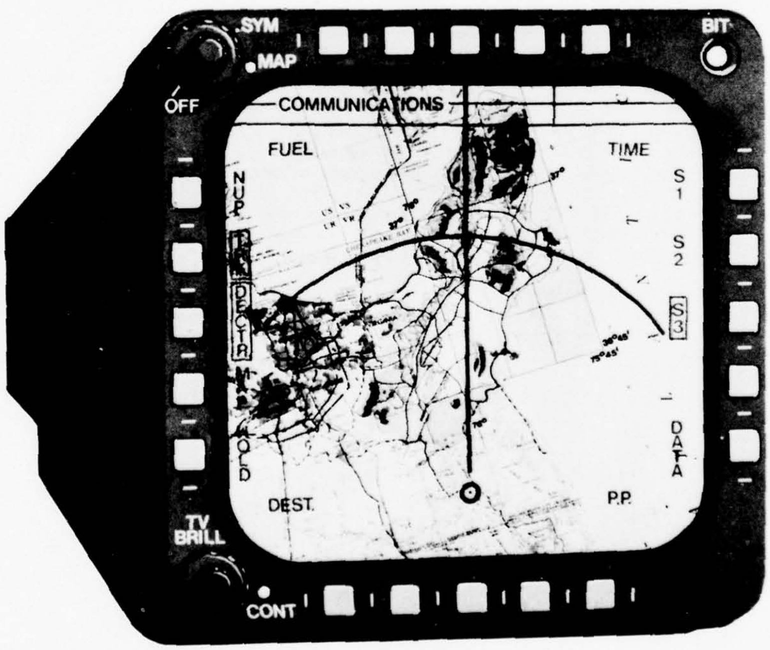

Figure 2
FORMAT FOR SINGLE-SEAT COCKPIT

METHODS FOR THE VALIDATION OF SYNTHESIZED IMAGES IN VISUAL FLIGHT SIMULATION

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SUMMARY

To validate the information content of synthetic visual flight simulation, objectively based methods and criteria are necessary which can show the influence of a variety of visual cues on pilots' perception. 56 pilots and 28 non-pilot enlisted men made height and distance judgements from landing approach scenes with different levels of detail. Some judgements of height and distance were made in relation to a previously shown standard scene. Other judgements (absolute) were given in ft or m . To evaluate the influence of scene information simplification on subjects' perception of height and distance, a number of measures were made including judgement time, error and the exponent of the stimulus-response relationship $R \sim S^n$. Judgement error and the exponent of fitted power-functions both were significantly influenced by scene stylization. The increase of judgement error and the decrease of power-function exponent respectively are more distinct when making absolute judgements than relative ones. Because results for pilots are quite different, non-pilots should not be used as subjects for visual research work with landing scenes.

1. INTRODUCTION

The advantages of using simulators in the field of vehicle guidance and control are unquestionable. For cost and safety reasons simulators are increasingly used for education and training, especially for pilot training. With both whole and part-task simulators a realistic full simulation is frequently necessary for the correct performance of complex operations. In addition to a realistic cockpit and realistic motion and noise simulation the simulation of a relevant outside view is frequently necessary for good flight simulation.

The demand for a wide range of simulation applications together with continuous advances in engineering has led to the development of Computer-Generated-Image Systems (CGI) [1] which have certain advantages in visual simulation over the well known film and model systems. Despite these advantages CGI systems suffer from the disadvantage of limited capacity and computing speed which severely restrict the amount of visual information which can be simulated and displayed. Since there is no possibility at present to fully simulate the natural outside view with these limitations the problem becomes one of selecting only that visual information which a pilot actually needs to perform his special tasks, i.e. which visual cues must be presented to obtain a good visual flight simulation.

Previous research in visual perception, has in most cases, used subjective methods. It has been tried, for example, by questioning pilots to reveal which of the external cockpit visual cues provide them with distinct information about distance, attitude, altitude, velocity and so on. Apparently such an approach will be somewhat inaccurate because the perception of most experienced pilots and the performance of many aspects of their tasks are unconscious. Therefore it would be impossible for them to give precise answer about the influence of special objects or elements in their visual perception on their visual judgement performance. That is the reason why scientifically correct statements about the requirements of visual scenes in flight simulation can not yet be made [2, 3].

2. RESEARCH OBJECTIVES AND VARIABLES

Because of an increase in the use of computer generated image systems for flight simulation there rises the question if such a stylized outside scene, which has limited visual information content, is sufficient for flying an aircraft. To answer this question it will be necessary - if possible - to separate out the influence of the visual information provided by the different objects, details and structures. For this purpose one needs appropriate objective methods which allow specific conclusions about the influence of visual information on pilot's perception.

Because of the large amount and variety of visual cues in a real environment , which increase in significance with vehicle motion (motion-parallax, streaming-patterns) first experiments dealt only with static external scenes. The information requirements of external visual scenes are strongly dependant on the specific task to be performed. Since pilots during the landing approach depend heavily on the use of visual information in the external scene, this flying phase was chosen for research purposes.

During the landing approach the pilot's perception of height and distance is of great importance. Our goal was to measure the influence of image stylization of landing approach scenes on pilot's perception of height and distance by means of two different estimation tasks.

An evaluation of the results should give hints to how useful these tasks are when testing visual simulation systems for various tasks or missions.

One estimation task was the making of "relative judgements" of the presented stimuli which seemed to be a straight forward task appropriate to a landing approach where the pilot in most cases checks his position in relation to the required glide-slope on the basis of a well learned model of runway shapes corresponding to various above, correct, and below positions at different relative distances [6].

Since it is likely that some pilots in certain situations, especially military pilots, may have to make "absolute judgements" (although not necessary of height and distance), it was decided to ask subjects to make absolute estimates during the second part of the program.

A comparison of research results between the two types of estimates may help to answer the question which task and which of the measures (explained later) will show the influence of scene stylization on judgement performance best.

3. EXPERIMENTS

3.1 RESEARCH METHODS

It was the aim of the research work to find objective methods by means of which the influence of image stylization of landing approach scenes on height and distance perception could be evaluated.

As visual space perception depends strongly on the information content of the external scene, a relation between a visual stimulus and the resulting response (perception) of an observer has to be established

$$R = f\,(S) \qquad \begin{array}{l} S \text{ - stimulus} \\ R \text{ - response} \end{array}$$

and the change of this relation with changes in stimuli has to be evaluated.

A relationship where a subjective reaction depends on an objective stimulus can be described as a psychophysical function. These psychophysical relationships have often been written as power functions [4].

$$R = a\,(S - S_o)^n$$

With suitable dimensions and a big stimulus area in relation to the perception threshold constants may be neglected :

$$R \sim S^n$$

There exist a number of different psychophysical measurement techniques which differ in the level of measurement scale which can be used, i.e. ordinal, interval or ratio scale [5].

The ratio scale is characterized by equidistant steps and the use of the zero as the origin. It is the most preferred type of scale in psychophysics because one can use all of the parametric statistics with it. Measuring methods which yield ratio scales include the magnitude estimation methods which have their origin in the work of STEVENS, S.S.

Relative estimates made during the first part of the program were estimates of height and distance from presented slides as percentages of "standard" height or distance from a "standard" slide scene shown before.

The absolute estimates made during the second part of the program were estimates of height or distance in ft or meter (as each subject preferred).

The "standard" slide was shown before each slide presentation in the first of the two runs which each subject made. During the second run the standard slide was only shown in the beginning of the run itself. This was done to reduce the running time of the experiments since the subjects were familiar with the standard slide after the first run.

3.2 MATERIAL (APPROACH SCENES USED AS STIMULI)

For our research we used colored slides, representing external visual scenes of a landing approach. As subjects were to make judgements from these pictures, height and distance had to be well known to the experimenter. Because of the great difficulties and costs of taking exact position pictures from the real world, slides were photographed from the LUFTHANSA model simulation system in Frankfurt/M. The TV camera could manually be driven at different positions in accordance with specific values of height and distance. The corresponding pictures of the outside scenes were taken from a color TV monitor.

In this way, slides of 21 positions were produced, of which 12 scenes represented 12 different distances from the runway threshold on a 3° glide-slope and 9 scenes represented 9 heights with a fixed distance from the runway threshold (Fig. 1).

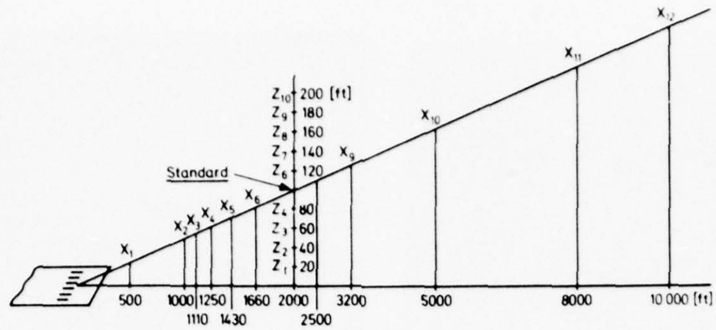

Fig. 1 Positions representing the different approach scenes

Because there was no CGI system at our disposal at the beginning of the research work, stylized scenes were produced on a drawing board. This was done by rear projection of the full scenes onto a translucent glass from which the visual elements were abstracted by stepwise selecting and coloring with translucent color sheets only the desired scene elements. Synthesized pictures were produced in five steps or degrees of stylization with the simplest ones consisting only of the trapezoidal looking runway and horizon (see table 1 and figure 2).

There were not raster points in the colored slides presented for judgement. The raster points on the runway in figure 2 were only used for printing to show the levels of contrast and detail respectively as they existed by color.

Table 1 Steps of Stylization

Step	reduction of scene elements
I	*texture, small objects, simple structure*
II	all 3-dimensional objects
III	all structure without runway and approach lights
IV	approach lights and runway threshold
V	runway centerline and border

3.3 EXPERIMENTAL APPARATUS

Selected color landing approach slides were projected on a rear translucent projection screen with a Leitz Pradovit S projector. Duration of slide presentation and changing of slides was regulated by an electromechanical circuit. When slides were changed the projector was darkened by a shutter.

Subjects sat in front of an acryl-plastic FRESNEL lens at a focal length distance which was equal to the distance between the lens and the projection screen (Fig. 3).

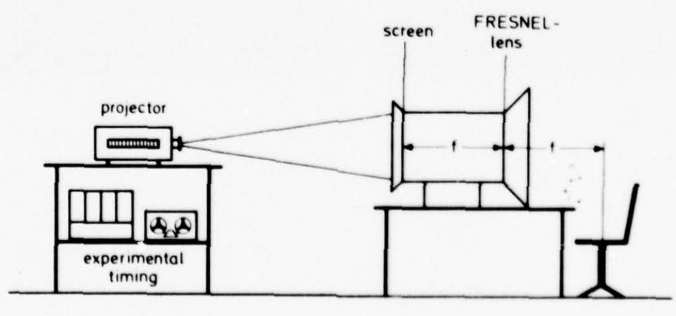

Fig. 3 Experimental apparatus

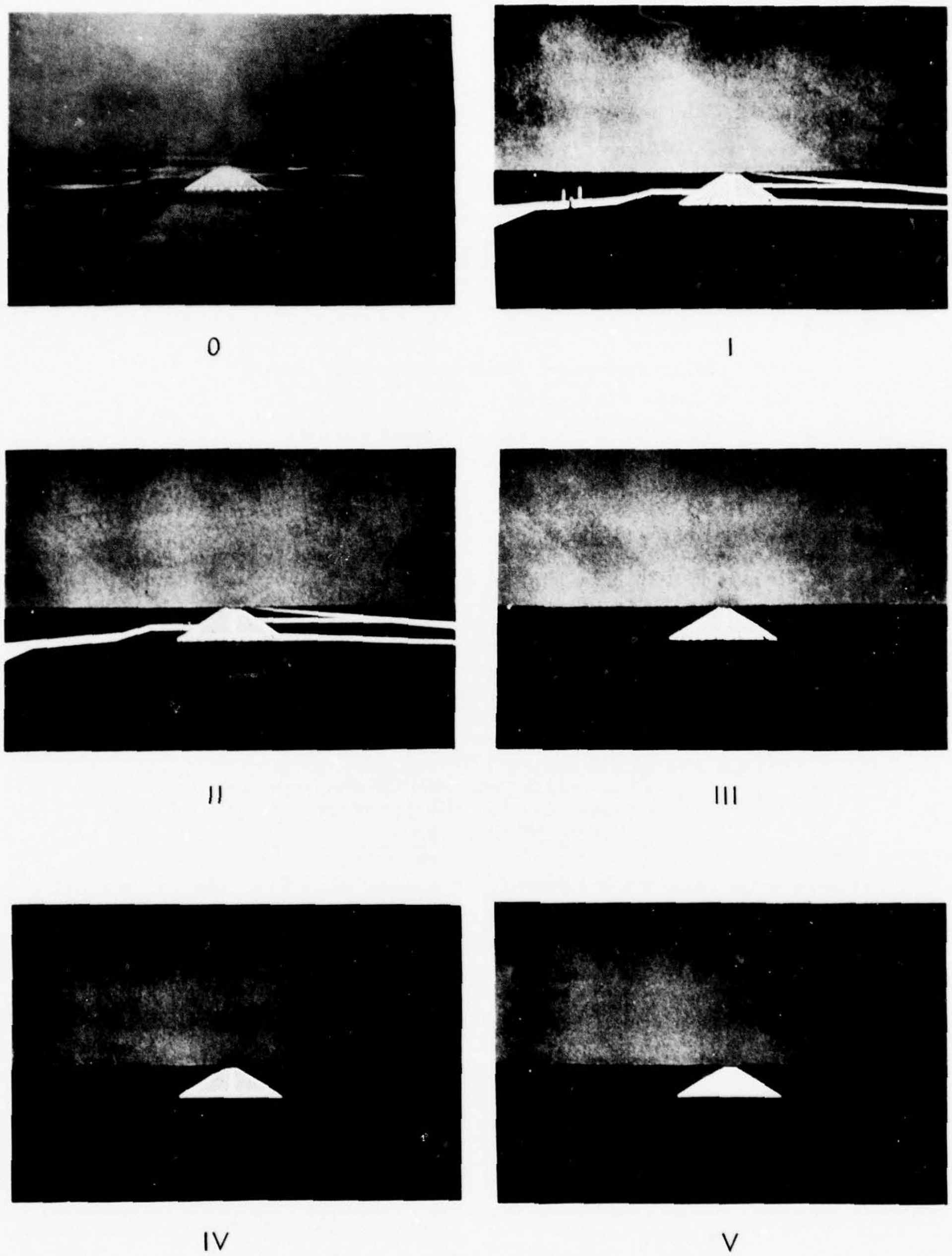

Figure 2 Slides representing the "full scene" (0) and the five steps of stylization (I - V)

By so doing, the visual angle subtended by objects in the slides would remain constant even if the subject moved his head in the direction of the optical axis of the system (i.e. closer or further away). 15-5

Use of the FRESNEL lens and a surrounding frame resulted in a depth impression similar to a real world scene looked at through a cockpit window.

Subjects sat on a chair which could be changed in height so that their eyes were coincident with the centre of the lens and the horizon of the projected scenes. The screen size of 40 x 60 cm provided a viewing situation to subjects with the same 48° horizontal and 36° vertical visual angle used in the Lufthansa model TV system.

3.4 SUBJECTS

For the relative estimate series of experiments there were 3 groups of subjects with 12 in each group. These were cargo (Transall) pilot, jet (Starfighter) pilot and non-pilot enlisted men groups.

For the absolute estimate series of experiments there were 3 seperate groups of subjects with 16 in each group. These were helicopter (Bell UH-1D) pilot, jet (Phantom) pilot and non-pilot enlisted men groups.

Pilot age ranged between 24 and 44 years with a median of 29 years.

Their flight hours were between 270 and 4000 with a median of 1300 hours.

The age of non-pilot enlisted men was between 19 and 25 years with a median of 20 years.

3.5 PROCEDURE

A general introduction of the purpose of our research was first given to all our subjects followed by more detailed information which was recorded to minimize variations in experimenter influence resulting from inconsistent presentations. Subjects were then familiarized with our automatic slide presentation procedure and given some practice in making "relative" or "absolute estimates". The approach scene slides were not used for practice, instead a substitute series of practice slides involving judgement of the length of various horizontal lines were used so that no training in height or distance judgement was given.

Right after familiarization with procedure and estimate practice, data collection began with the presentation of a group of slides. With the exception of half of the non-pilot subjects making relative estimates subject groups made height judgements first, and after a break of 20 minutes to 2 hours made distance judgements.

The slides representing various combinations of variables were arranged in a random sequence as determined by drawing numbers out of a hat. Experimental variables used were the degree of stylization, judgement height, and judgement distance. The "standard" slide was a "full scene" slide which was taken at a position corresponding to a height of 100 ft and a distance of 2000 ft from runway threshold. This slide was shown prior to every slide to be judged for the relative and the absolute estimation during the first run. The entire group of slides was presented twice to subjects. But because of the great number and variety as well as the random order of the slides, subjects did not realize that they judged the same slides twice.

Slide presentation duration was 5 seconds for the standard scene and 8 seconds for other slides.

4. DATA TREATMENT AND RESULTS

To measure the influence of image stylization on height and distance perception from landing approach scenes, the exponent (n) of fitted power-functions, the absolute value of the relative estimation error (ERA) and the estimation time (t) were calculated.

The values of the exponents of the stimulus-response relationship express a tendency of under or overestimation of great stimuli in the range used. When calculating the absolute value of relative estimation error there is the possibility of determining the amount of the error for each stimulus separately as well as the over-all error.

Calculation of the research data was done by means of ANOVA 45, a computer program for analysis of variance [8]. Basically a multi-factor plan was selected, and the variances of the dependent variables were proved against their interaction with subjects [9].

For comparison of the two different tasks used and the different measures selected, with regard to reliability, rank-correlations were calculated [11].

It was examined for which judgement task and which measure there was the best correlation between an increase in scene stylization and a change in the value of the specific measure.

For this purpose rank numbers were assigned to the mean estimation errors and the calculated power-function exponents of each subject for the six succeeding degrees of stylization with regard to their magnitude. By means of this procedure coefficients can be calculated representing a measure of correlation between two rank-orders.

4.1 POWER-FUNCTION

The first measure used was the exponent n of the psychophysical relationship $R \sim S^n$. By means of fitted power-functions of subjects' height or distance judgements a systematic change of the psychophysical relationship as a result of scene stylization was determined.

15-6

From the data for each subject which were in the form of $[(S_i, R_i); i = 1, 2, \ldots, m)]$ power-functions were calculated for height as well as for distance judgements for each degree of stylization.

By transformation the problem can be reduced to solving a lineare regression [10]

$$R \sim S^n$$

$$\ln R = n \cdot \ln S + k$$

and the exponents can be computed by means of a program.

Figure 4 gives the power-function exponent for height judgements. For each degree of stylization the exponent n has been calculated for all pilots as a group as well as for non-pilot subjects for both relative and absolute estimation tasks.

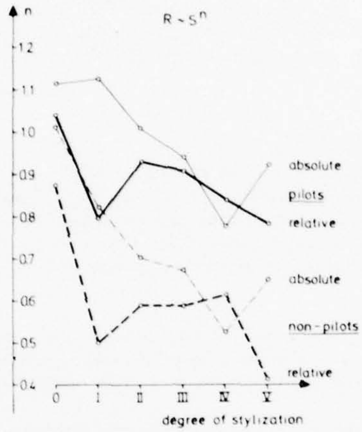

Fig. 4 Power-function exponent of height judgement

There is an over-all tendency for power-function exponent to decrease with scene simplification which will lead to a larger amount of underestimation of height from more stylized scenes.

But there are clear differences between the exponents of pilots and non-pilots respectively.

For both relative and absolute height judgements the values of n are smaller for non-pilots than for pilots.

The differences are smaller for unstylized scenes than for stylized or simplified scenes.

The influence of image stylization is greater for absolute estimates than for relative estimates. Analysis of variance shows that there are significant differences between relative and absolute results as well as between results of pilots and non-pilots.

In figure 5 the power-function exponent n of distance judgements by pilots and non-pilots with both estimation tasks are shown.

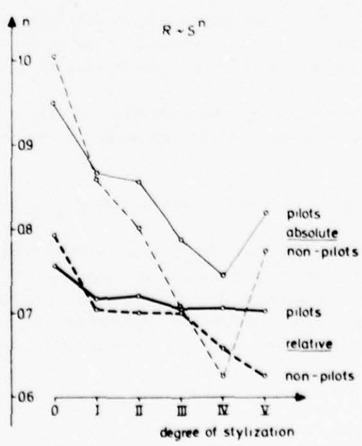

Fig. 5 Power-function exponent of distance judgement

As seen in figure 4 for distance judgement too there is the tendency of power-function exponent to decrease with scene simplification.

While the values of n in the two tasks are significantly different the differences between pilots and non-pilots within each task were not significant.

The stronger decrease of the exponent for absolute distance judgements than for relative estimates indicates a clearer influence of scene stylization with the absolute task. This is verified by an analysis of variance which shows more significant differences between degrees of stylization with the absolute distance judgements by pilots than for relative judgements. For non-pilots a clearer influence of scene stylization with absolute distance estimates was not shown by analysis of variance.

4.2 ESTIMATION ERROR

Another measure for calculating the influence of scene stylization on visual space perception was the error made by subjects in making height or distance judgements. To get this measure the difference between each subjects' response and the actual correct height or distance was calculated. The absolute amount of this difference was divided by the correct height or distance value in order to obtain relative estimation errors, i.e. errors normalized relative to the magnitude of the height or distance judged.

Figure 6 gives the relative estimation error for height judgements.

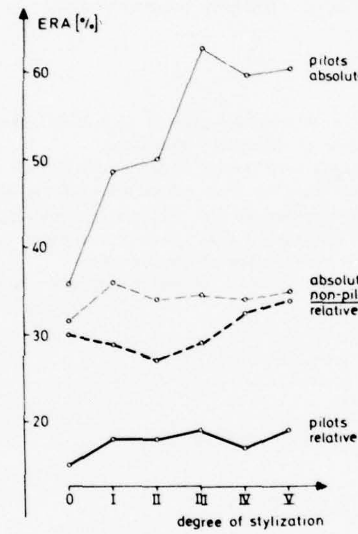

Fig. 6 Height judgement error

As can be seen in figure 6, for both estimation tasks and both subject groups there was an over-all tendency for estimation error to increase with increased degrees of scene stylization.

The differences in errors between pilots and non-pilots for both estimation tasks are quite large. There are smaller errors for pilots than for non-pilots when making relative estimates, but there are larger errors for pilots than for non-pilots with the absolute judgements.

There are no significant differences between tasks with non-pilots.

The error differences between tasks with pilots are very large and very significant.

There are no significant differences between estimation errors for different degrees of stylization with absolute height judgements by non-pilots and only few significant differences with relative estimates by non-pilots and pilots.

Error differences between successive degrees of stylization were not significant for all conditions except the absolute judgement by pilots, where there are most significant differences between degrees of stylization at all.

There were generally significant differences between unstylized scenes and stylized scenes for both subject groups and both tasks.

Figure 7 gives the distance judgement error for both estimation tasks and for pilots as well as for non-pilots.

For both subject groups and tasks there is an over-all tendency for error to increase with stylization step increases. This general tendency is significant in all four cases.

The large differences seen between pilots and non-pilots with height judgements do not occur with distance judgements. Mean error for pilots with absolute judgements was significantly larger than for all other cases.

There were no significant differences between any of the other cases.

Results from the analysis of variance program showed that with distance judgements there were significant differences between stylized and unstylized scenes but no significant differences between the degrees of stylization except for two steps with the absolute task for pilots.

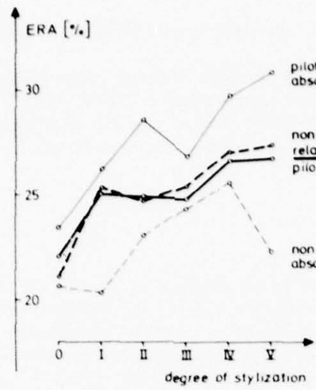

Fig. 7 Distance judgement error

4.3 REACTION TIME

The third measure used to evaluate the information content of outside scenes for visual space perception was the reaction time of subjects when making height or distance judgements.

Because subjects were asked by instructions to answer exactly and quickly as possible, calculating the reaction time seems convenient. "Reaction time" was the time from presentation of the scenes on the screen, marked by an acoustic signal, until the verbal estimation response by the subjects. It was supposed that stylization of the outside scene, because of the decrease in visual information, would reduce the power of judgement [7]. Therefore subjects may deliberate longer which would produce an increase of reaction time.

Figure 8 shows the reaction time and the standard deviations for estimating height and distance for the different degrees of stylization.

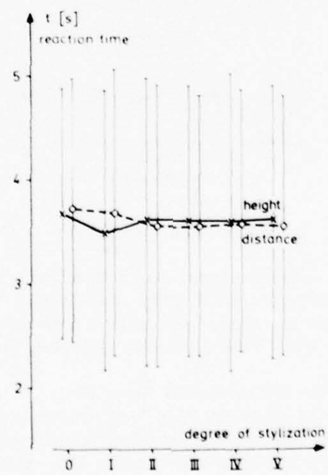

Fig. 8 Reaction time of height and distance judgement

The mean reaction time of 3.6 sec is the same for height as well as for distance judgement. No significant differences in reaction time between different degrees of stylization were found. There were also no significant differences between subject groups or between height or distance judgement reaction times.

The calculated standard deviations for both height and distance judgement reaction times were nearly equal for all degrees of stylization. The standard deviation for each stylization step was rather large, being around 1.3 seconds on the average which indicated large individual differences between subjects. On the other hand, variations in reaction time for each subject across the stylization steps was very small, indicating that subjects' reaction time was rather consistent regardless of the degree of stylization.

No correlations between estimation error and reaction time or between flying experience and estimation error or flying experience and reaction time were found. Results indicate that judgement reaction time do not represent the time required for height or distance judgement but rather the longer time for transforming this judgement into numerical data and verbal reactions.

Therefore reaction time may not be an appropriate measure for evaluating scene stylization influences on visual space perception.

4.4 SUMMARY AND DISCUSSION OF RESULTS

Six groups of military pilots and non-pilot enlisted men made height and distance judgements from landing approach scenes.

Two tasks were used to get subject responses from static image stimuli.

One half of the subject groups made height and distance estimation in relation to a standard approach scene whereas the other groups made absolute judgements in ft or meter.

To measure the influence of image simplification, estimation error (ERA), the exponent of psychophysical power-functions (*n*) and judgement reaction time (*t*) were calculated.

Results with relative as well as with absolute judgements of height and distance were quite different between the various measures.

Judgement reaction times did not show any significant differences between stylization steps. Therefore reaction time probably is not an appropriate measure to validate synthesized scenes for simulation.

For comparison of estimation error and power-function exponent with both judgement tasks with regard to reliability, rank-correlations were calculated. Table 2 gives the means of correlation coefficients for height and distance judgements with both research tasks (relative and absolute) and for both measures (error and exponent).

Table 2 Coefficients of rank-order correlations

method	measure	r_s (height)	r_s (distance)
relative	error ERA	0.18	0.28
	exponent *n*	0.21	0.26
absolute	error ERA	0.37	0.33
	exponent n	0.37	0.32

Calculation of rank-correlation coefficients shows higher values with the absolute than with the relative task for height and distance judgement as well as for the measures error and exponent respectively.

This indicates, that with the absolute estimation method the influence of scene stylization is demonstrated more distinct than with the relative method. Between coefficients for estimation error and power-function exponent there is no remarkable difference. Both measures seem to be equivalent for demonstrating the influence of scene stylization.

But power-function exponents are rather erratic, giving highly different values as a consequence of a large number of variables (e.g. stimulus range, spacing of standard, naming of standard etc. [12, 13, 14]). Furthermore power-function exponents are not a measure of specific stimulus-response events but are rather arbitrary descriptions of groups of events. Therefore power-function exponents seem to be less dependable and less appropriate as a measure than estimation errors. On the other hand estimation errors seem to have a direct relevance to the task of flying the landing approach because height and distance estimates by the pilot are directly involved in the task of flying the approach and making a landing. Consequently estimation errors seem to be the more appropriate and credible measure.

During calculation of estimation errors from subject responses of height judgement an effect was obtained, which might lead to another possible research method : For only three distinct images at step IV of stylization some pilot subjects made estimation errors up to 700 % which were unequal to errors for other scenes of the same degree of stylization.

An exact examination of these approach scenes led to the assumption that the reason for this misjudgement was a shortcoming in producing the stylized approach scenes. By the appearance of a contrast threshold at the color TV-monitor where the slides of the stylized approach scenes were taken from the left side of the white runway surrounding was broadened and became parallel instead of being perspectively reduced. This led to an overestimation of height by some military subjects because from very high positions these lines seem to be nearly parallel. For the subjects concerned the remaining information content in the scenes of stylization step IV was not sufficient to get the correct perception of height but was concealed by the geometric error. This indicates that there will be the possibility to produce geometric or perspective errors on purpose and to evaluate up to which extent of stylization subjects get correct visual space perception. By this method there will be the possibility to find cues supporting correct visual perception or others being of no influence.

15-10

Though there had been used static landing approach scenes as stimuli for our research, it should be possible to use methods and criteria for dynamic research too. One can expect that there will be more visual information by the additional cues of motion (e.g. parallax, streaming patterns, etc.) for perception of height and distance too. By that an improvement of visual perception will be obtainable.

It should be verified however whether there will be a fundamental change in the tendency of error or power-function exponent data by dynamic experiments compared to the results of our previous static experiments. Further research will have to be conducted in order to show in how far static data is transferable to dynamic situations.

Besides the intention to produce appropriate research methods there was the question whether non-pilots could be used as subjects for visual research with flight scenes. The reason is the well known need for pilot subjects and the better availability of non-pilots as well as a simplification of research program and performance if the procedure could be done in the laboratory.

Though there were no general differences in the tendency of the results between pilots and non-pilot subjects the analysis showed that there were significant differences between the two subject groups at least for height judgements. These differences depend on different experience and therefore on the use of different visual cues of external scenes in visual space perception.

This is a reason while non-pilots should not be used as subjects for visual research work with landing scenes.

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DESIGN CONSIDERATIONS FOR IMPLEMENTING
INTEGRATED MISSION-TAILORED FLIGHT CONTROL MODES

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ABSTRACT

Acceptance of full authority Fly-By-Wire (FBW) flight control techniques, as a viable technology for further expanding classical design boundaries, has paved the way for continued development and exploitation of other innovative flight control concepts. These include decoupled flight path control, mission-tailored control modes, and fault tolerant system mechanizations. The purpose of this paper is to highlight some of the more critical design considerations for successfully integrating these advanced concepts in a multi-role, high performance fighter aircraft design, to achieve improved overall mission effectiveness and cost of ownership without sacrificing system reliability and safety. The concept of integrated mission-tailored control modes is the next logical step in the progression of advanced flight control technology. Results from the recently completed Fighter Control Configured Vehicles (CCV) flight test program have provided valuable insight and substantiating technical data for future design and application of active control technology. This program was primarily concerned with the development and evaluation of decoupled six degrees-of-freedom flight path control techniques. Specific CCV features evaluated during the flight test program included (a) maneuver enhancement/gust alleviation, (b) direct lift and sideforce control, (c) independent fuselage pointing and translation, and (d) variable relaxed static stability. Implementation of these CCV capabilities presents unique pilot interface considerations, which must be addressed in terms of required displays, controllers, vehicle response dynamics and mission segment applications. The resulting matrix of possible mode configurations and mission segment/task applications is considerably expanded over conventional mechanizations. Application of digital technology is considered a prerequisite for integrating these advanced CCV features in a practical and affordable mission-tailored multimode flight control system design.

LIST OF SYMBOLS AND ACRONYMS

A_N	Direct Lift Control
α_1	Elevation Fuselage Pointing
α_2	Vertical Translation
A_y	Direct Sideforce Control
β_1	Azimuth Fuselage Pointing
β_2	Lateral Translation
C*	Longitudinal Flying Qualities Parameter
c.g.	Center of Gravity
g	Acceleration due to Gravity
K ft	1000 feet
M	Mach Number
mac	Mean Aerodynamic Chord
Xs	Composite Sensor Coverage
Xc	Computer and I/O Coverage
Xa	Actuator Coverage
Xz_2	Total Second Fault Coverage
λs	Composite Sensor Failure Rate
λc	Computer and I/O Failure Rate
λa	Actuator Failure Rate

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* Program Manager, Digital Flight Control System Advanced Development Program
** Program Manager, Control Configured Vehicles Advanced Development Program

λ	Channel Failure Rate
ACM	Air Combat Mode
A/D	Analog to Digital
ASG	Air to Surface Gunnery
ASB	Air to Surface Bombing
BIT	Built-In-Test
CCV	Control Configured Vehicles
D/A	Digital to Analog
DIGIPACT	Digitized PACT Control Laws
DOF	Degrees of Freedom
FBW	Fly-By-Wire
GPC	General Purpose Computer
HQDT	Handling Qualities During Tracking
HUD	Heads-Up-Display
IFM	In-Flight Monitoring
I/O	Input/Output
ME	Maneuver Enhancement
MFD	Multi-Function Display
PACT	Precision Aircraft Control Technology
SAS	Stability Augmentation System
SFCS	Survivable Flight Control System

1. INTRODUCTION

To take full advantage of decoupled flight path control techniques and to avoid saturating the pilot with a potentially excessive number of discreet control modes, each with its own peculiar dynamic characteristics, a higher level of total system integration is required. Organizing primary flight control modes and associated control laws based on specific mission/weapon delivery requirements offers the opportunity for enhancing overall mission effectiveness, while at the same time emphasizing the pilot's role as a mission manager rather than a subsystem operator. In consonance with this design philosophy, the following five basic mission tailored control modes are considered appropriate for fighter aircraft applications: (a) Normal Mode with ancillary functions for take-off/landing, refueling, formation, cruise and pilot relief, (b) Air Combat Mode including air-to-air gunnery, (c) Air-to-Ground Gunnery Mode, (d) Air-to-Ground Bombing Mode and (e) Emergency Mode, which reconfigures inner loop control laws to provide optimum flight characteristics based on available functioning system elements, flight condition and aircraft configuration. In each of these primary modes, the flight control system provides the necessary flight path decoupling, and desired vehicle response characteristics, specifically tailored and optimized for the appropriate mission segment. Achieving operational simplicity requires careful consideration of vehicle dynamics, logic functions, control law blending, gain schedules, feedback parameters, display formats and controller harmonization.

Digital technology is especially suited for implementing these sophisticated inner loop control functions and logic schemes required for successful multimode application. The primary benefits afforded by digital implementation are as follows: (a) high degree of flexibility and growth potential for accommodating system changes through software rather than hardware, (b) high computational capacity required for implementing multimode functions and redundancy management logic, (c) opportunity to reduce system redundancy from quadruplex to triplex and still retain acceptable mission reliability and fault coverage through a combination of comparison and in-line monitoring schemes, and (d) ability to maintain compatible interface for integration with other subsystems, such as fire control, propulsion, mission avionics and associated controls and displays, through a common digital data bus mechanization. Realizing these benefits in a practical and affordable design is dependent upon adequate consideration of several key technical factors including: overall system architecture, redundancy management schemes, self-test features, I/O interface, software management, electromagnetic compatibility, reliability, maintainability, safety/survivability and life cycle cost implications.

2. BACKGROUND

In March 1974, the Air Force Flight Dynamics Laboratory (AFFDL) contracted with McDonnell Aircraft Company to conduct an advanced digital flight control system definition study. This program included both analytical studies and manned simulations to explore potential payoffs and to establish technical feasibility of a digital Fly-By-Wire (FBW) flight control system, incorporating mission-tailored control modes, advanced multi-purpose displays and decoupled flight path control features. The baseline aircraft configuration used in these studies was the former FBW test aircraft YF-4E SN-12200, modified with differentially controlled horizontal canards. In a related effort under the Fighter Control Configured Vehicles (CCV) Advanced Development Program, AFFDL contracted with General Dynamics in December 1973 to develop and evaluate independent six degree-of-freedom decoupled flight path control techniques for improving fighter aircraft mission effectiveness. Results from these efforts have contributed significantly to the technology base required in the pursuit of advanced flight control concepts, such as decoupled flight path control, mission-tailored control modes and fault tolerant mechanizations. Because of their significance and impact on the mission-tailored control mode design philosophy, Fighter CCV program results are highlighted below.

3. FIGHTER CCV PROGRAM SUMMARY

Flight testing of the Fighter CCV marked the first exploitation of decoupled six degree-of-freedom (6DOF) flight control concepts. The Air Force Flight Dynamics Laboratory's CCV YF-16, in an 87 flight, 125 hour test program, accomplished validation of the new control concepts and demonstrated significant capabilities to improve overall mission effectiveness of fighter aircraft. Completed in June 1977, this program included pilot evaluation of the CCV control modes applied to air-to-air and air-to-ground mission oriented tasks. As background to this paper, the flight test results will be highlighted with specific emphasis on lessons learned regarding pilot/controller/aircraft interfaces. Figure 1 shows the CCV YF-16 test aircraft in flight.

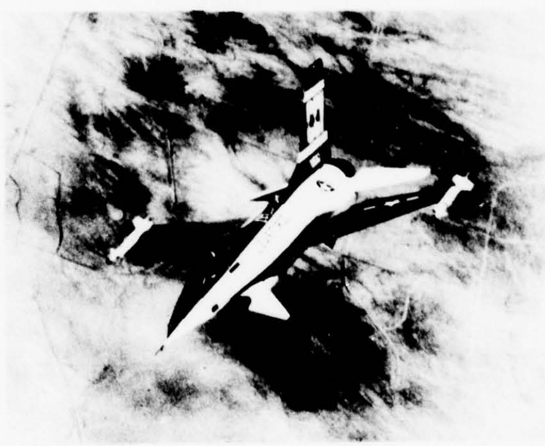


Figure 1 CCV YF-16 Testbed Vehicle

The primary thrust of the CCV YF-16 program was to generate and control uncoupled 6DOF aircraft motion and validate that innovative use of this capability could achieve new measures of fighter effectiveness. The ability to provide independent control of the six rigid body degrees of freedom allows flight modes that significantly differ from the conventional modes of current aircraft; i.e., direct lift and sideforce, decoupled angle-of-attack and sideslip control for fuselage aiming independent of flight path, and vertical and lateral translation. Response tailoring is also provided through blending of direct force and conventional longitudinal control. Quickened pitch response, gust alleviation and weapon line stabilization are available features. A description of the CCV YF-16 features follows.

Direct Force Controls

Direct force control is provided through multiple control surface inputs about a single axis so that trimmed aerodynamic force levels can be varied independent of angle-of-attack or sideslip. Command of flap deflection with coordinated horizontal tail deflection produces Direct Lift that controls vertical flight path at constant angle of attack (Figure 2). Similarly, command of vertical canard deflection with coordinated rudder inputs produces Direct Sideforce that controls directional flight path at zero sideslip angle (Figure 2).

Alternate use of these paired control surfaces can be made to vary angle of attack or sideslip independent of trimmed aerodynamic force levels. Maintaining constant flight path, pitch and yaw attitude can be varied to provide fuselage pointing (Figure 3). By changing the commanded surface deflections, vertical and lateral velocity control at constant pitch and yaw attitudes can be obtained. These are called Vertical and Lateral Translation modes (Figure 4).

On the CCV YF-16, each of these modes was implemented through open loop command by a two-axis, "coolie hat" force controller that replaced the trim switch on the sidestick controller. Input commands were proportional over a ±3 pound force range. Lateral mode inputs could also be input through the rudder pedals. Figures 5 and 6 illustrate the implementations.

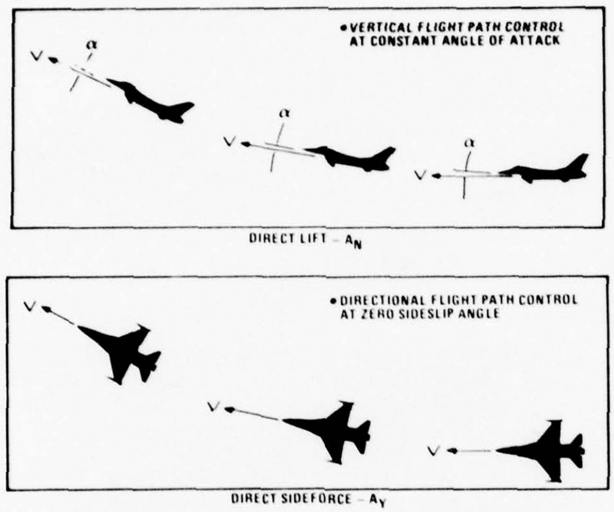


Figure 2 Direct Force Control Modes

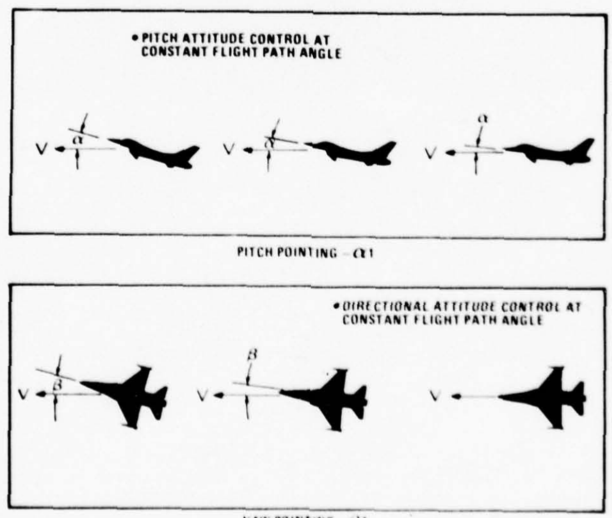


Figure 3 Airplane Pointing Control Modes

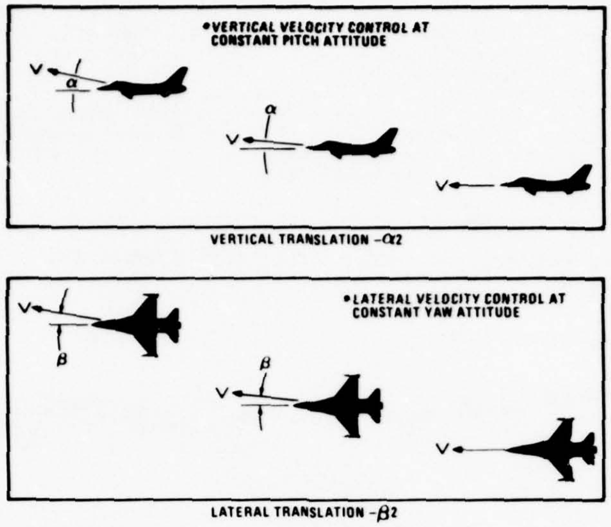


Figure 4 Airplane Translation Control Modes

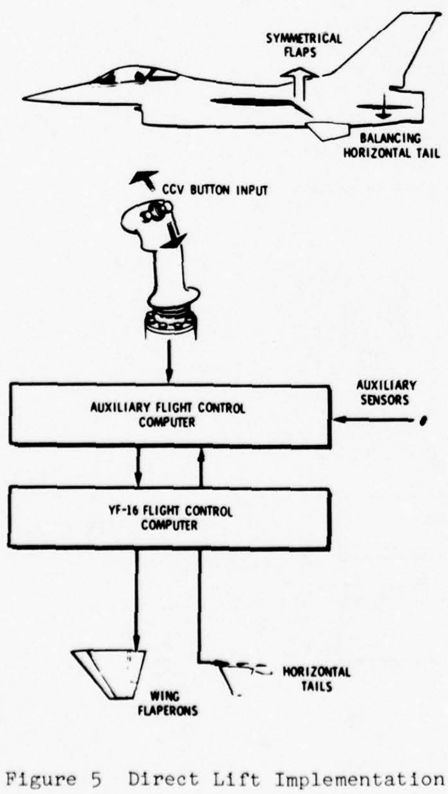


Figure 5 Direct Lift Implementation

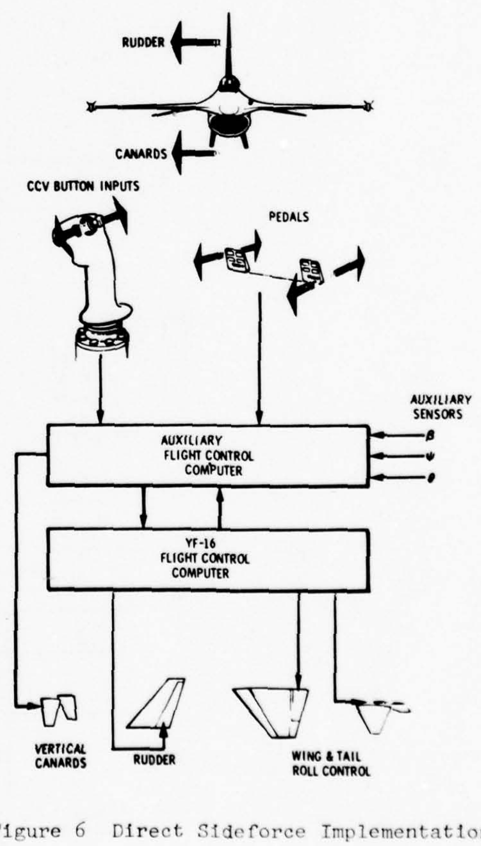


Figure 6 Direct Sideforce Implementation

16-6

Blended Force Controls

The one closed loop control mode implemented on the CCV YF-16 was called Maneuver Enhancement (M.E.). Direct lift was blended into the basic longitudinal control loop to improve transient response characteristics. The sensed difference between pilot commanded g and aircraft g drove the flaps to null the g error. The result was quickened g response, a better balance between g and pitch rate response, and a measure of gust alleviation. The aircraft's response to turbulence appears as uncommanded g and Maneuver Enhancement opposes the gust response. The manual direct force modes were useable, individually or in combination, with Maneuver Enhancement engaged. Figure 7 illustrates the blended control mode and Figure 8 shows its mechanization.

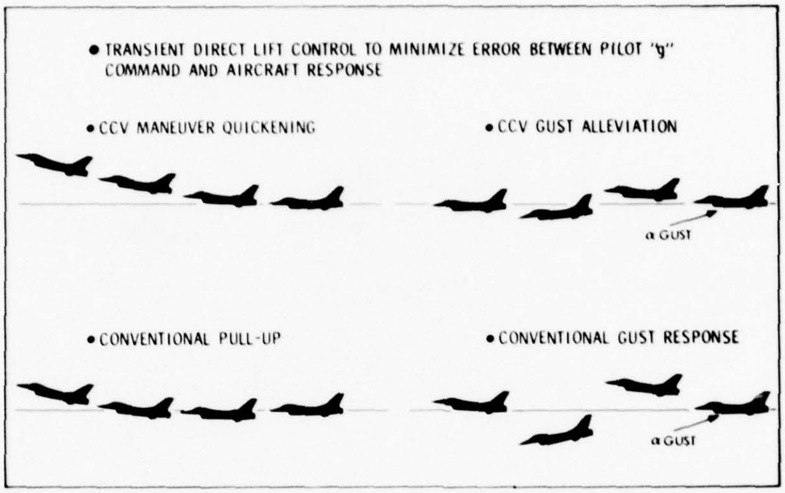


Figure 7 Blended CCV Control Mode

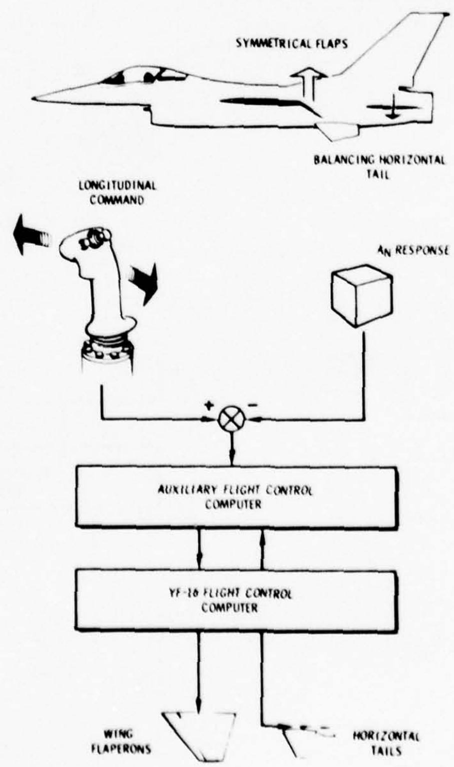


Figure 8 Maneuver Enhancement Implementation

CCV Flight Testing

Flight testing of the CCV features included a detailed engineering evaluation of mode gain characteristics, mode maneuver evaluations, air-to-air Handling Qualities During Tracking (HQDT) exercises, and an operational type evaluation that concentrated on typical air-to-ground tasks. Two CCV project pilots and four pilots from the F-16 Joint Test Force participated in the evaluation. Comments in this paper are limited primarily to demonstrated and projected applications and limitations found in evaluation of the CCV modes. A discussion of each type of CCV control mode follows. Endorsement of a mode implies that the pilot achieved better performance to the basic YF-16 as implemented or recognized a potential improvement if the mode were implemented more effectively.

Maneuver Enhancement - This closed loop, blended control mode was endorsed by a majority of the pilots because of improved control precision during maneuvering flight. Significant improvement in air-to-air tracking was obtained due to better longitudinal control through the pilot's ability to precisely adjust g. Figure 9 illustrates that with M.E. active, the g response more directly follows stick force commands. The commands are reduced in magnitude indicating tighter pitch control. Improvements are evident in smoother angle-of-attack response and reduced pitch rate response.

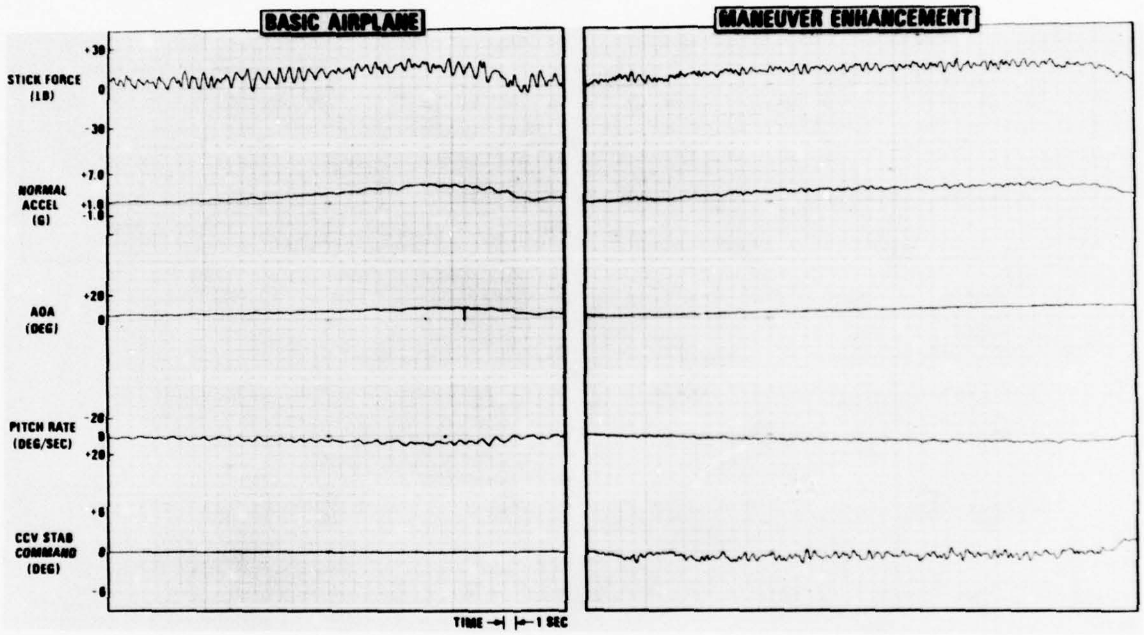


Figure 9 Windup Turn Tracking Input Commands and Response Comparisons
0.8M at 22,000 Feet

Figure 10 is a plot of pipper position during the Handling Qualities During Tracking (HQDT) task. All pilots found M.E. to improve their air-to-air HQDT performance. The gust alleviation feature was also found to reduce pipper upset. Figure 11 shows the reduction in gust response during low-level operations. For air-to-ground tasks M.E. was also identified as a benefit due to quickening of the pull-up recovery.

The gain/response of M.E. did not suit all pilots. Over-sensitivity was noted and the uncommanded gust alleviation was at times disconcerting, especially at low altitude. The need for task oriented control law tailoring of M.E. was indicated. For example, minimum gust response of a weapon line stabilization (g and attitude) mode for air-to-air and air-to-ground tasks may require different tailoring than for a low-level ride improvement feature.

Direct Lift and Sideforce Modes - These manual control modes were found to have potential for air-to-air tracking. Since flight path changes could be made directly without pitching or rolling-to-turn, quick and precise responses were obtained. Deadbeat response is obtained without objectionable pipper transients. A command immediately relocates pipper position; remove the command and the pipper remains where located. The button mechanization allowed the pilot to "beep" direct force inputs for precise changes in pipper position. Single axis flight path corrections were easily accomplished. Figure 10 illustrates the tracking improvement one pilot achieved by using Direct Lift and Sideforce.

Air-to-ground tasks showed significant payoff for the sideforce mode. Since flight path could be driven laterally, directly to the target (wings level steering), quicker line-up was possible. Without having to bank, pendulum effect with the fixed depressed reticle sight was eliminated. Direct Lift was not particularly useful for air-to-ground tasks since the pipper is normally allowed to "walk" to the target.

16-8

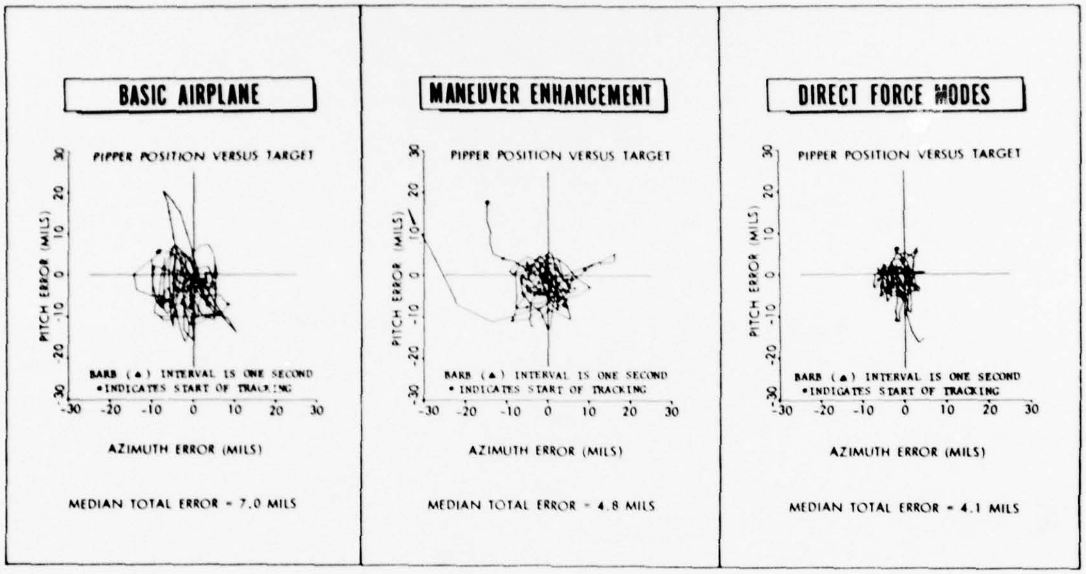


Figure 10 Windup Turn Tracking Error Comparison - 0.9M at 22,000 Feet

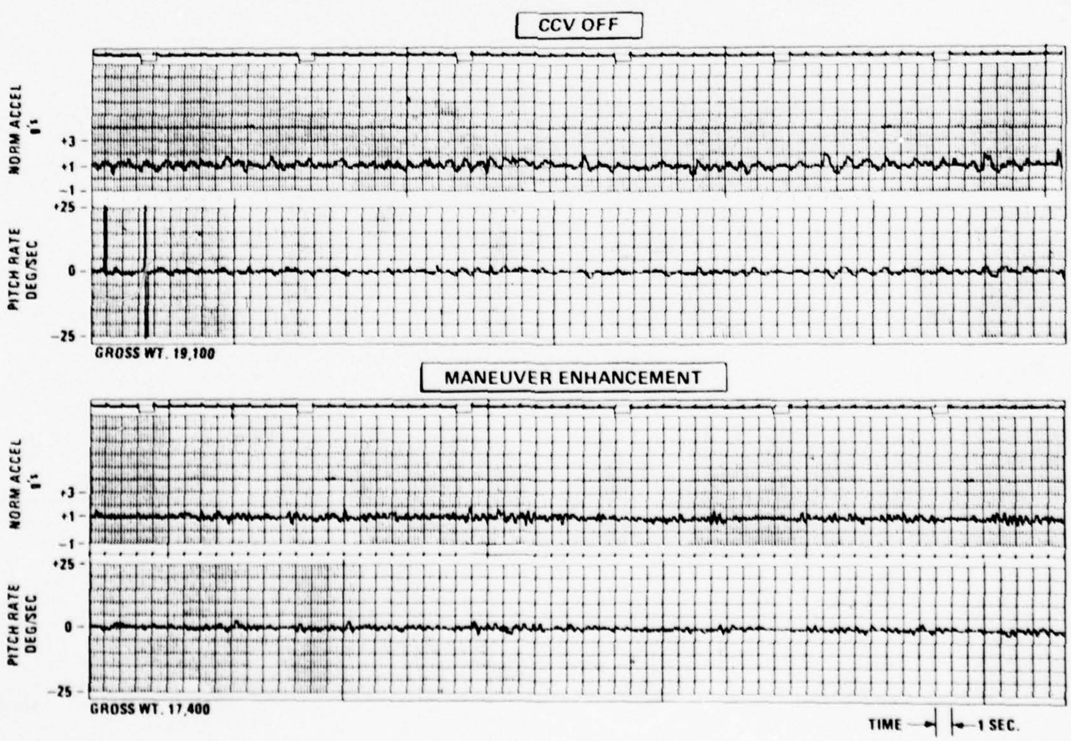


Figure 11 1g Gust Run - 0.85 Mach at 2,000 Feet Above Ground Level

Except for "beep" inputs, the pilots preferred using the rudder pedals for precise lateral inputs. Again, a task oriented design is needed. Gains set for air-to-air tasks were too sensitive and lacked the required control precision for air-to-ground tasks. Dual gradients to desensitize small inputs are also desirable. Air data scheduling is needed, particularly for high angle dive bomb deliveries. The pilots projected payoffs for Direct Lift and Sideforce in air combat maneuvering and for defensive tactics. Force levels of 2-3 g were desired; however, the CCV YF-16 capability was limited to a nominal one g level.

Fuselage Aiming Modes - Rapid changes in pipper position were possible with the Fuselage Aiming modes. However, manual implementation of these modes severely limited their practicality. Real utility may be limited to automatic applications such as Integrated Flight/Fire Control where a director fire control system would provide the pointing commands. Deficiencies in the test implementation of this mode were found in control laws, controllers and displays.

The proportional button controller on the sidestick controller was judged unsatisfactory. It was impractical and unnatural to fly flight path with sidestick controller and use the thumb to point off from the flight path. Cross coupling of axis inputs also gave poor response. In the tracking task, pipper positioning could be attained but since flight path was not changed by these mode inputs, saturation of pointing ability was rapidly reached and pipper position lost. A HUD presentation showing velocity vector and pointing authority may be required. Also, closed loop mechanization where a "trim follower" corrects flight path to the weapon line could be useful. Additionally, integral "beep" type pointing control inputs are desired.

Another problem was authority mismatch. Lateral pointing capability was about ±60 mils and longitudinal ±30 mils. Equalizing authorities and desensitizing inputs could improve the manual use of these modes.

One useful application of pitch pointing was in air-to-ground straffing. By pointing nose-down, minimum altitude could be raised, achieving the straffing solution at 400-500 feet as compared to 250 feet with the baseline system.

Translation Modes - The Vertical and Lateral Translation modes were found suitable for small position changes during formation flight, and for crosswind drift correction and matching ground target motion. Simulated landing approaches showed a crosswind trim capability of 15 knots. In a gunnery pass, the translation capability was sufficient to track targets, moving over 30 knots normal to the flight path.

Deficiencies were found in the translation mode control laws and controller design. The translation velocity buildup was too slow and overall response time characteristics were poor. Translation is achieved by an initial acceleration that washes out as steady state velocity is reached and is stopped in a reverse manner; some pilots complained of an uncomfortable "feel" to the lateral mode. Use of the modes was restricted to small bank angles (less than 10°) and was locked to a heading hold. These restrictions limited useful application of these modes. Control law changes and further evaluation is required. Controller deficiencies were also found. Integral command inputs or a closed loop velocity command system should be evaluated.

Fighter CCV Conclusions

The Fighter CCV program has demonstrated that uncoupled 6DOF control can be effectively used to provide innovative capabilities and improved mission effectiveness of fighter aircraft. However, this program was certainly but a first step in fully validating the capability. The design was intended for a capability demonstration and aimed at maximum capability within the aircraft's constraints. Specific task orientation was outside the scope of this initial work. It was found that the use of the different modes, controller preference, control gradients, mode authorities and dynamic characteristics were highly dependent upon the specific task. Task-tailored multimode control law implementation and selection capability is clearly required in future applications. While pilots quickly adapted to the new control modes, continued work towards simplified switchology and controllers is necessary. Another area of urgent need is the development of appropriate handling qualities criteria.

4. MULTIMODE CONTROL LAW DEVELOPMENT

The basis for consideration of mission-tailored or task optimized control mode concepts lies in the ability of the FBW flight control system to tailor basic aircraft response characteristics and flying qualities to meet a wide range of mission dependent requirements. Figure 12 illustrates a classical variation in longitudinal response characteristics achievable through appropriate tailoring of the flight control laws using the time domain C* flying qualities criteria.

The non-crosshatched area represents the C* boundary of acceptable normalized longitudinal response characteristics (blended pitch rate and normal acceleration). The dashed line response (basic F-4E with Stability Augmentation System (SAS)) falls outside the boundary, thereby indicating objectionable flying qualities at this flight condition. The solid line response (SFCS YF-4E) on the other hand lies well within the C* boundary and is achievable through appropriate blending of feedback parameters, gain scheduling and command signal filtering. It should be noted that many of the classical flying qualities parameters are not completely applicable to multimode FBW systems, since the forced response

16-10

nature of closed loop systems tend to mask free airframe dynamics. For this reason, great care should be exercised in defining system performance requirements and criteria, which are compatible with the application of other innovative flight control concepts, such as decoupled 6DOF flight path control.

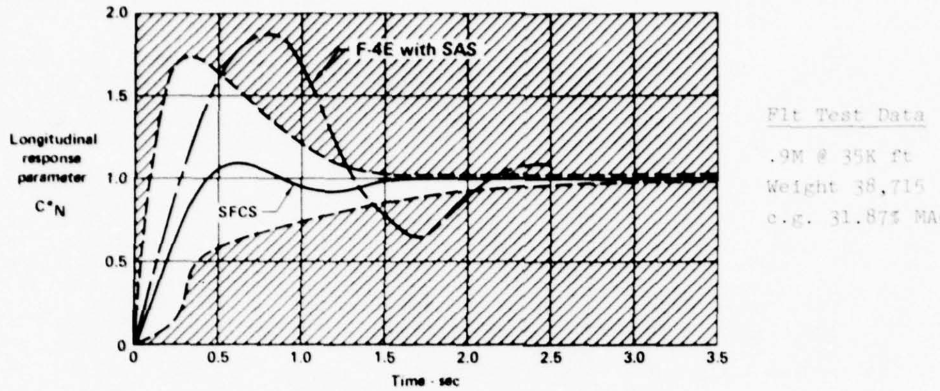


Figure 12 Variation of Longitudinal Response Characteristics

Control law synthesis is an iterative process with considerable reliance on past experience, trial and error analysis and piloted simulations. Major emphasis is typically centered around the definition and assessment of feedback parameters/blending, gain scheduling, air data inputs, command signal shaping, compensation/structural filter dynamics and specialized mode functions. Incorporation of CCV features in the context of a mission-tailored flight control system, adds a new dimension in the control law development process, by emphasizing the need for more extensive consideration of the pilot/vehicle interface, in terms of controller harmonization, display formats and system logic functions. Successful integration of these advanced multimode concepts is dependent on translating specific mission requirements into meaningful flight control system design parameters, along with adequate consideration of the pilot/vehicle interface. Representative improvements in tracking accuracies attributable to customized mission-tailored control modes are illustrated by the piloted simulation results depicted in Figures 13, 14 and 15. Each of the specialized combat modes, i.e. Air Combat Mode (ACM), Air-to-Surface Gunnery (ASG) and Air-to-Surface Bombing (ASB) employed a unique set of task matched control laws and CCV features based on the specific mission/weapon delivery requirements of each mission segment.

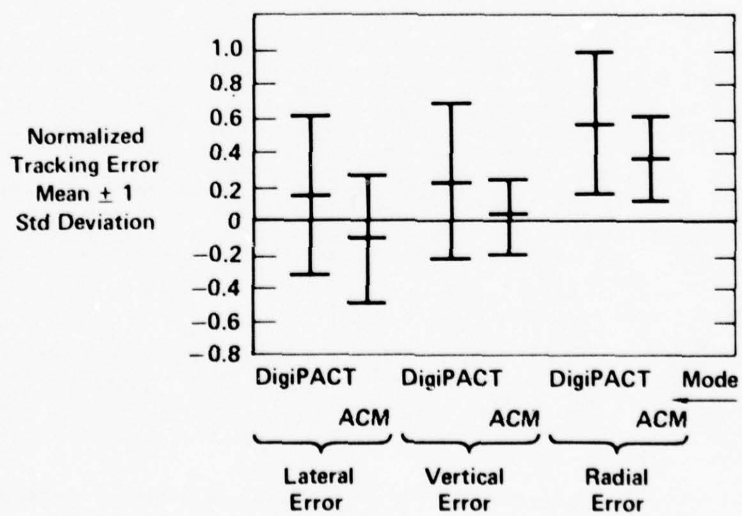


Figure 13 Representative Air-to-Air Gunnery Effectiveness

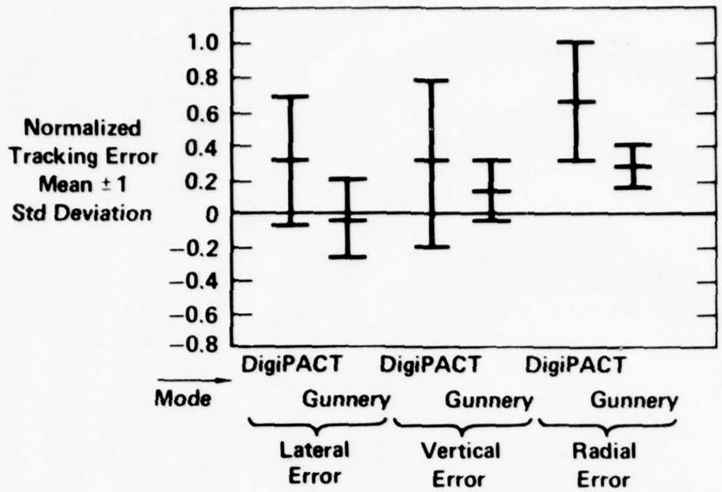


Figure 14 Representative Air-to-Surface Gunnery Effectiveness

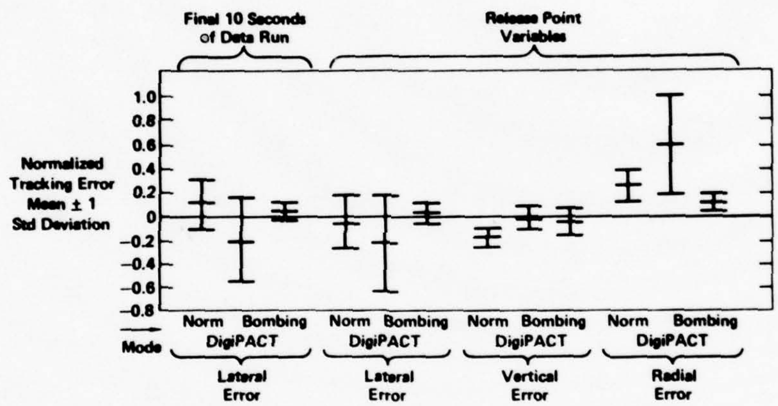


Figure 15 Representative Air-to-Surface Bombing Effectiveness

Examination of these data reveals significant reductions in tracking error for the specialized mission-tailored combat modes when compared to the baseline DigiPact mode, which employed only conventionally optimized control laws. The baseline aircraft configuration used in this particular simulation study was a FBW YF-4E modified with differentially controlled horizontal canards.

An example where the multimode design approach offers unique advantages, in terms of simplicity and effectiveness is a modern high performance fighter, which is designed and optimized for an air superiority mission. Such a system generally dictates a low wing loading design to achieve maximum maneuverability. In adapting this type of vehicle for an air-to-ground role, one must consider the potentially adverse effect of low wing loading on gust sensitivity during low altitude, high speed operations. Typically, air-to-ground tracking is somewhat compromised because the low wing loading vehicle is more sensitive to gust disturbances. The closed loop maneuver enhancement mode (balanced maneuvering flap and horizontal tail) currently implemented in the CCV YF-16 offers the potential for not only improving air-to-air tracking performance through tighter g and attitude control, (Figures 9 and 10), but also enhances air-to-ground tracking performance by reducing gust sensitivity (Figure 11) associated with a low wing loading design. In a multimode application, this feature would be implemented within the flight control computer, containing separate task optimized control laws for achieving desired weapon line stabilization characteristics for both air-to-air and air-to-ground operations and would not require extensive structural or aerodynamic changes/compromises.

5. CREW STATION IMPLICATIONS

16-12

Coordinating the mission-tailored control laws with advanced multi-purpose control/display concepts offers additional potential for enhancing operational versatility, mission management, and reduced pilot workload. The advanced fighter cockpit layout shown in Figure 16 allows considerable flexibility in tailoring the controller/display functions to the appropriate mission segment task.

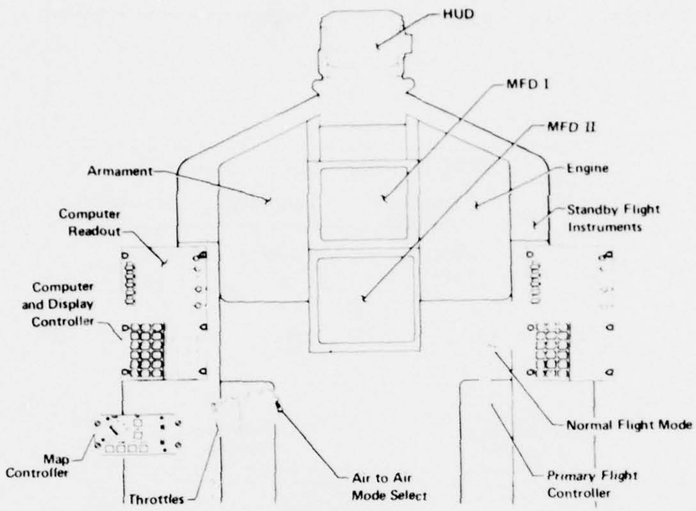


Figure 16 Cockpit Arrangement of Mode Related Displays and Controllers

Enhanced functional capabilities provided by the applications of multi-purpose controls/displays concepts include:

- Integrated stores management, communication and navigation
- Vertical Situation (Electronic ADI)
- Horizontal Situation (Moving map area navigation)
- Variable format/symbology
- Simplified mission mode selection ("hands on")
- Automated checklist
- Selective Routine/Emergency Information Annunciation
- Display redundancy/commonality
- Flight Plan/Mission Management

In a total mission context the power of the digital computer can be brought to bear on the complex task of integrating mission relevant controls, displays and vehicle dynamic characteristics, in such a way that the pilot assumes the role of a mission manager rather than a subsystem operator. *Crew station integration vis-a-vis multimode control* laws and coordinated multi-purpose controls/displays can be a significant influencing factor in achieving total uncompromised mission performance. Additionally, this approach facilitates the implementation of other advanced technologies, such as articulating (reclined) high g seat, integrated fire control/flight control and standarized avionic modules. Application of CCV features, such as independent fuselage pointing and direct force control represents a significant challenge to the pilot/vehicle interface and as such will require innovative crew station and control system design approaches to fully exploit this added capability.

6. SYSTEM IMPLEMENTATION CONSIDERATIONS

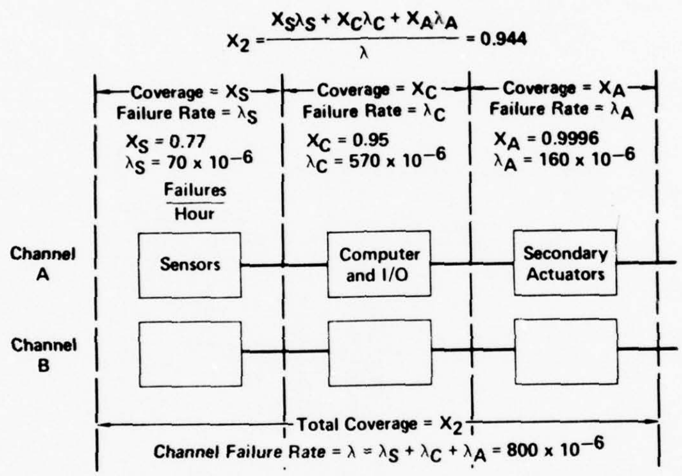
Digital implementation of sophisticated multimode control laws is an attractive approach based on the inherent advantages delineated in Section 1. As with any full authority FBW design approach, there are a number of critically important design considerations which must be addressed in order to satisfy not only system performance requirements, but also to establish a high degree of system integrity.

16-13

Reliability

To a large extent, the level of redundancy required to satisfy a specified mission reliability requirement (typically on the order of 10^{-7} catastrophic failures per flight hour) is dependent on the type of failure monitoring scheme (in-line and/or cross-channel comparison), component failure probabilities, desired fail-operational capability and overall confidence in achieving the desired reliability. Existing full-authority three-axis analog FBW mechanizations employ quadruplex redundancy to achieve the desired mission reliability; however, with digital implementation schemes it is feasible to consider triply redundant systems which can be mechanized to confidently satisfy mission reliability requirements. Reduction in the level of redundancy not only reduces overall system complexity, but also has an appreciable impact on weight and volume savings, along with reduced acquisition and life cycle costs. From a design standpoint and based on state-of-the-art in electronics, redundancy management and implementation schemes can have the largest single impact on overall integrity, reliability and cost of ownership of a FBW system.

In a triplex digital mechanization adequate second fault coverage is the primary concern. Probability of first fault coverage is essentially 1.0 using cross channel comparison monitoring. Coverage in this case is defined as the probability of detecting, isolating and recovering from an internal system failure. Typical second fault coverage and failure rates for critical system elements are shown in Figure 17.



Two-Channel DFCS with Lumped Elements Subject to In-Line Monitoring

Figure 17 Second Fault Coverage (X_2) in a Triplex System

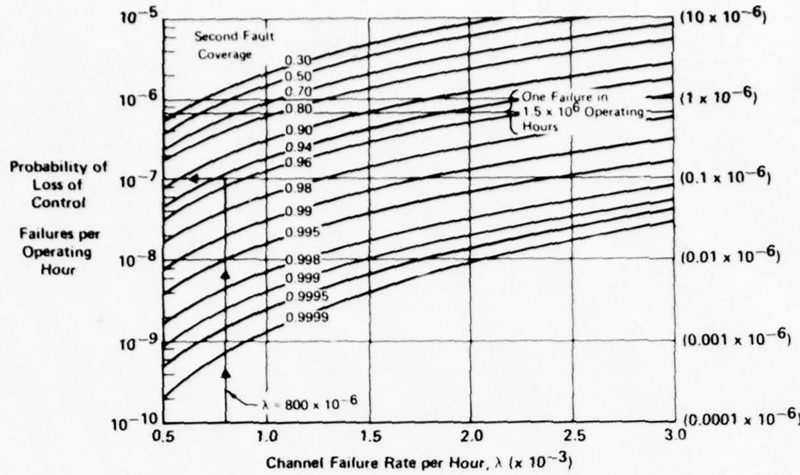
Figure 18 illustrates the relationship between second fault coverage, single channel failure rate and mission reliability, i.e., probability of loss of control. From these figures it is evident that one can achieve an overall mission reliability of 10^{-7} failures per operating hour for a triplex digital system, which has a second fault coverage of at least .944 and single channel failure rate not exceeding 800×10^{-6} failures per hour.

A group of generic quadruplex and triplex systems are compared in Figure 19. Quadruplex system Q-1 has a third fault coverage of 0.95. Achieving this level of coverage requires the same in-line monitoring techniques used in the triplex system. System Q-2, on the other hand, requires only a simple "heads-or-tails" test for third fault coverage. Triplex system T-1 has a second fault coverage of 0.95 which is conservative and readily achievable with known in-line monitoring techniques. It is theoretically possible to achieve a second-fault coverage of 0.99 and the probability of loss of control shown by T-2. For any triplex system to have as low a probability of loss of control as quadruplex system Q-2 requires a second fault coverage of 0.9996 (system T-3) which is extremely unlikely. Based on studies and analyses conducted to date, it is considered feasible to achieve satisfactory mission reliability (10^{-7}) with a triplex digital configuration employing a combination of cross channel comparison and in-line monitoring schemes.

Software

The need for effective software management cannot be overemphasized. Experience indicates that software can be a major source of problems in the development and operation of digital systems. Adequate software planning during the initial system definition phase is essential. As a minimum, the plan should address design specifications, system interfaces, configuration control, documentation procedures, modular coding and testing, support software, module integration, hardware and software integration and verification.

16-14



Note: First Fault Coverage = 1.0
Coverage = the Probability of Detecting, Isolating and Recovering from Failures

Figure 18 Triplex System Failure Probabilities

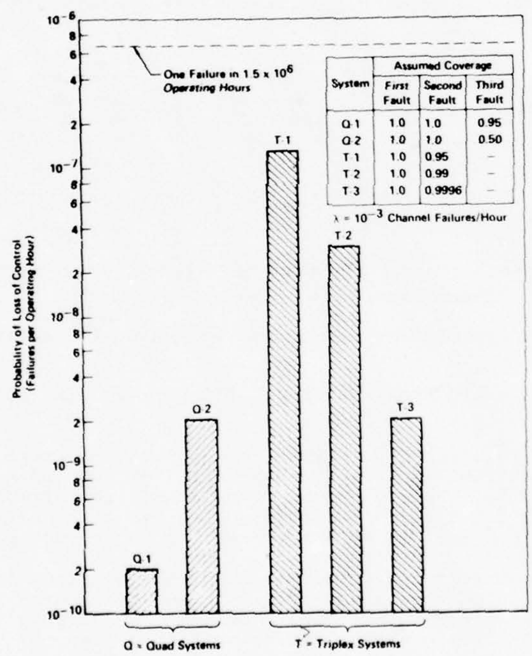


Figure 19 Comparison of Quadruplex and Triplex System Reliabilities

Software development for digital flight control system applications must consider at least four major elements, i.e., Executive, Redundancy Management, Built-in-Test (BIT) and Control Laws. Comparison of estimated total memory requirements for triplex and quadruplex configurations of a 3-axis mission-tailored multimode flight control system is shown in Table 1.

The executive software provides the main skeleton into which other software elements are integrated. It provides the synchronization, computational rate structure and redundancy configuration of the software, in the event the fault monitoring and/or BIT routines indicate a failure. The size will vary depending on the particular processor, control logic and input/output mechanization used. The executive function also provides the

Category	Memory	
	Triplex	Quadruplex
Control Laws	6,000	6,000
BIT	2,000	3,270
Redundancy Management	3,350	3,400
Executive	3,990	3,840
Total	15,340	16,510

Table 1
Total Memory Required for Flight Control Computers

proper management and interface with other subsystems such as displays, air data, fire control system and navigation.

Redundancy management provides the necessary fault detection, isolation and recovery logic to satisfy overall system reliability and fail-operational requirements. Signal selection algorithms and fault recovery strategies are key elements in the design of a fault tolerant flight control system. Achieving .95 second fault coverage with a triplex configuration requires application of both cross-channel monitoring and in-line monitoring techniques. Other more advanced concepts which could be considered are analytical redundancy and reconfiguration.

Functional implementation must address the following fundamental considerations:

Cross Channel Monitoring

- Number of Voting Planes
  - Sensor and Controller Inputs to Computer
  - Intermediate Computed Parameter
  - Computer Outputs to Actuators
- Computer Monitoring of Actuators
  - ΔP Comparison
- Redundant Data Comparison and Voting
  - Signal Selection
  - Averaging
- Computer Synchronization
  - Bit by Bit
  - Frame
  - Comparison of Intermediate Computational Results
  - Asynchronous
- Computer Interchange of Redundant Data

In-Line Monitoring

- Computer Self-Test
- Computer Test of Sensors and Controllers
  - Data Reasonableness
  - Torquers
  - Dither
- Computer Test of Actuators
  - Actuator Models
- Computer Test of I/O and Multiplex
  - Parity
  - Wraparound
  - Rebound

- Computer and Hardware Test of Power Supplies

16-16

- Data Reasonableness

Software design, programming standards, verification and control procedures must be sufficient to assure that no catastrophic single failure points exist. A case in point is the addition of multiple voting planes. Inclusion of voting planes is one of the ways by which a redundant flight control system is made fault tolerant. Theoretically, the more voting planes, the more fault tolerant the system. When the system is analog, the addition of voting planes, over and above what are needed to achieve the required system reliability, are costly due to the signal buffering required to prevent fault propagation between channels and the additional dedicated analog voters. With a digital computer, the buffering and cross-channel data transfer are readily facilitated, and the same software voting algorithm can be used to vote on many different signals. Consequently, there is a temptation to consider using many voting planes to increase fault tolerance and improve system reliability. On the other hand, interconnections between redundant channels required to implement a large number of voting planes tends to increase the potential for common mode failures. For this reason, it is desirable that voting planes be limited to planes B, C and E as shown in Figure 20. In this manner data cross-strapping is confined to a single digital data exchange bus, which can be properly buffered and monitored. In general, interconnections between redundant channels should be minimized to limit the potential for propagating system failures. Monitor thresholds at these voting planes must be sufficiently tight to limit excessive switching transients but yet loose enough to avoid frequent nuisance disconnects.

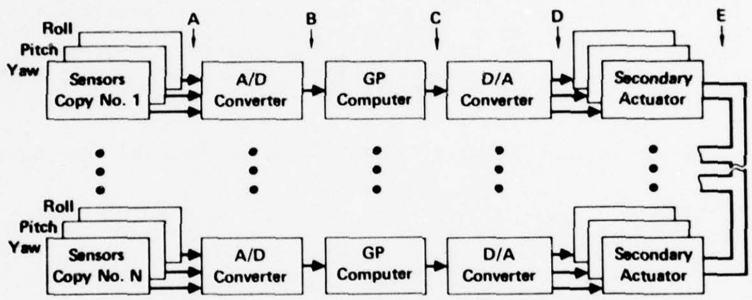


Figure 20 Potential Voting Plane Locations

One of the major advantages of a digital architecture is the ability to accommodate extensive self-test features required for implementing reliable in-line monitoring schemes. After one failure in a triplex system and two failures in a quadruplex system, in-line monitoring must be used to resolve any channel differences, if the flight control system is to continue to operate. When in-line monitoring is used, the computer must first test itself; then it is in a condition to check other system elements. Self-testing of digital computers involves a mix of hardware and software. Certain basic portions of the computer must be operable before any self-testing can be conducted, e.g., power supplies and clocks. Failure of these basic portions must be detected by hardware. With these basic portions of the computer operating, self-testing of the computer can begin. The design of the self-test program is based on the inverted pyramid test philosophy. That is, the program first tests the instructions that require a minimum of logic for their execution, and the memory locations that contain the self-test program. These verified instructions and memory locations are then used to test instructions and memory on the next higher level. This process is continued until all of the instructions, memory and I/O have been verified.

Built-in-Test (BIT) as distinguished from In-Flight Monitoring (IFM) and its associated self-test features is a term used to describe a sequence of internal software controlled tests conducted on the ground, to validate system integrity and/or troubleshoot the total flight control system. It is generally desirable to incorporate two levels of BIT, one to be used for pre-flight checkout and the other for organizational maintenance purposes, without the need for specialized external test equipment and/or highly skilled technical personnel. Implementing BIT and IFM functions will require basic decisions regarding hardware vs. software mechanization. Implementation of monitoring tasks in hardware rather than software generally results in a higher channel failure rate due to the added piece part count. For this reason, software monitors should be used where performance can be met.

The variation in control laws needed to establish a full-up mission-tailored multimode control system, incorporating CCV features requires sophisticated inner loop functions to provide the necessary mode switching logic, feedback blending, signal shaping, gain schedules, filters, compensation networks, etc. The simplified lateral-directional axis block diagram of Figure 21 illustrates an example of the level of switching required to alter the basic Normal mode to a specialized Air Combat mode.

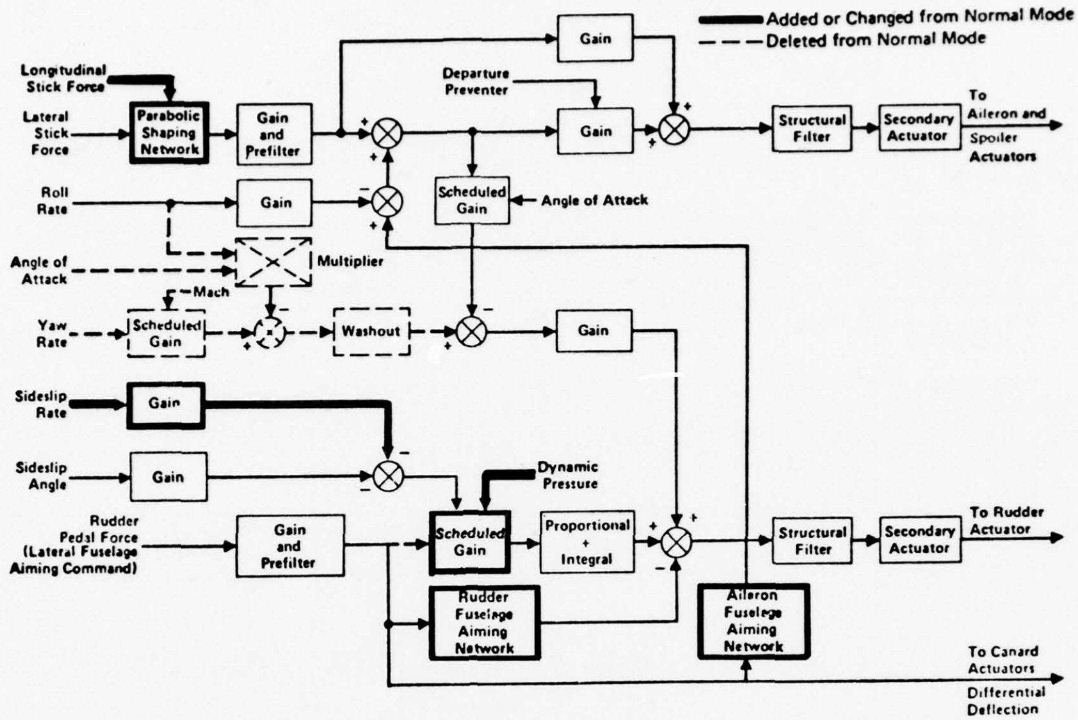


Figure 21 Air-to-Air Combat Mode Lateral-Directional Axes Block Diagram

Implementation of multimode functions places tremendous demands on the computational elements of the flight control system and is a major factor in driving the technology towards a digital mechanization. In mechanizing these multimode control laws, one must consider two fundamentally different design approaches. The first approach and perhaps the most widely accepted is digitization of continuous analog transfer functions. This approach uses a transformation method which transforms the continuous filters (analog) that comprise the control laws into difference equations which are solved by the digital computer. The resulting difference equation mimics the continuous filter in the frequency domain. The quality of the digital transformation is judged on the ability to match the analog filter without consideration of the original performance specification. The second approach, direct digital design, presumably overcomes this restriction by allowing direct digital synthesis in either the W or Z planes. The issue of direct digital design versus the transformation approach tends to be somewhat of a moot point, since any Z transfer function has an analog counterpart for a given conversion technique. Given sufficient understanding of the distortion produced by a particular conversion, proper adjustments in the analog form can be made for compensation. In most cases, either technique can provide a satisfactory digital implementation with little effect on required computer resources.

Computational Element

Digital computers must be selected with sufficient instruction repertoire, throughput, memory and input/output provisions to accommodate sophisticated inner-loop multimode control laws and also provide adequate dynamic performance in the control system frequency range. Of particular concern are the effects of aliasing and output granularity in relationship to system iteration rates. Another somewhat controversial area in the design of digital flight control systems is the need for synchronous versus asynchronous operation of redundant digital computers. Frame synchronization is desirable for a number of reasons. First, near time identical samples of redundant sensor signals can be taken, processed, equalized, voted and a common signal selected for use in subsequent computations in all computers, thereby minimizing tracking errors, preventing channel divergence, and facilitating the detection of failed computers and sensors. Additionally, near simultaneous mode selection can occur in all computers. Asynchronous operation on the other hand also offers some unique advantages by simplifying the computer/sensor interface and eliminating the need for synchronizing algorithms and highly reliable external clocks. This approach also minimizes the possibility of introducing single failure point failures. However, disadvantages of asynchronous operation are significant. Specifically, large time skews in sensor data sampling can occur, which could result in an unacceptable balance between large disengage transients and frequent nuisance disconnects. Another drawback is the possibility of different control modes existing in different computers for at least one computation frame. Overcoming these drawbacks requires higher sample rates, which could be impractical to achieve in multimode control system design.

7. SUMMARY

The combined result of a digitally implemented mission-tailored flight control system design philosophy provides the opportunity for achieving improved mission performance, operational versatility and reduced cost of ownership without sacrificing system safety and reliability.

Results from the recently completed YF-16 CCV flight test program have provided valuable insight and substantiating technical data for future design and application of decoupled flight path control technology. Application of CCV technology in the context of a mission-tailored flight control system adds a new dimension to the control law development process by emphasizing the need for more extensive consideration of the pilot/vehicle interface, in terms of controller harmonization, display formats and system logic functions.

Digital technology is especially suited for implementing these sophisticated inner loop control functions and interfacing with other avionic subsystems, such as fire control and mission related controls and displays. Realizing these benefits in a practical and affordable design requires a high level of total system integration including consideration of several key technical factors addressed in this paper.

The concept of integrated mission-tailored control modes is the next logical step in the progression of advanced flight control technology. Advanced development and flight test evaluation of these concepts and others are currently planned under the Air Force Flight Dynamics Laboratory sponsored Advanced Fighter Technology Integration (AFTI) Program.

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3. Wood, R.A., et al, YF-16 CCV Operational Potential, Flying Qualities and Performance Evaluation, AFFTC-TR-77-23, Air Force Flight Test Center, Edwards AFB, CA, January 1978

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The fourth alternative in Table 1 is a fully custom minicomputer using mostly MSI logic. This machine uses very high speed bipolar devices and allows complete freedom in its architectural design. The result is

TARGET MARKER PLACEMENT FOR DIVE-TOSS DELIVERIES WITH WINGS NON-LEVEL

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SUMMARY

In a dive-toss, air-to-ground weapon delivery, the pilot steers a target marker symbol or sight reticle (pipper) so as to overlay the target with that symbol. He then depresses a target designation (pickle) switch which commands the computer to record all available target sensor data. From these data the weapon delivery computer first calculates the location of the target and then generates steering signals to guide the pilot in steering the calculated weapon impact point onto the target whereupon the computer automatically issues the weapon release signal.

This paper contains an analysis of the motion of the calculated impact point during a banked pullup or climbing turn. The objective of this analysis is to determine the path followed by the calculated impact point during such a maneuver. Placement of the sight reticle along this path allows the pilot to pull straight back on the stick after designating the target without first unrolling to a wings level attitude.

NOMENCLATURE

A_N	Normal acceleration of the aircraft
CCIP	Continuously computed impact point, same as manual release
CCRP	Continuously computed release point, same as automatic or dive-toss release
F	The dimensionless quantity, $\sin\theta \cos\gamma (\tan\gamma + \mathrm{ctn}\, \gamma_I)$
g	Gravitational acceleration
HUD	Head up display
R_B	Ballistic range of the weapon
t_f	Time-of-fall (release to impact)
t_{PR}	Parameter nominally equal to 2.5 sec for dive-toss and level laydown deliveries and equal to zero for CCIP deliveries
T_R	Bomb trail
V_x	Down track component of weapon velocity at release or computed release point
V_y	Cross track component of weapon velocity at release or computed release point
V_z	Downward component of weapon velocity at release or computed release point
V_{zI}	Downward component of weapon velocity at (computed) impact
X-Y-Z	Down track, cross track, and downward, respectively; a right-handed, earth-fixed, Cartesian coordinate set; the X-axis is parallel to the aircraft's ground velocity at the target designation time; the Y-axis is positive to the right facing down track
X	Down track position coordinate of weapon at (computed) release
X_I	Down track position coordinate of the (computed) impact point
Y	Cross track position coordinate of weapon at (computed) release
Y_I	Cross track position coordinate of the (computed) impact point
Z	Altitude of weapon above target at (computed) release
γ	Weapon flight path angle, the angle between the weapon's velocity vector and the horizontal plane (positive in a dive), at the (computed) release point.
γ_I	Flight path angle of weapon at (computed) impact

In the design of this system, the cost and computer availability to a fairly large degree determined the functional partitioning of the system and to a certain extent the complexity of the algorithms used. It is interesting to note that the use of the computer to perform the basic tracker algorithms led to the

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δ Drift angle, the angle between the X-axis and the horizontal component of the weapon's airspeed vector at (computed) release.

θ Angle between the horizontal plane and the line-of-sight to the computed impact point

θ_T Angle between the line-of-sight to the computed impact point and the line-of-sight through the target marker symbol at or before target designation

ϕ Aircraft roll angle (positive with right wing down)

ϕ_I Projection onto the sight plane of the angle between the X-axis and the direction of motion of the computed impact point

$(\dot{\ })$ Time derivative of the quantity ( ).

I. INTRODUCTION

Most computerized air-to-ground weapon delivery systems have at least two delivery modes. One is manual release or continuously computed impact point (CCIP) mode. The other is an automatic release mode, also referred to as dive-toss or continuously computed release point (CCRP).

In the manual or CCIP mode, the computer displays the resulting impact point if weapon release were to occur at the present time. The pilot steers the aircraft so as to overlay the target with this impact point symbol. He then depresses the weapon release button which manually triggers the weapon release.

In the dive-toss or automatic mode, the computer displays a target marker symbol (pipper) which is elevated above the computed impact point. This elevation or lead angle is necessary so that the target marker symbol will pass the target in advance of the release time. The pilot steers the aircraft so as to overlay the target with the pipper and then depresses a "target designation" or "pickle" switch. This action signals the computer to record all available target sensor information such as the line-of-sight azimuth and depression angles, slant range, and altitude. From these data the computer calculates the target location and generates steering signals which direct the pilot to steer the computed impact point toward the target. As the computed impact point crosses the computed target position, the computer automatically issues a weapon release signal.

The term "dive-toss" as applied to this delivery mode comes about because the pilot, after pickling (designating) the target in a dive, usually pulls back hard on the control stick to initiate a high g pullup. By this pullup maneuver the pilot gets rid of the bomb as soon as possible. He can then initiate evasive action to avoid antiaircraft fire.

The subject of this paper is the proper placement of the target marker symbol (pipper) prior to target designation (pickle). If the pipper position is short of the computed impact point (negative lead angle), the aircraft would already be past the release point when the pilot designated the target. On the other hand, if the pipper position is above the horizon, the pilot cannot position it over the target, which is presumably on the ground. By this line of reasoning, the weapon delivery system must position the target marker somewhere between the impact point and the horizon, but where is the best place to put it within these limits?

Some currently operational weapon delivery systems set the elevation coordinate of the target marker to zero depression. Others match the depression angle of the target marker to that of the aircraft's velocity vector. As for the azimuth coordinate, most of these current systems employ a drift stabilized sight. That is, the target marker symbol lies in the azimuthal plane of the aircraft's ground velocity vector.

The effect of drift stabilizing the sight is to place the target marker symbol in the path of the computed impact point (neglecting cross trail) provided the aircraft's ground velocity does not change direction between pickle and release. In other words, the steering signals generated by the computer, after the pilot designates the target, will call for wings level flight. The pilot can still pull up, but he must do so with wings level.

Because the pilot is usually working very hard to steer the target marker symbol over the target, the aircraft is often in a bank at the time of pickle. In such a case the pilot must first unroll to a wings level attitude before initiating his pullup maneuver. The natural tendency of pilots, however, is to pull straight back on the stick after pickling the target, ignoring the wings level steering commands.

The cause of this mismatch between system operation and the pilot's instinctive reaction is the system design decision to drift stabilize the sight. The system designer can achieve a better match between pilot and system by having the system anticipate the wings nonlevel pullup and position the target marker accordingly. Such anticipation is achievable, for example, by positioning the target marker to the left of the aircraft velocity vector, when the aircraft is in a bank to the left, and to the right of the aircraft velocity vector, when the aircraft is in a bank to the right.

throughput rate than an AC. As seen in Table 4, the simplest TMC's operate at about the throughput rate of 21-5
the most advanced AC while the more powerful custom designed missile computers require a throughput rate four
... typical Avionics processor. (Special applications such as Real Synthetic Array mapping can

The left-right displacement of the target marker as a function of roll angle provides an auxiliary benefit which may prove to be more significant than the elimination of the wings level pullup requirement. This feature gives the pilot direct control of the left-right motion of the target symbol. He simply has to roll the aircraft left or right. The sight pipper moves accordingly, and the response is immediate. This greatly simplifies the pilot's task of steering the target marker onto the target, and hence will result in better aiming (closer coincidence of target and target marker symbol at pickle) by the pilot and more accurate weapon deliveries. 18-3

To better appreciate the advantage of this direct control of the sight reticle, consider the aiming task of the pilot when using the drift stabilized sight. In this case, the pilot must move the velocity vector of the aircraft in order to move the target marker. The velocity vector, however, is one integral removed, in a dynamic sense, from the attitude of the aircraft, which is what the pilot controls. Consequently, the velocity vector and the target marker lag the pilot's control actions, and it takes a considerable degree of pilot skill and training to steer the target marker into coincidence with the target.

The foregoing discussion establishes the desirability of having the target marker symbol, in a dive-toss weapon delivery mode, move laterally across the sight in response to the roll attitude of the delivery aircraft. The principle objective of this paper is to determine the proper amount of sight pipper displacement as a function of the aircraft's roll angle. The next section attacks this problem by analyzing the motion of the impact point during a constant bank angle, turning pullup. Placement of the pipper along this line will achieve the desired result. Namely, it will permit the pilot to pickle the target and pull straight back on the stick without changing his bank angle.

II. ANALYSIS OF IMPACT MOTION

The first objective of the analysis is to determine the relationship between aircraft acceleration and the apparent motion of the impact point on the sight or HUD against the target background. Equation (1) is the general impact equation which expresses the position (X_I, Y_I) of the bomb's impact point in terms of the position (X, Y) and ground velocity (V_x, V_y) of the bomb at release.

$$\begin{aligned} X_I &= X + V_x t_f - T_R \cos\delta \\ Y_I &= Y + V_y t_f - T_R \sin\delta \end{aligned} \tag{1}$$

where t_f is the time-of-fall, T_R is the bomb trail, and δ is the drift angle.

The analytical procedure will be to differentiate Eq. (1) with respect to time to obtain a relationship between the impact point motion, $\dot{X}_I$ and $\dot{Y}_I$, and the aircraft acceleration. It will greatly simplify the analysis to neglect the T_R terms, in effect restricting the analysis to the zero drag case. This is not so restrictive as it might seem at first, because the dive-toss mode is usually employed with low drag bombs anyway, and such bombs approximate a zero drag bomb reasonably well. Furthermore, the purpose of this analysis is to determine the most convenient placement of the target designating pipper. If, due to approximations in the analysis, the pipper location is not quite in line with the impact point motion as the pilot pulls up, the computer will command a small compensatory steering correction. If the pilot follows this steering command he will avoid the bombing error which otherwise would have occurred.

Neglecting T_R in Eq. (1) based on the foregoing rationale, we proceed to differentiate Eq. (1) with respect to time as follows.

$$\left.\begin{aligned} \dot{X}_I &= V_x + \dot{V}_x t_f + V_x \dot{t}_f \\ \dot{Y}_I &= V_y + \dot{V}_y t_f + V_y \dot{t}_f \end{aligned}\right\} \tag{2}$$

By definition, the cross track velocity V_y is zero at the target designation point. At this time, therefore, Eq. (2) reduces to

$$\left.\begin{aligned} \dot{X}_I &= V_x + \dot{V}_x t_f + V_x \dot{t}_f \qquad \text{(down track)} \\ \dot{Y}_I &= \dot{V}_y t_f \qquad \text{(cross track)} \end{aligned}\right\} \tag{3}$$

Equation (3) shows us that, at target designation time, the cross track velocity of the computed impact point is directly proportional to cross track acceleration with t_f being the constant of proportionality. Further reduction of the along track impact motion equation requires an expression for $\dot{t}_f$. For the zero drag bomb approximation, the time-of-fall is a function only of altitude above target and vertical velocity. Hence,

$$\dot{t}_f = \frac{\partial t_f}{\partial Z}\dot{Z} + \frac{\partial t_f}{\partial V_z}\dot{V}_z = -\frac{\partial t_f}{\partial Z}V_z + \frac{\partial t_f}{\partial V_z}\dot{V}_z \tag{4}$$

The partial derivatives $\partial t_f/\partial Z$ and $\partial t_f/\partial V_Z$ can be expressed analytically (for a zero drag bomb) as follows.

are Gallium Arsenide Field Effect Transistor (GASFET) logic, Transverse Electron Devices (TEDs), Josephson junction logic and Dielectrically Isolated Emitter Coupled Logic (DECL). These technologies will offer subnanosecond propagation delays and with the potential of speeds almost an order to magnitude better than

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$$\frac{\partial t_f}{\partial Z} = \frac{1}{\sqrt{V_z^2 + 2gZ}} = \frac{1}{V_{zI}} \tag{5}$$

$$\frac{\partial t_f}{\partial V_z} = \frac{1}{V_{zI}}\left(\frac{V_z - V_{zI}}{g}\right) = \frac{-t_f}{V_{zI}} \tag{6}$$

where $V_{zI} = V_z + gt_f = \sqrt{V_z^2 + 2gZ}$. Substituting Eqs. (5) and (6) into Eq. (4) gives us

$$\dot{t}_f = -\frac{\left(V_z + t_f \dot{V}_z\right)}{V_{zI}} \tag{7}$$

In a constant speed, coordinated maneuver the along track, cross track, and vertical accelerations are all related to the aircraft's normal acceleration, A_N, as follows.

$$\begin{aligned} \dot{V}_x &= A_N \cos\phi \sin\gamma \\ \dot{V}_y &= A_N \sin\phi \\ \dot{V}_z &= -A_N \cos\phi \cos\gamma + g \end{aligned} \tag{8}$$

Substitution of Eqs. (7) and (8) into Eq. (3) and use of the identity, $V_z + gt_f = V_{zI}$, yields

$$\left.\begin{aligned} \dot{X}_I &= V_x + A_N t_f \cos\phi \sin\gamma - \frac{V_x V_z}{V_{zI}} + \frac{V_x t_f}{V_{zI}}(A_N \cos\phi \cos\gamma - g) \\ \dot{X}_I &= A_N t_f \cos\phi (\sin\gamma + \cos\gamma \operatorname{ctn}\gamma_I) \quad \text{(down track)} \\ \dot{Y}_I &= A_N t_f \sin\phi \quad \text{(cross track)} \end{aligned}\right\} \tag{9}$$

where $\operatorname{ctn}\gamma_I = V_x / V_{zI}$ and represents the slope of the (computed) impact trajectory with respect to the vertical.

The apparent down track motion $\dot{X}_I$ in the plane of the sight will appear to be foreshortened by the sine of the depression angle θ to the (computed) impact point. Hence the tangent of the direction of apparent motion of the impact point in the sight plane is

$$\tan\phi_I = \frac{\dot{Y}_I}{\dot{X}_I \sin\theta} = \frac{\tan\phi}{F} \tag{10}$$

where $F = \sin\theta \cos\gamma (\tan\gamma + \operatorname{ctn}\gamma_I)$.

If the quantity F in the denominator of Eq. (10) were equal to unity, then $\tan\phi_I$ would equal $\tan\phi$, and the apparent motion of the impact point would be "straight up the sight," parallel to the ordinate or normal axis of the sight. Actually, the quantity F is always less than unity (see Appendix A for proof) for any dive angle less than 90 degrees. Because F is less than unity, Eq. (10) tells us that the angle ϕ_I will be greater than the angle ϕ. This means that the impact point will track off at an angle slightly to the right of the sight's ordinate axis in a right-hand bank or slightly to the left of that axis in a left-hand bank. Equation (11) expresses the amount of this angular deviation mathematically.

$$\tan(\phi_I - \phi) = \frac{\tan\phi_I - \tan\phi}{1 + \tan\phi_I \tan\phi} \tag{11}$$

Using Eq. (10) to eliminate $\tan\phi_I$ from the right-hand side of Eq. (11), we have

$$\tan(\phi_I - \phi) = \frac{\left(\frac{1}{F} - 1\right)\tan\phi}{1 + \frac{1}{F}\tan^2\phi} \tag{12}$$

or

$$\boxed{\tan(\phi_I - \phi) = \frac{(1 - F)\tan\phi}{F + \tan^2\phi}} \tag{13}$$

THRU PUT [1]	CPU POWER	CPU [2,3] COST/UNIT	PC BOARD AREA

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Equation (13) is the desired relationship which describes the track or direction of motion of the computed impact point across the sight during a banked but coordinated, turning pullup. Figure 1 is a graphical plot of Eq. (13) for several values of the quantity F. Figure 2 shows the quantity F plotted for various dive angles, altitudes and air speeds. This figure indicates that $0.70 < F < 0.95$ for dive angles of 20 to 40 degrees, air speeds between 400 and 600 kts, and altitudes up to 2000 m. Because F is generally greater than 0.70, we see that $\phi_I - \phi$ is generally less than 10 deg (175 milliradians).

Figure 3 illustrates how the sight might look while pickling in a 20 degree bank to the right in a 20 degree dive at 400 kt from an altitude of 610 m above ground level. The sight screen is shown rolled 20 degrees to the right. The computed impact point is shown displaced slightly to the right of the ground track plane to represent the cross trail effect of a left to right crosswind.

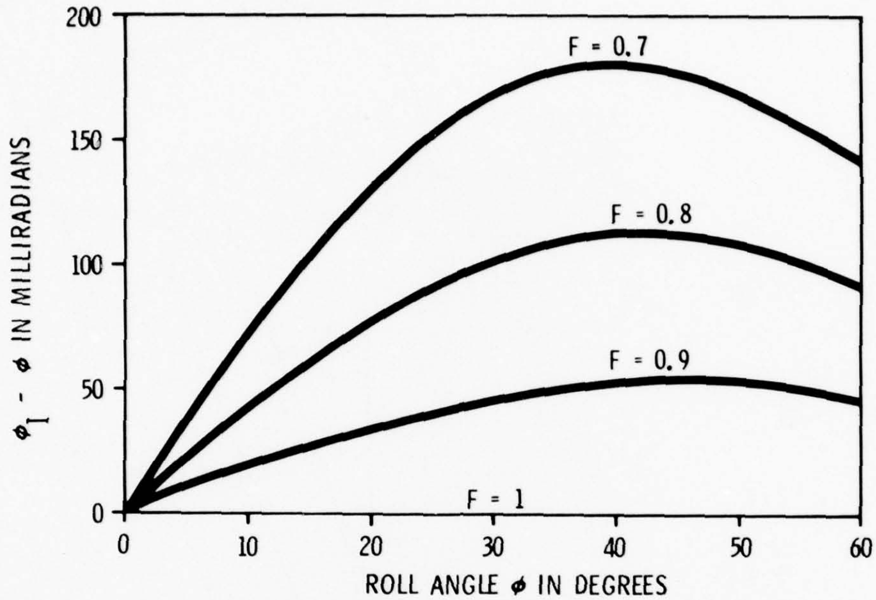


Figure 1. $\phi_I - \phi$ vs Roll Angle for Various Values of F

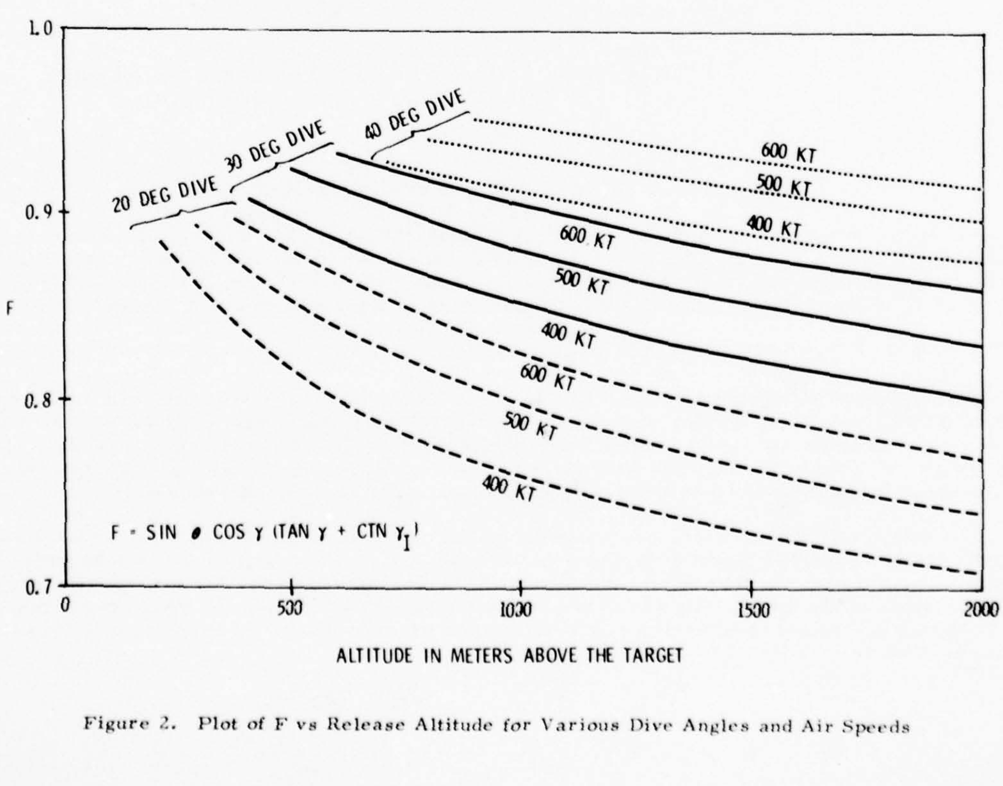
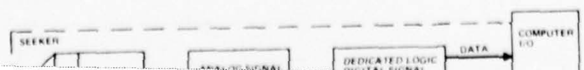


Figure 2. Plot of F vs Release Altitude for Various Dive Angles and Air Speeds



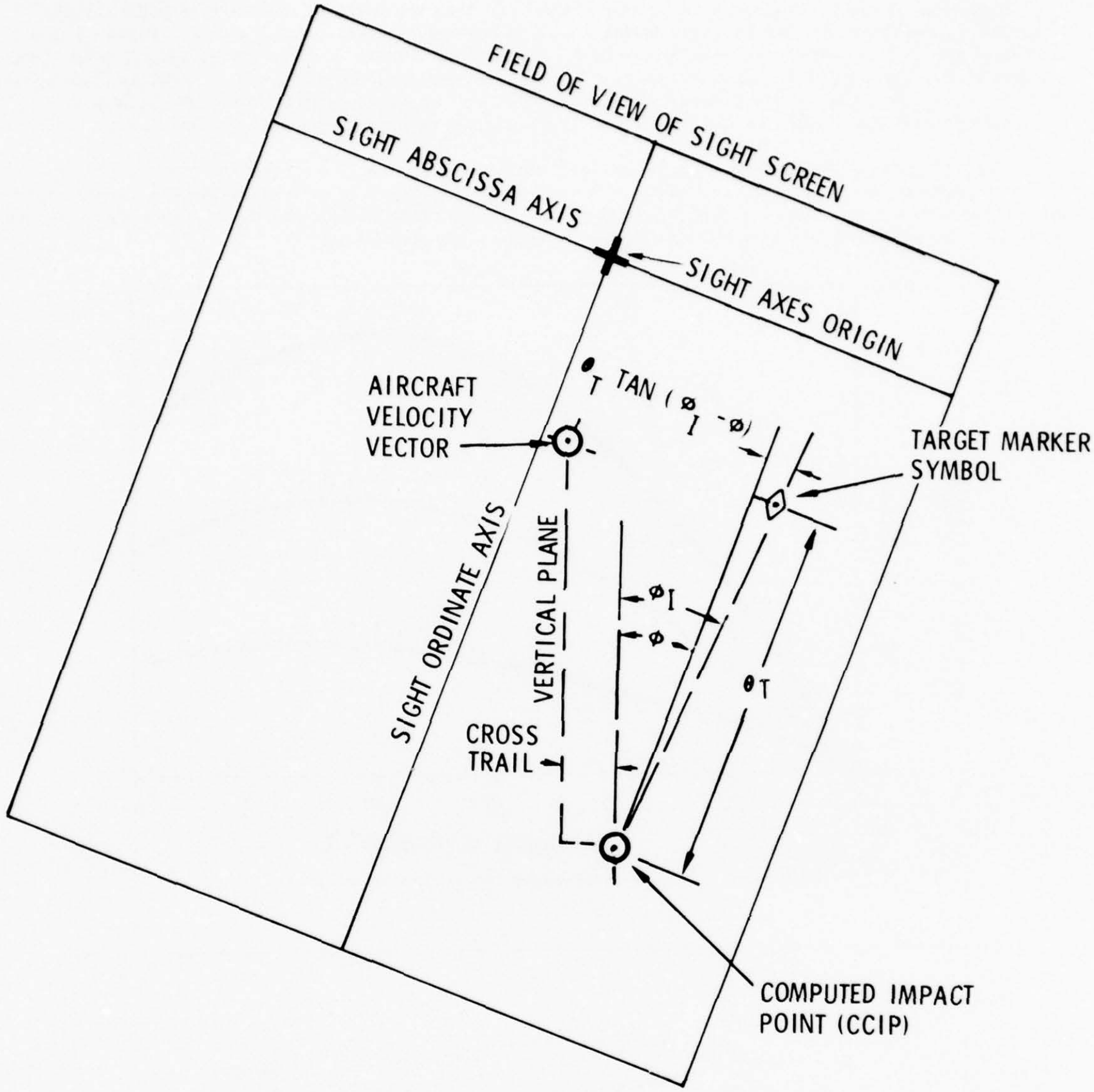


Figure 3. Sight Geometry in a 20 Degree Roll

From this computed impact point, displace the target marker symbol up the sight (parallel to the sight ordinate axis) by an amount θ_T (to be determined shortly) and to the right (parallel to the sight abscissa axis) by an amount $\theta_T \tan(\phi_I - \phi)$. This places the target marker symbol approximately in line with the track of the computed impact point if the pilot pulls straight back on the stick without unrolling.

III. ANGULAR DISPLACEMENT BETWEEN COMPUTED IMPACT POINT AND TARGET MARKER

The last question to be resolved is, "How large should we make θ_T?" The choice of θ_T is somewhat arbitrary, because it, together with the magnitude of the pull up acceleration, merely determines the time between pickle and release. Most pilots prefer to make this time as short as possible so that they can get rid of the bomb and begin their evasive escape maneuver as soon as possible. However, it still has to be long enough to allow time for making at least small steering corrections.

The only absolutely necessary constraints on θ_T are that it be neither negative nor so large as to point the target designating pipper at or above the horizon. If θ_T were negative, the release point would already be passed when the pilot pickles the target. If the pipper is above the horizon, the pilot can not place the pipper on the target. One way to insure that the pipper will always lead the computed impact point and still be directed toward the ground is to compute θ_T according to the expression derived in Figure 4. That is,

$$\boxed{\tan\theta_T = \frac{\left(\frac{V_x t_{PR}}{R_B}\right)\left(\frac{Z}{R_B}\right)}{1 + \left(\frac{Z}{R_B}\right)^2 + \frac{V_x t_{PR}}{R_B}}} \qquad (14)$$

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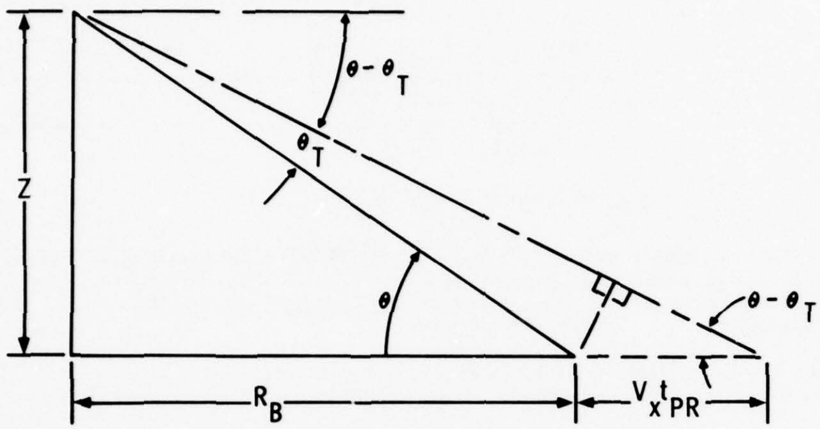

$$\frac{Z}{R_B + V_x t_{PR}} = \tan(\theta - \theta_T) = \frac{\tan\theta - \tan\theta_T}{1 + \tan\theta \tan\theta_T}$$

$$(1 + \tan\theta \tan\theta_T)\frac{Z}{R_B + V_x t_{PR}} = \tan\theta - \tan\theta_T$$

$$\left(1 + \frac{Z\tan\theta}{R_B + V_x t_{PR}}\right)\tan\theta_T = \tan\theta - \frac{Z}{R_B + V_x t_{PR}}$$

$$\tan\theta_T = \frac{\frac{Z}{R_B} - \frac{Z}{R_B + V_x t_{PR}}}{1 + \left(\frac{Z}{R_B}\right)\left(\frac{Z}{R_B + V_x t_{PR}}\right)} = \frac{V_x t_{PR} Z}{R_B^2 + Z^2 + R_B V_x t_{PR}}$$

$$\boxed{\tan\theta_T = \frac{\left(\frac{V_x t_{PR}}{R_B}\right)\left(\frac{Z}{R_B}\right)}{1 + \left(\frac{Z}{R_B}\right)^2 + \frac{V_x t_{PR}}{R_B}}}$$

Figure 4. Derivation of Expression for θ_T

This method succeeds in pointing the target designating symbol at a point on the ground which is beyond the impact point by an amount approximately equal to $V_x t_{PR}$. The parameter t_{PR} represents the time interval between pickle and release for an aircraft in straight and level flight. Selection of a value for t_{PR} which is between 1 and 4 seconds should result in an operationally acceptable time between pickle and release.

For example, let us compute θ_T from Eq. (14) for the following typical set of CCRP pickle conditions.

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Dive Angle	30 deg	$(V_x = 231.5\ m/s)$
True Air Speed	450 kts	
Altitude, Z	1117 m	$(\theta = 37.65\ deg)$
Bomb Range, R_B	1448 m	

For a value of $t_{PR} = 2.5$ sec, we compute the following value of θ_T for this case.

$$\left.\begin{aligned} \tan\theta_T &= \frac{\left(\dfrac{231.5\ \frac{m}{sec} \times 2.5\ sec}{1448\ m}\right)\left(\dfrac{1117\ m}{1448\ m}\right)}{1 + \left(\dfrac{1117}{1448}\right)^2 + \dfrac{231.5 \times 2.5}{1448}} = 0.1546 \\ \theta_T &= 153.4 \text{ milliradians} = 8.79 \text{ degrees} \end{aligned}\right\} \quad (15)$$

Note that use of the parametric value of 2.5 sec for t_{PR} yields a pipper placement which is 1.14 degrees ($\theta - \theta_T = 28.86$ deg) above the flight path angle ($\gamma = 30$ deg). This is between the two popular pipper placement schemes which place the target marker (1) in the pitch plane of the velocity vector or (2) on the aircraft boresight.

The parameter t_{PR} in Eq. (14) provides a degree of software control over the characteristic time between pickle and release in the dive-toss mode. It can be adjusted to suit pilot preference.

Equations (13) and (14) are the mechanization equations which satisfy the stated purpose of placing the target designating symbol (see Figure 3) in such a position that, after pickling the target, the pilot can pull straight back on the stick without first unrolling to a wings level attitude.

IV. DISCUSSION

4.1 Validity of Assumptions and Approximations

The foregoing derivation of the target pipper placement equations depended on two approximations. The first is that pipper motion during pullup is independent of bomb drag. The second is that the pipper moves in a straight line during a coordinated, turning pullup.

The zero bomb drag approximation really introduces no new pipper placement errors beyond those already present in current pipper placement schemes. The cross trail actually does change during the pullup in both the current schemes and in the scheme proposed herein, and the pilot must compensate for this variation by making a small steering correction during the pullup maneuver in either case. The new scheme is no worse than the present ones in this respect.

The second assumption, namely that the impact point moves in a straight line during the pullup will be valid to the extent that F, the denominator of Eq. (10), remains constant during the pullup. Of course, F does change during the pullup as the altitude and dive angle of the aircraft change. This alters the slope of the path of the impact point in the sight or HUD causing the impact point to move in a curved instead of in a straight line path.

To estimate the magnitude of this slope change, let us calculate the value of F both at pickle and again at release during a typical dive-toss delivery. The following statements summarize the pickle and release conditions.

Pickle:	450 kt speed, 1000 m altitude, 30 deg dive
Release:	450 kt speed, 863 m altitude, 19.7 deg dive

Using the figures from the above example to calculate F as in Figure 2, we find that

$$F = \sin\theta\ \cos\gamma\ (\tan\gamma + ctn\ \gamma_I) = \begin{cases} 0.867 \text{ at pickle} \\ 0.788 \text{ at release} \end{cases} \quad (16)$$

The corresponding values of $\tan(\phi_I - \phi)$ for a 20 degree roll are, from Eq. (13)

$$\tan(\phi_I - 20\ deg) = \begin{cases} 0.0484 \text{ at pickle} \\ 0.0838 \text{ at release} \end{cases} \quad (17)$$

The tasks resulting from a functional decomposition of the system may be allocated according to two philosophies :

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The maximum change in slope from pickle to release is 0.0354 radians in this example. The mean change in slope is 0.0177 radians. This will multiply by the angular difference, θ_T, between the impact point and the target designating symbol at pickle to cause a lateral pipper placement error. Once again, this is not necessarily a bombing error because the pilot can still compensate for it by nulling the steering signal during pullup. The whole point of this exercise is to see how big a compensatory steering correction he has to make.

The magnitude of θ_T, if computed according to Eq. (14) with t_{PR} = 2.5 sec and with the foregoing conditions at pickle, is 171 milliradians. This angle when multiplied by the mean change in slope between pickle and release would result in a lateral pipper placement error of about 3 milliradians. We can conclude from this example that the straight line impact point motion assumption results in an acceptably small if not negligible steering error signal.

In summary, the pilot steering corrections needed to compensate for pipper placement errors caused by the zero bomb drag and straight line impact path assumptions are no larger than those steering corrections needed in the current pipper placement schemes.

4.2 Additional Pilot Control Over Prepickle Pipper Motion

As mentioned earlier, the pipper placement scheme proposed herein gives the pilot a positive, direct azimuthal control over the pipper position, contrary to the currently operational pipper placement mechanizations which are keyed to the aircraft velocity vector. In order to change the azimuth orientation of a drift stabilized pipper, the pilot has to change the azimuth direction of the aircraft's velocity vector. This is an integration process with an inherent time lag. The pilot first has to roll the aircraft toward the direction in which he wants to move the pipper and pull back on the control stick. The pipper then gradually moves toward the desired azimuth.

In contrast, with the proposed mechanization, the pipper immediately rotates about the computed impact point on a lever arm equal to θ_T as the pilot rolls the aircraft (see Figure 3). The angle of rotation, ϕ_I, of the pipper is slightly greater than the roll angle, ϕ, itself. The pipper is stabilized against pitching and yawing motion, because (except for the small cross trail term and ejection velocity direction corrections in the impact point computation), θ_T, $\tan(\phi_I - \phi)$, and the computed impact point are all independent of the pitch and yaw attitude of the aircraft. However, the pilot can make a last second azimuth adjustment to place the pipper over the target simply by changing the roll angle of the aircraft.

The sensitivity of this pipper response to roll control action is proportional to the lever arm, θ_T. By increasing or decreasing θ_T, (through variation of the parameter t_{PR}) one can adjust the pipper roll sensitivity to match the amount desired by the pilots.

4.3 Adaptability to CCIP and Level Laydown Modes

The primary objective of this study was to devise a pipper placement scheme for dive-toss weapon deliveries which would not require the pilot, after designating the target, to unroll into a wings level attitude before pulling up to release. However, upon reviewing the resulting mechanization, one sees that it can be generalized to include CCIP and level laydown weapon deliveries as well. Traditional weapon delivery systems treat these as separate modes. Combining them essentially into a single mode would both simplify the software and decrease the number of mode selection decisions and actions imposed on the pilot.

A glance at Figure 3 shows that if θ_T equals zero, we have a CCIP weapon delivery mode. Hence, one can view the parameter t_{PR} in Eq. (14) as providing a continuum of weapon delivery modes with CCIP being one extreme, namely $t_{PR} = 0$. In other words, the same software that is used for the dive-toss mode could also provide the CCIP mode simply by setting $t_{PR} = 0$.

The level laydown mode is also akin to the dive-toss mode in that it requires the target designating symbol to be above the currently computed impact point by some amount θ_T. The chief difference is that θ_T must also be less than the angular distance between the line-of-sight to the impact point and the aircraft velocity vector, because with the aircraft in level flight, the velocity vector is already above the target. The second difference is that the level laydown delivery uses high drag bombs instead of low drag bombs.

Equation (14) for θ_T satisfies all of the above conditions. Its derivation (Figure 4) is applicable to both high and low drag bombs, and it directs the target designating symbol toward a point on the ground which is beyond the impact point. In fact, in level laydown the time between pickle and release is exactly equal to t_{PR}.

4.4 Incorporation Into An Operational Weapon Delivery System

The Litton LW 33B air-to-ground weapon delivery system includes the wings nonlevel pullup capability described herein. First operational deployment of these systems is on the McDonnell Douglas F4E's being delivered to Turkey and to Greece.

This figure merits two comments :

The natural reluctance on the part of users to place complete reliance on a new, untried innovation inhibited full utilization of the beneficial simplifications potentially achievable by combining weapon delivery modes. Instead, the "CCRP" mode still retains the traditional drift stabilized sight mechanization. However, the "LAYDOWN" mode does employ the target marker placement algorithm which permits wings nonlevel pullups.

As discussed in the preceding sections, the software for the algorithm is the same for level laydown deliveries as it is for dive-toss deliveries. Hence the pilot can perform a dive-toss delivery using the LW 33B even though the mode switch is in the "LAYDOWN" position. This dual mode mechanization offers pilots the unique opportunity to try out and compare both dive-toss mechanizations, wings level pullup versus wings nonlevel pullup, on successive passes simply by changing the mode switch from "CCRP" to "LAYDOWN".

V. CONCLUSIONS

5.1 Algorithm for Wings Non-Level Pullup

Displacement of the target marker symbol from the computed impact point by an amount θ_T parallel to the sight's ordinate axis and by an amount $\theta_T \tan(\phi_I - \phi)$ parallel to the sight's abscissa will permit the pilot to designate (pickle) in a roll and pull straight back on the stick without first unrolling into a wings level attitude. The equations for computing the quantities θ_T and $\tan(\phi_I - \phi)$ are

$$\tan \theta_T = \frac{\left(\frac{V_x t_{PR}}{R_B}\right)\left(\frac{Z}{R_B}\right)}{1 + \left(\frac{Z}{R_B}\right)^2 + \frac{V_x t_{PR}}{R_B}}$$

$$\tan(\phi_I - \phi) = \frac{(1 - F)\tan\phi}{F + \tan^2\phi}$$

where

F = $\sin\theta \cos\gamma (\tan\gamma + \text{ctn}\,\gamma_I)$

R_B = ground range to computed impact point

t_{PR} = arbitrary parameter nominally equal to 2.5 sec for dive-toss and level laydown deliveries or equal to zero for CCIP deliveries

V_x = ground speed

Z = altitude above target

γ = flight path angle relative to the horizontal (positive in a dive)

γ_I = impact angle of computed trajectory with respect to the horizontal

θ = depression angle (below the horizon) of the line-of-sight to the computed impact point

ϕ = aircraft roll angle (positive for right wing down)

5.2 Effect of Approximations

Approximations employed in deriving the foregoing equations introduce acceptably small pipper placement errors for which the pilot can compensate by following the steering commands indicated by the computer during the wings nonlevel pullup.

5.3 Pipper Response to Rolling Motion

The foregoing pipper placement algorithm provides the pilot with a direct, positive control over the azimuth location of the pipper prior to pickle. By rolling the aircraft one way or the other he can move the pipper back and forth across the sight.

In reality, especialy in on-board system, power is distributed to subscribers from a central point [15] and it is no use providing a highly-secure damage-proof

5.4 Adaptation to CCIP and LAYDOWN Modes

Through variation of the parameter t_{PR}, the foregoing pipper placement algorithm can also accommodate CCIP and level laydown deliveries in addition to the banked pullup, dive-toss delivery for which it was designed.

APPENDIX A. PROOF THAT THE PARAMETER F IS LESS THAN UNITY

By definition

$$F = \sin\theta \cos\gamma \,(\tan\gamma + \operatorname{ctn}\gamma_I) \qquad \text{(A-1)}$$

where the three angles, θ, γ, and γ_I are, respectively, the angles between the horizontal plane and (1) the line-of-sight from the aircraft to the computed impact point, (2) the weapon's velocity vector at the computed release point, and (3) the weapon's velocity vector at the computed impact point. The downward trajectory curvature caused by gravity guarantees the following inequalities for all dive angles less than 90 degrees.

$$\gamma < \theta < \gamma_I \qquad \text{(A-2)}$$

By virtue of the above inequalities

$$\operatorname{ctn}\gamma_I < \operatorname{ctn}\theta \qquad \text{(A-3)}$$

Substitution of (A-3) into Eq. (A-1) yields

$$F < \sin\theta \cos\gamma \,(\tan\gamma + \operatorname{ctn}\theta) = \cos(\theta - \gamma) \qquad \text{(A-4)}$$

According to (A-2), $\theta \neq \gamma$ for any dive angle less than 90 deg, and (A-4) thus reduces to

$$F < 1 \quad \text{Q.E.D.} \qquad \text{(A-5)}$$

However, if a higher connectivity is required, the nodes are completely modular ; two four-port nodes may be connected together to form a single six-port node.

EXPENDABLE DIGITAL COMPUTERS IN TACTICAL MISSILES TRENDS AND TRADEOFFS IN SOFTWARE AND HARDWARE

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Computers are shown to be an effective way of achieving lower cost and improved performance in tactical guided missiles. Within the confines of currently available semiconductor technology a tactical missile computer is shown to be best applied in a central processor architecture. In this type of system a single computer performs the computational tasks of each of the guidance units' subfunctions and also serves as the overall system integrator. Two examples of this structure are given.

The complexity of the required tactical missile computer depends on the complexity of the missile to which it is applied. Three classes of missile complexity and four classes of computers are identified. Which computer is required for each of the missile complexity classes is then demonstrated.

Missile computer requirements differ considerably from those of other computer systems. The nature of and reasons for these differences are discussed.

Finally, an extrapolation of current technology trends is made and projections as to the effect of these trends on tactical missile computer architecture are presented.

The computer revolution is one of the most pervasive and dramatic forces in modern electronics. From its very beginning, this revolution has influenced ordnance systems. In fact, the ostensive purpose of the first true electronic computer (ENIAC) was to support military systems. Avionic systems were also quickly swept up in the rush to computer control. However, as rapidly as most military systems have assimulated computer technology, it has only been the advent of modern Medium and Large Scale Integration that has made this technology applicable (or at least appealing) to tactical guided missiles. The cause for this tardiness is the unique nature of the size, power dissipation and recurring cost requirements of such missile systems. Even with modern technology, these requirements are still operative and make the tradeoffs involved in tactical missile guidance different from those in almost all other forms of ordnance.

This paper will present one viewpoint on how and why the requirements of tactical missiles are unique and show how these requirements have led to the configurations that are presently being designed for production systems. The discussion of what is currently being implemented will be followed by a description of how these present-day design decisions will be changed by the technologies that are now beginning to emerge.

Advantages of Computers in Missiles

In light of the current industrial trends, it almost seems superfluous to justify the incorporation of a computer in any system. Nonetheless, it is instructive to review the actual benefits that are obtained from the use of a computer, particularly when, as will be seen, many of these benefits can be obtained for some systems with methods more cost effective than the use of a full scale computer. For the purposes of this discussion, a computer will be defined as a digital programmable processor consisting of both a program memory (PM) and an operand memory (OM) operating in conjunction with a central processing unit (CPU) which has the capacity to generate memory addresses, interpret instructions and perform arithmetic operations.

Reasons for using a computer are generated from two basic sources: the advantages of a digital system in general, and the advantages of the computer structure in particular. Looking first at the advantages of digital systems in general, it is seen that they should:

1. Require no calibration - This reduces the cost of assembly.

2. May be constructed from a set of standard modules (gates, arithmetric logic unit multiplexers, etc.) - This allows the cost effective use of LSI and MSI technologies in that relatively complex modules may be combined to realize a variety of functionally different systems. The large market produced in this way lowers the component cost to each individual system.

3. Suffer no system performance degradation with age or temperature.

4. Operate with perfect repeatability and predictability.

5. Intrinsically perform logical and arithmetic functions - These are exactly the operations required by control and signal processing algorithms.

The basic computer structure adds to these advantages:

1. Programmability and reprogrammability - This allows a wide variety of functional changes to be made in the system which in turn allows correction of errors even in production, and considerably slows product obsolescence. Interestingly enough, this also makes possible the size and cost reductions that can be obtained with LSI. Normally, the greater the level of integration, the more complex the circuit and hence the more specific it is to its application. Most sophisticatal applications, in advanced military systems, have limited markets and hence limited production quantities, making it difficult to recoup the large non-recurring cost of LSI. However, computers are highly complex elements with their system specifity in their program not their circuit structure. This allows LSI computers to enjoy a large production volume even though there may be only a limited demand for each of their individual system applications.

2. Intrinsically a time shared device - The basic nature of a Von Neumann type computer is to use a single arithmetic element to perform a large number of possibly unrelated arithmetic computations. This is "time sharing with a vengeance" and can save a good deal of hardware over a "dedicated logic" digital approach*.

A key point in the above discussion is the emphasis on cost and performance benefits. It is not anticipated that tactical missile computers (TMC's) will be frequently reprogrammed nor that they will be expected to perform any extensive interfacing with human operators. They are incorporated in a guidance unit only because they are the most cost effective way of obtaining the required performance in the missile's single flight.

Computer Utilization in Present Missile Designs

Recognizing the advantages that computers have to offer, this section will show how these advantages are being harvested in present missile designs. Because of the dynamic nature of recent technological advancement, the time frame in question must be carefully noted. The discussion here is of existing rather than future designs.

As defined here, missile guidance units are composed of a number of distinct functional processes. While there is a wide variety in the content of each of the members of the guidance family, a list of the most common subfunctions would include the:

target tracking seeker
rear data link
target detection device (fuze)
launcher interface
navigation
flight control

Each of the processors performing these subfunctions is realized with a mix of analog and digital circuitry with the digital circuitry performing lower speed signal and data processing functions. The digital functions may themselves be partitioned on the basis of speed. Most subfunctions have a requirement for some high speed (real time) computational logic. Examples are digital spectrum analysis, demodulation and decoding. These functions require speeds in excess of that available from computers and are performed with specialized dedicated logic. Following the high speed digital computation are a series of functions which tolerate slower speed numerical calculation and use a large amount of logical interpretation. Examples of these functions are threshold calculations, parameter estimation and most post-detection processing. These functions not only can be performed by a computer, but also make the most effective use of a computer's logical capabilities.

Figure 1 exemplifies the approach Hughes' current designs have taken in applying computers to missile guidance. This is a central processor system where a single computer services each of the subsystems. The purpose of the computer is to perform those parts of the subfunction calculations that are most amenable to computer implementation and at the same time serve as the integrating element between all the subfunction processors.

The alternative to the Central Processor is a Distributed Processor. This would dedicate an individual computer to each of the subsystems with a communications network connecting them. The author's choice of a centralized system is based on an evaluation of presently available technology (with the definition of "available" as derived from the committment phase of the project in question). Currently there appear to be four types of computers that are applicable to tactical missile systems. They are described in Table #1. The first two computers in the table are based on commercially available microprocessors such as the Intel 8080 and the Texas Instruments 9940. These computers are quite low in recurring cost, but are also quite slow. The basic structure of these NMOS devices allows little flexibility in either their architecture or their instruction set. The next computer type in the table is realized with bipolar LSI technology. Examples of these computers are fabricated from the Intel 3000 and the American Microdevice 2900 families. The bipolar logic allows relatively high speed operation while the use of LSI techniques results in fairly high densities at low cost. Further, they are microprogrammable which permits the tailoring of their instruction sets and their architectures to the specific applications at hand.

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*Some of the recent researchers in distributed computer systems have facetiously referred to this single Arithmetic Unit approach as one of the greatest setbacks to computer architecture. However, if early systems had been based on a more operationally parallel structure, the Von Neumann system would have been invented anyway, just to save hardware. As will be seen, this is the key to an economically expendable computer.

The fourth alternative in Table 1 is a fully custom minicomputer using mostly MSI logic. This machine uses very high speed bipolar devices and allows complete freedom in its architectural design. The result is a very high throughput rate, and unfortunately, a high cost and fairly large power dissipation.

A detailed study of the requirements of each of the subfunction processor shows that they call for the throughput capabilities of a bipolar computer. Consequently, if a distributed computer system were proposed for a tactical missile it would force the designer to distribute the more expensive computers shown in the last two entries of Table 1. Trade off studies have shown this type of system to be less cost effective than a central high speed computer.

These trade off studies are confirmed by a traditional method of estimating computer costs known as Grosche's law. Grosche's Law is an empirically derived rule which states that a computer's cost is proportional to the square root of its throughput. In other words, the throughput of a single computer can be doubled by architectural modifications for a 40 percent increase in cost. This is obviously less expensive than using two computers.

Grosche's Law was formulated in terms of the early mainframe computers, but it has been shown to be relatively accurate in predicting costs within a given modern semiconductor technology. It should be noted, however, that the rule does not appear valid in comparing computer systems realized in different technologies. Several, say, NMOS computers may indeed represent a lower cost solution than a single bipolar computer. The majority of the subfunction speed requirements do not allow this to be a valid missile computer alternative within currently available technology.

Which of the last two entries in Table 1 are chosen for a given missile depends on the complexity of that missile. One may postulate three classes of complexity in modern tactical missiles. They are outlined in Table 2. The first (Class A) is a very complex and hence very expensive missile manufactured in low quantities. Missiles of this type provide high performance and require extremely sophisticated electronics. While cost is always a concern, the primary design criterion is performance with reliability. The second missile (Class B) is a medium cost, medium performance configuration. Its mission requirements do not demand the ultimate in guidance sophistication. On the other hand, production is relatively large and cost becomes a design constraint as important as performance. The third category of missile (Class C) has a relatively simple guidance structure and is produced in massive quantities. Mission requirements are less demanding, but recurring cost is of extreme importance.

In general, the more complex the missile's function, the more powerful a computer it will require. This is largely due to the extensive amount of logical operations needed in the post detection signal processing of the more sophisticated missile seeker and rear links. As a consequence, MSI minicomputers are used in class A complexity missiles while bipolar chip sets are found in class B missiles. The high production class C missiles do not yet use computers. This is largely due to the simplicity of their on board processing. While the required functions must be performed in real time, they are not complex enough to justify the cost of an embedded expendable computer. Actually the class C production volumes in the hundreds of thousands justify the development of custom LSI devices (which were not in existance when the present generation of missiles was developed) for the specific functions found in these missiles.

Table 3 summarizes the observations that have been made on computer selection for the various complexities of tactical missiles. Some typical memory requirements have also been included to give an indication of the program sizes that are being discussed.

The concepts that have been presented may become clearer after investigation of some examples. The first example will be a class B missile used in an air to ground mission and employing an IR imaging guidance system. A block diagram is shown in Figure 2. In operation, the seeker output signals are first sent to an avionics display. The pilot slews the missile seeker head until the target image is centered in the field of view (FOV). At this time, the missile will begin automatic tracking. As soon as the missile acquires the designated target it will be launched and begin to home on the target by using signals derived from the target's image parameters.

From the perspective of this paper, the guidance unit contains four major components: the seeker electronics, the avionics interface, the autopilot and the computer. (Fuzing and power regulation functions may use the computer but are not discussed here).

The sensing element of the seeker is an IR imager mounted on a gimbaled platform. This unit generates video signals which are first processed by analog circuitry. The analog section provides amplification, AGC control, prefiltering and coarse gating. Following this is a digital scan converter which outputs display compatible composite video and provides A/D conversion and buffering into the computer. The computer uses the scan converter data to maintain target discrimination against background, calculate target gate sizing, and compute target angular displacements which are used to provide seeker servo and autopilot commands. The remaining major component of the seeker is a dedicated logic frame to frame correlator which reduces the system blind range in the terminal phase of the mission and assists in overcoming temporary loss of target lock due to imperfect sensor stabilization. Outputs from the frame to frame correlator replace those of the scan converter at the command of the computer.

Target angle signals are used to calculate seeker servo and autopilot control commands. The computer also performs some slow speed calculations for these servo loops. Examples of these are the g-bias gimbal restoration computations for the seeker servo and the lead and lag computations for the autopilot. The majority of the higher speed stabilization filtering is left to conventional analog circuitry.

All mode changes and system control functions are reserved for the computer. Avionics slewing commands are passed through the computer, while high speed video data is passed directly through the umbilical to the display.

γ Weapon flight path angle, the angle between the weapon's velocity vector and the horizontal plane (positive in a dive), at the (computed) release point.

γ_I Flight path angle of weapon at (computed) impact

21-4

In the design of this system, the cost and computer availability to a fairly large degree determined the functional partitioning of the system and to a certain extent the complexity of the algorithms used. It is interesting to note that the use of the computer to perform the basic tracker algorithms led to the relegation of most of the high speed servo control logic to analog circuitry. This was necessary because of the computer speed limitations. The particular partitioning used is based on the belief that while the servo loops are easily performed by analog methods, and are not likely to change during the life of the missile type, the tracker logic is most efficiently performed with the logical and computational resources of a computer and is likely to change with new mission requirements or understanding.

Figure 3 shows a second example of computer utilization in tactical missiles. This system is a class A missile using dual mode radar guidance to intercept a highly maneuvering target in an air-to-air encounter. This system contains five major subsystems: the seeker, the rear receiver, the autopilot, the avionics interface and the computer. (Fuzing and power regulation again will not be discussed).

In a typical operation this system utilizes a semiactive midcourse and an active terminal guidance with the same seeker receiver used in both modes. Radio frequency (RF) signals are received by the monopulse-antenna, converted to IF and then to video after pre-amplification and filtering. The signals are then A/D converted and passed through a dedicated logic Fast Fourier Transform processor which performs doppler filtering and detection. The detected information is now sent to the computer which performs target discrimination (against clutter) and tracking functions to extract target angle information. These data are used to calculate seeker autopilot steering commands, and system mode control parameters and to control the seeker receiver.

In addition to generating steering commands to the autopilot and seeker servo control systems, the computer also performs all of the stabilization computations. This essentially reduces the autopilot and seeker servo circuitry to A/D and D/A converters.

The computer also receives data from the rear receiver. This is an essentially analog receiver with a digital message demodulation circuit. Messages are decoded and applied to the other systems by the computer itself.

Again it can be seen that the computer performs an integration function between the major subsystems of the guidance unit and simultaneously performs many of the real time computational and logical functions required by these subsystems. The difference between class A and class B missiles is the amount of computing power available and the degree of computer involvement in the subsystem functions. The more complex the guidance the more powerful the computer and the more functions the computer must perform. Hughes has found the seeker signal processing functions to require the most speed and computational complexity. Consequently, these are the functions that normally set the requirements for the missile computer. Once a match to the seeker requirements has been made, the other, slower functional blocks are fitted into the program time loading of the computer.

Characteristics of Tactical Missile Computers

Two types of available computer have been identified as being applicable to tactical missiles. One is based on a bipolar LSI chip set and the other is a custom high speed MSI design. In both cases the ordnance designer is given a good deal of flexibility. This is fortunate for the tactical missile environment generates a set of unique requirements. The military system which comes closest in structure to the tactical missile computer (TMC) is the avionics computer (AC). The two types of TMC are compared with a typical AC in Table 4. In this table the first entries are essentially physical parameters. The most apparent difference between a TMC and an AC is that the former is an embedded unit while the latter is a replaceable subassembly.

Avionics systems are in essentially continuous operation and must be constructed in such a way that they can be continuously maintained and repaired quickly in the field. Conversely, missiles are used only once and will not encounter anywhere near the failure rate that avionics systems encounter. For this reason the TMC can be totally embedded in the missile; using the missile power supply and a hardwired system interface. The AC must on the other hand, be easily accessible and replaceable. It must have its own power form generating circuitry and interface with the rest of the avionics system through a set of standard connectors. The missile's simple system interface is one of the factors that helps reduce the size and power dissipation of the TMC. Any improvement in these areas is crucial as the volume and power supply capabilities of a missile are severely constrained when compared to those of an Avionics system.

Another major difference between Avionics and Missile Systems is the thermal environment. An Avionics system must operate continuously for an indefinite time. Consequently, it requires an efficient, usually active, cooling system such as forced air. Ignoring flight and ground testing, a missile need only operate over the duration of its flight; a time that rarely exceeds a minute in tactical systems. Consequently, operational cooling is only required during this period of time. Usually a TMC operates with a passive cooling or heat sinking system that is designed to be only effective during the flight of the missile and is therefore less costly, smaller and lighter than that required in avionics. An external auxiliary cooling system is provided for system testing.

A TMC differs from an AC in functional configuration as much as it does in physical configuration. For instance, while both types of computers require 16 bit data words, Avionics systems are gradually evolving toward 32 bit word lengths while it is unlikely that the TMC will ever need this added precision. It is true that some missile functions require double precision operations, but these functions are usually found in the less time consuming portions of servo control loops and do not represent a significant time load on the computer.

Both types of computers require efficient addressing structures, hardware multiplication and division instructions and efficient branching capabilities. However, TMC's are not reprogrammed with the same frequency as AC's. For this reason they do not require many of the programming conveniences found in AC's or general purpose computers. For instance, floating point hardware and program relocatability have little utility in the TMC. This decrease in complexity is balanced by the fact that a TMC must operate at a higher pilot and system by having the system anticipate the wings nonlevel pullup and position the target marker accordingly. Such anticipation is achievable, for example, by positioning the target marker to the left of the aircraft velocity vector, when the aircraft is in a bank to the left, and to the right of the aircraft velocity vector, when the aircraft is in a bank to the right.

throughput rate than an AC. As seen in Table 4, the simplest TMC's operate at about the throughput rate of the most advanced AC while the more powerful custom designed missile computers require a throughput rate four times that of a typical Avionics processor. (Special applications such as Real Synthetic Array mapping can require a typically high throughput rate in some advanced avionics).

Perhaps the greatest difference between an Avionics and a Missile Computer is in their memory structure. To begin with, the simpler tasks of the TMC require far less total memory than found in the AC. In addition, the structure of this memory is different. A missile computer is a black box, it is programmed once and always executes this program (except for design changes that must be incorporated at a factory or depot level). The typical AC, on the other hand, is reprogrammed often. Different programs are used for different mission and encounter situations, instrumentation runs, training exercises, etc. Consequently, a TMC's program is normally stored in hardware Read Only Memories (ROMs) whereas an AC's program is stored in magnetic core memories which are programmed from magnetic tape at the beginning of each mission. The capability of using fast, dense ROM program memory is one of the principal ways by which the TMC achieves its high speed.

The embedded nature of the missile computer also yields some functional simplicity. Because the input/output interfaces may be hardwired and specially designed, it is not necessary for the TMC to communicate through a generalized interface. This saves both hardware logic and interfacing time.

The last differences shown in Table 4 are probably the most obvious. A tactical guided missile is a piece of expendable ordnance, an aircraft is not. Consequently, missile components must be lower in cost and produced in larger quantities than those for aircraft.

Programming the Tactical Missile Computer

The previous section has shown that the special requirements of the tactical missile application yield a unique computer structure. These same requirements reverse some of the current trends in the way in which computers are used. Throughout most of the computer industry there has arisen a consciousness of the extreme cost of software. Computer programming is very labor intensive. Moreover, modern computer programs have become highly complex. This complexity has led to systems which are extremely difficult to validate or even verify. Further, most computer controlled systems are constantly being revised, updated or improved. This results in the high cost of software becoming a recurring one. As a result almost any price will be paid to simplify the preparation, documentation and maintenance of software. High Order Language (HOL's) are becoming mandatory for most military systems and in addition, there are several on-going efforts to develop a single standard HOL for all military applications. Coupled with the increasing cost of software is the declining cost of hardware. This has led many system designers to use hardware mechanisms to absorb software functions. An example of this trend is the growing popularity of hardware based floating point arithmetic elements.

In most military systems these trends are completely justified. However, in current tactical missiles, they are not. The use of a HOL can incur a significant penalty in execution speed and program memory size over assembly language programs. These in turn result in increased hardware complexity and cost.

As described in the previous section, a missile computer is an embedded "black box". It is reprogrammed very infrequently and then only at the factory or depot level. Consequently, software costs are nonrecurring. Hardware costs, however, are still recurring. Further the large production quantities involved with most missile systems generate a large multiplying factor for the cost of any hardware addition. Missile systems simply cannot affort to use a HOL and all programming is done in Assembly Language. Further, the decision to add ahardware element such as a floating point arithmetic element is made on the basis of its impact on system performance and life cycle cost; not on its effect on simplifying the programming task.

This does not mean that software costs are ignored in the development of TMC systems. Even if it is nonrecurring, the high cost of program development cannot be ignored, especially when development is bid in a competitive environment. As a consequence most of the modern software development methodologies are applied to missile design. These include the use of "top-down" programming with "chief programmer" type teams and the use of structured programming.

The TMC's software, like the TMC itself is an embedded part of the missile. Cost trade-offs involving this missile component must be made on the basis of overall system life cycle cost rather than just the cost of the software or hardware alone.

Future Tactical Missile Computer Designs

To this point the discussion has centered on the trends and trade-offs involved in present TMC design. The central processing approach using one of two generic types of computers has been chosen based on existing technology and the examples used have been of systems that are currently in prototyping or production. This section will present the authors' projections on how future requirements and technologies will effect the next generation of TMC's.

The current trend in missile design is to demand more functions in a smaller sized and a lower cost structure. The next generation missile will require the complexity of a class A missile (see Table 1) with the price of a class B missile. Further it can be expected that these functions will be performed in a 5" - 8" diameter airframe as opposed to the 12" - 15" diameters in today's missiles. Some of the driving forces for the increased guidance complexity are the desire for multimode missiles which might contain both IR and radar (millimeter wave) guidance capability. Further, in the interest of lower cost systems there is a strong desire to decrease the allowable guidance error budget and hence allow smaller warhead sizes. All these added complexities will have the net effect of increasing the amount of function performed by the tactical missile computer system.

Future semiconductor device technology will offer two alternatives to meeting the demanding requirements of new tactical missiles. The architecture of the future TMC will depend on which type of semiconductor technology is chosen. The first of the technology types is very high speed logic. Included in this option

The partial derivatives $\partial t_f/\partial Z$ and $\partial t_f/\partial V_Z$ can be expressed analytically (for a zero drag bomb) as follows.

are Gallium Arsenide Field Effect Transistor (GASFET) logic, Transverse Electron Devices (TEDs), Josephson junction logic and Dielectrically Isolated Emitter Coupled Logic (DECL). These technologies will offer subnanosecond propagation delays and with the potential of speeds almost an order to magnitude better than today's components. A TMC based on these super fast devices would again be a central processor. The increased functional capabilities would be obtained through the increased speeds and extended time sharing available. Unfortunately, the future high speed technologies will, at best, and the same densities and power dissipations of the present high speed logic families. Further, he motivating force behind their development is the mainframe computer market. It is unlikely that they will ever be available at low functional cost or be produced for a large commercial market. The super fast devices are currently in research laboratories and their availability to missile computer design is at least five years in the future.

The second new technology type is the very dense LSI logic. Included in this option are Integrated Injection Logic (I^2L), Silicon on Saphire Complementary Metal Oxide Semiconductor (SOS CMOS) logic, and High Performance Metal Oxide Semiconductor (HMOS) logic. These families will probably never exceed the speed of present bipolar logic families but will offer higher density, lower power dissipation, and lower cost functions. The basic driving force behind these technologies is the commerical market and such devices are already becoming available.

Applying the high density LSI families to the more demanding requirements of future missiles will require systems that contain several parallel computers. This distributed processing approach has up to now been found inferior to central processing (single high speed computer). The penalties that have reduced its acceptance are primarily due to interprocess communication and synchronization. It is the author's opinion, however, that the nature the TMC's application will ameliorate many of these difficulties. As was described previously, the TMC is used to perform the computational and logical tasks of the various subsystems of the missile guidance unit and at the same time integrate the functions of these tasks. If one or perhaps two microcomputers are dedicated to each subfunction and if another microcomputer is dedicated to the integrating function, each of the microcomputers would operate relatively independent of each other with the intercomputer communications being relatively slow. The reason a distributed system of this sort is not used now is that a certain minimum speed is required of the subsystem dedicated computers. Currently this speed is available only in the bipolar chip sets which are too expensive to proliferate through the system. The new technologies will make this speed available in single chip processors at a low cost and thus lead to a cost effective structure.

The distributed computer approach to the TMC is shown in Figure 4. This is the authors' prediction of the next generation of TMC. In addition to being cost effective, such a system will lead to a more modular system and will probably eliminate the two tier approach to design. Both class A and class B missiles will use the same type of microcomputer components shown in the figure, the difference being in the number of processors and the level of integration.

Conclusion

Tactical missile computers are already used to or proposed for many of the computational and logical processes for many of the subfunctions found in tactical missile guidance units. In addition they perform the overall integration and control of these subfunctions. Presently, TMC's require throughput rates that are only available in computers based on bipolar technology. There are two types of such computers available, one based on LSI bipolar chip sets and the other based on a custom MSI design. The bipolar chip sets offers a relatively low cost, consume very little volume, and have medium power requirements. Unfortunately their instruction throughput rate is only on the order of 400 - 500 KIPs. Computers of this type are used in medium complexity missiles having production quantities up to 30,000. For higher complexity, lower production missiles, the custom architectures based on available MSI devices are used. These computers feature very high throughput rates but also incur additional volume, power and cost penalties. Cost tradeoffs have shown that a central processing system architecture is the most efficient in the light of available technology.

Missile computers are considerably different from those found in avionics systems. In general, they must be faster, lower cost and smaller than avionics computers. Fortunately, proper utilization of the TMC's embedded nature, shorter operating time and infrequent reprogramming allows the meeting of its rather demanding performance objectives and physical constraints.

It is expected that the new semiconductor technologies will have a dramatic effect on tactical missile computer architecture. High density, low cost processors with the speeds of currently available in bipolar device technology will make LSI microprocessors cost effective in distributed system architectures in tactical guided missiles.

This new distributed structure will reduce the computer hardware differences between the complexity classes of missiles and will in addition expand the functional utilization of computers. Such improvements are mandatory if the conflict between the increasing mission demands and the decreasing volume resources of future guided missiles is to be resolved.

or

$$\tan(\phi_1 - \phi) = \frac{(1 - F)\tan\phi}{F + \tan^2\phi} \qquad (13)$$

CLASS/DESCRIPTION	THRU PUT [1] (KIPS)	CPU POWER DISSIPATION (WATTS)	CPU [2, 3] COST/UNIT $	PC BOARD AREA CM^2
8 BIT MOS μ PROCESSOR	~25	2	75	160
16 BIT MOS μ PROCESSOR	~170	2	200	160
BIPOLAR LSI CHIP SET	~500	20	1000	400
MSI CUSTOM MINICOMPUTER	~2100	60	2000	600

1 16 BIT OPERATIONS
2 ~1000 QUANTITY
3 COST FIGURES ARE FOR RELATIVE COMPARISON ONLY

TABLE 1. MISSILE COMPUTER CLASSES

CLASS	EXAMPLE	MISSION/GUIDANCE	APPROX COST	APPROX PROD VOL	APPROX PROD RATE
A	PHOENIX	AIR TO AIR/ RADAR SEMIACTIVE/ACTIVE GUIDANCE	HIGH	1,000	55/MO
B	MAVERICK	AIR TO GROUND/ TV OR IR IMAGING	MEDIUM	30,000	300/MO
C	TOW	SURFACE TO SURFACE/ WIRE GUIDED	LOW	300,000	3,000/MO

TABLE 2. MISSILE CLASSES

MISSILE CLASS	COMPUTER TYPE USED	PROGRAM WORDS	OPERAND MEMORY WORDS
A	MSI MINICOMPUTER	16K	3K
B	BIPOLAR CHIP SET	4K	256
C	NONE		

TABLE 3. COMPUTER UTILIZATION IN MISSILES

CHARACTERISTIC	BIPOLAR LSI CHIP SET MISSILE COMPUTER	MSI CUSTOM MISSILE MINICOMPUTER	TYPICAL AVIONICS COMPUTER
FORM FACTOR	EMBEDDED COMPUTER FORM FACTOR TAILORED TO MISSILE	EMBEDDED COMPUTER FORM FACTOR TAILORED TO MISSILE	REPLACABLE UNIT FORM FACTOR ATR BOX
TOTAL COMPUTER CIRCUIT BOARD AREA	570 CM^2	1200 CM^2	5600 CM^2
POWER DISSIPATION	25 W	130 W	~500 W
THERMAL ENVIRONMENT	PASSIVE HEAT SINK	PASSIVE HEAT SINK	FORCED AIR
DATA PRECISION	16 BITS	16 BITS	16 - 32 BITS
THRU-PUT	440 KIPS	2.1 MIPS	500 KIPS
WHEN REPROGRAMMED	AT FACTORY OR DEPOT ONLY	AT FACTORY OR DEPOT ONLY	FIELD REPROGRAMMING ON A DAILY BASIS
MEMORY SIZE	4.25K WORDS	19K WORDS	32K - 64K WORDS
PROGRAM MEMORY TYPE	ROM	ROM	MAGNETIC
INPUT/OUTPUT STRUCTURE	SPECIALIZED	SPECIALIZED	GENERAL
COST OF COMPUTER	$2K	$7K	$70K
PRODUCTION VOLUME	25,000	> 1,000	< 1,000

TABLE 4. COMPARATIVE COMPUTER CHARACTERISTICS

ALTITUDE IN METERS ABOVE THE TARGET

Figure 2. Plot of F vs Release Altitude for Various Dive Angles and Air Speeds

21-8

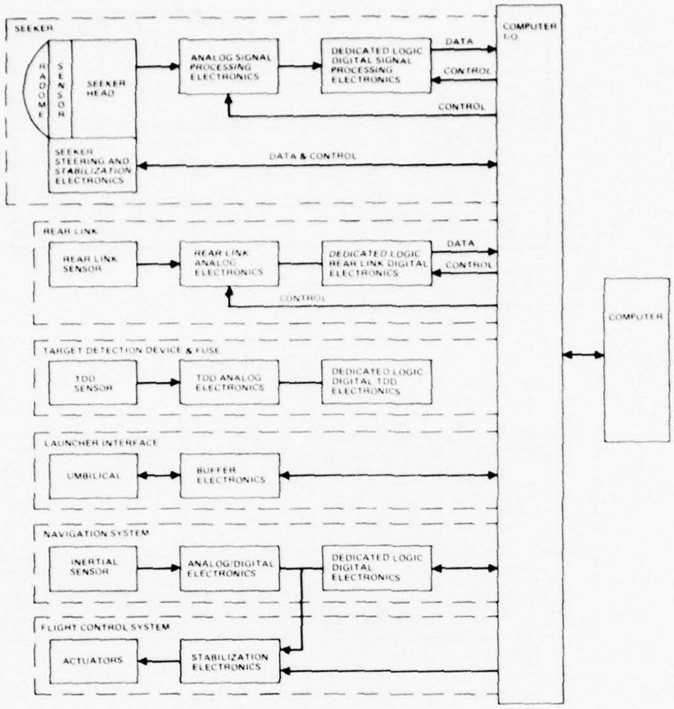

FIGURE 1. GENERALIZED TACTICAL MISSILE GUIDANCE UNIT USING A COMPUTER

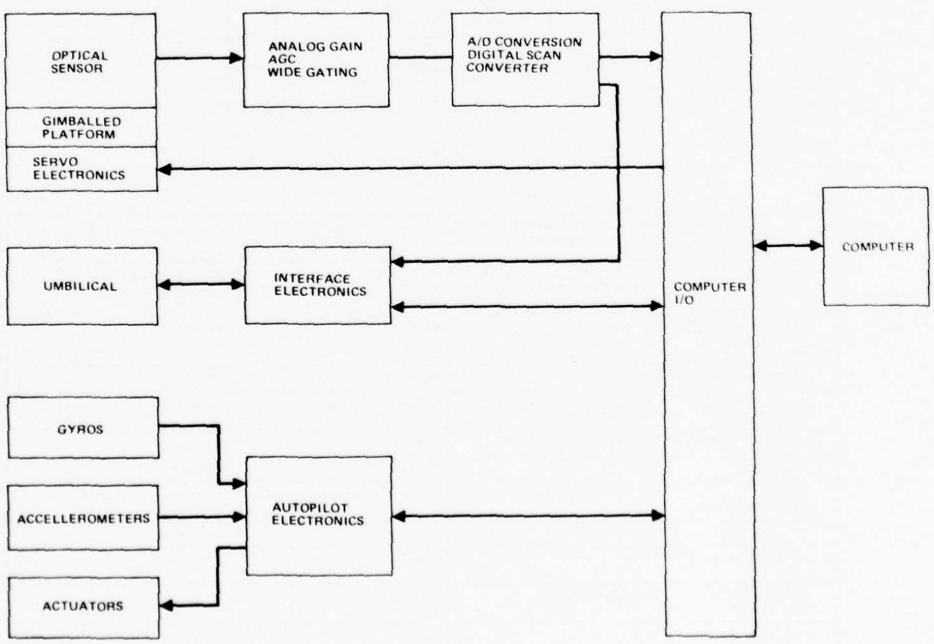

FIGURE 2. CLASS B COMPLEXITY MISSILE UTILIZATION OF A COMPUTER

point and still be directed toward the ground is to compute θ_T according to the expression derived in Figure 4. That is,

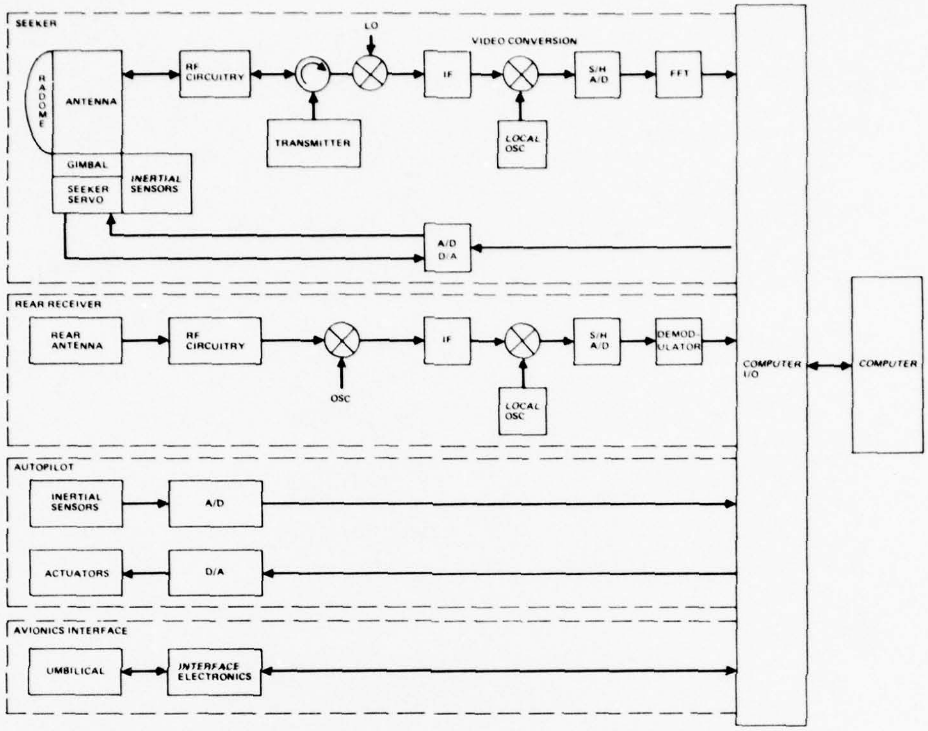

FIGURE 3. CLASS A COMPLEXITY MISSILE UTILIZATION OF A COMPUTER

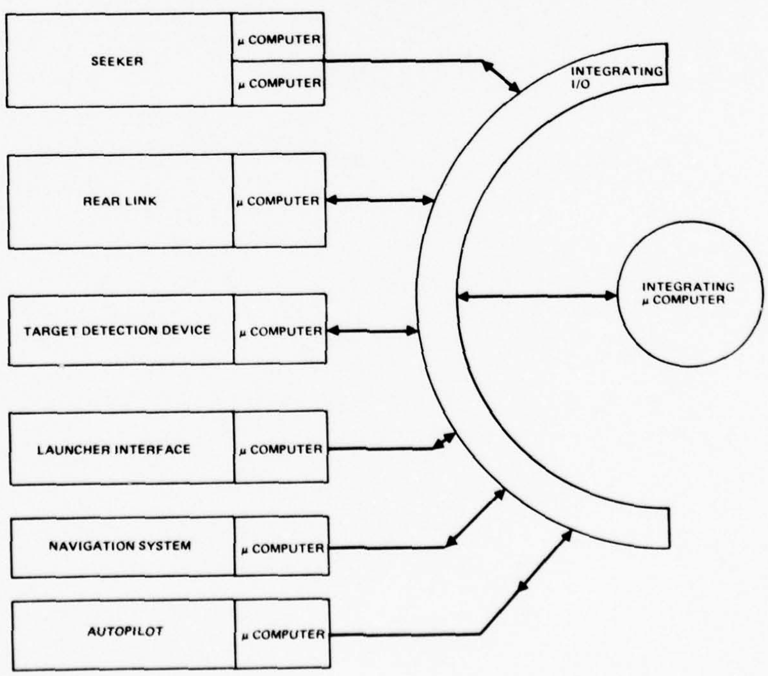

FIGURE 4. FUTURE TACTICAL MISSILE COMPUTER SYSTEM

For example, let us compute θ_T from Eq. (14) for the following typical set of CCRP pickle conditions.

22-1

A RELIABLE AND SURVIVABLE DATA TRANSMISSION SYSTEM FOR AVIONICS PROCESSING

BY

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SUMMARY

This paper describes a reliable and survivable system (RHEA) for interconnecting real-time processing elements. Motivations are given for the choice of the two-level RHEA structure : a distributed "irregular network" and a set of local "star structures". The irregular network has active nodes that carry out automatic signal routing whereas the star structure has a passive central node based on a loosely-coupled pulse transformer. Two types of communication control are presently under analysis : "contention control" and a "decentralized daisy chain" ; the final choice will be based on security and modularity criteria.

INTRODUCTION

The last few years have been marked by a spectacular progression in the complexity of the systems tackled by automatic control and computer science. This is due to the evolution of the missions assigned to Man which reach and often pass the limits of his possibilities. This is particularly so in aeronautics as a consequence of the missions that are assigned to him : transport, interception, ground attack,... This progression has been made possible thanks to the technological revolution which has led to the availability of highly-integrated and modestly-priced processing and memory elements.

When one adds to this the criteria of operating costs and versatility, one is naturally led to the notion of decentralization of the processing related to the different functions to be carried out and the notion of the integrated system.

However, a significant factor slowing down the progression of system automation and integration has been the lack of confidence in distributed computing systems regarding their aptitude to satisfy the operational security criteria : reliability, availability, survivability (invulnerability), maintainability,...

The security objectives of a distributed computing system must be satisfied at two levels :

- resource management level : measurement of the state of the resources, resource allocation in function of the system objectives,
- communication level : realization of an operationaly secure system for transmitting data between the computing elements.

This paper presents a study concerning the latter level.

In distributed real-time computing, the global real-time control process is divided into tasks that may be executed in parallel. Such a decomposition is attractive for several reasons :

- from an application viewpoint ; partitioning the system into tasks is conceptualy simple, this is especialy so since systems are frequently defined on a subsystem by subsystem basis,
- from a technological viewpoint ; if sufficiently simple, tasks may be executed by computing elements realized with a relatively slow technology whereas the overall system throughput may be increased,
- from an operational security viewpoint ; tasks may mutualy survey each other and initiate task reconfiguration upon detection of an anomaly or malfunction.

This work was supported by the Direction des Recherches, Etudes et Techniques, under contract N° 76/274

$$\tan(\phi_I - 20 \text{ deg}) = \begin{cases} 0.0484 \text{ at pickle} \\ 0.0838 \text{ at release} \end{cases} \quad (17)$$

22-2

The tasks resulting from a functional decomposition of the system may be allocated according to two philosophies :

- physical centralization : the tasks are functionaly distributed on a single parallel processing machine [1] [2] ,

- physical distribution : the tasks are functionaly and physicaly distributed on distinct machines [3] .

The first approach leads to the realization of a fault-tolerant multiprocessing machine that communicates with the system sensors and actuators over one or several fast input-output channel(s).

The second approach means that the tasks are executed by a set of physicaly distributed computing elements. These computing elements, together with sets of sensors and actuators, form a set of loosely-coupled "*subscribers*" that must communicate with each other in order to co-ordinate their actions so as to achieve the global system objective.

The study presented in this paper adopts the second approach in order to realize a system that is completely distributed, both functionaly and physicaly.

The RHEA system, which signifies "Réseau Hiérarchisé pour Environnements Agressifs" ("Hierarchical Network for Agressive Environments") is thus aimed at the reliable and survivable interconnection of relatively autonomous subscribers (sensors, computing elements, actuators), in an enlargened set of failure hypotheses which are characteristic of real-time and on-board systems : component failure mechanical damage, electromagnetic perturbations.

A reliable and survivable interconnection system must feature a good operational security at three levels :

- the architectural or structural level,

- the physical implementation level,

- the communication control level.

The first part of the paper thus gives the motivations for the choice of the RHEA structure which consists of two levels :

- a distributed "irregular network",

- a set of local "star structures".

The second part of the paper describes the physical implementation and the operating principles of the RHEA structure.

The third and last part of this paper deals with reliable communication control which is an essential part of any operationaly secure data transmission system.

I. DEFINITION OF THE RHEA STRUCTURE

I.1. The data processing structure

Current distributed control systems are normaly realized according to one of four data processing structures [4] . These structures are shown in figure 1. It should be underlined that these structures are defined according to a data processing definition that results from the way messages are vehiculed between computing elements and does not necessarily infer a physical implementation of that structure. An example of a structure whose physical implementation is fundamentaly different from its data processing definition is OSIRIS [1] [5] . In this system a central fault-tolerant multiprocessor communicates with input-output organs on a master-slave basis, messages are either master to all slaves or slave to master. The data processing definition is thus that of a global bus with a central switch (see figure 1). However, the physical implementation of the OSIRIS system is not at all a bus structure but takes the form of an irregular network.

As described in the introduction, we are interested in realizing a highly secure loosely-coupled system. Consequently, the global bus structure would seem a likely candidate since all the subscribers are connected together homogeneously and there is no central point susceptible to failure other than the "bus" or communication medium.

In reality, the subdivision of a control system into tasks often leads to groups of physicaly localized tasks that communicate more frequently among themselves than to other tasks in the system. For this reason, a hierarchical structure such as the bus window approach would seem interesting.

The RHEA system consists of a two-level hierarchical bus window structure and can be represented schematically as shown in figure 2.

4.4 Incorporation Into An Operational Weapon Delivery System

The Litton LW 33B air-to-ground weapon delivery system includes the wings nonlevel pullup capability described herein. First operational deployment of these systems is on the McDonnell Douglas F4E's being delivered to Turkey and to Greece.

This figure merits two comments :

- the bus window or inter-level interfaces may be considered as a simple subscriber when viewed from either level, 12-3
- the information is transfered through blocks purposely labeled as communication media since the latter may be realized in several different physical implementations (bus, network, star,...)

I.2. The communication media

The two different levels of communication media defined in paragraph I.1. may be studied as independant global "bus" structures for which the physical implementation must be chosen in order to obtain a high operational security.

With this aim in mind, we have carried out an analysis of several hypothetical physical structures realizing the global bus data processing function. These structures are defined and commented in the table of figure 3.

A complete comparative study of these structures must take account of diverse criteria related to operational security, operating costs and versatility.

The initial study that we have carried out is a qualitative analysis involving mainly two components of the operational security : reliability and survivability.

This will later be completed by a quantitative study including other criteria such as maintainability, cost, versatility, etc...This quantitative study will be carried out according to two hypotheses :

- an unmaintainable environment : repair during the mission is impossible and successive failures must be envisaged ; this study makes use of both the MARKOV and the minimal path approaches [12] [13] ,
- a maintainable environment : repair during the mission is possible and the corresponding reliability models may be simplified by assuming that no further failure occurs during the repair periods [14].

The qualitative analysis enables the choice of one of the six structures defined in figure 3 in function of the environment and the geographical distribution of the subscribers.

I.2.1. Subscribers geographicaly distributed in a non-hostile environment

When the subscribers are geographicaly distributed, the star structure may be eliminated due to the high implied wiring costs.

Also, a non-hostile environment means a very low probability of multiple localized failure : we need only account for single random failures. Consequently, all those structures intended to provide multiple paths (network structures) are needlessly complicated.

Thus, the structure most capable of tolerating random single failures is the protected bus structure, the node bus brings unnecessary complication only in order to tolerate bus section short circuits.

I.2.2. Subscribers geographicaly distributed in a hostile environment

The star structure may be eliminated for the same reasons as above.

A hostile environment means that we must account for multiple localized failures due to physical damage (accident or agression) or electrical perturbation.

Consequently, all the bus structures must be eliminated since the busses must all meet in certain places such as the actual connections to the subscribers.

The only structure permitting multiple local failures without total disruption of communication is the network structure. Due to the constraints imposed by the geographical location of the subscribers the topology of the network is usually irregular.

However, when active nodes are considered the problem of supplying them with electrical power is posed. It is often supposed that the subscribers are independently powered and the choice of an irregular network is justified by the fact that the nodes may be powered from the same source as corresponding subscriber.

azimuth location of the pipper prior to pickle. By rolling the aircraft one way or the other he can move the pipper back and forth across the sight.

In reality, especialy in on-board system, power is distributed to subscribers from a central point [15] and it is no use providing a highly-secure damage-proof data communication structure if the power distribution system does not present the same qualities :

- if power is distributed by a cable harness then the equivalent data communication structure is the <u>star structure</u>,
- if power is distributed by bus-bars then the equivalent data communication structure is a <u>bus structure</u>.

Consequently, if subscriber electrical power is not idependently provided the irregular data communication network may only be justified if it is accompanied by an irregular network for power distribution.

I.2.3. <u>Geographicaly localized subscribers</u>

In the case of geographicaly localized subscribers, the fact as to whether or not the environment be hostile is irrelevant since an external perturbation could affect all of the system as easily as a part of it.

Also, <u>the star structure</u> no longer presents any wiring problems and it is in fact this structure that seems the best if the central node can be designed in such a way that no single failure leads to total loss of communication.

1.3. <u>Final choice</u>

The RHEA system is intended to operate in an environment where physical damage is a possibility and as a result of the above analysis the physical structures chosen for the two levels are :

- Level 1 (geographicaly distributed) : Irregular network,
- Level 2 (geographicaly localized) : Star structure.

An example topology is given in figure 4.

II. <u>DESCRIPTION OF THE RHEA STRUCTURE</u>

II.1. <u>Level 1 : an irregular network</u>

The heart of the RHEA system is the geographicaly distributed irregular network providing fault and damage tolerance.

In such a network, the nodes must be active in order to prevent pulse dispersion due to multiple signal paths and, in non fiber-optic systems, to isolate link short-circuit failures. As a consequence of the former point, each node must choose one port at a time as an input port. One way of achieving this is to control the directivity of the node by a control algorithm [5] .

The method used in RHEA is to carry out signal-routing automatically by "electing" an input port by arbitration of the received signals and retransmitting the elected input out of the remaining ports.

II.1.1. <u>Operating principle of irregular network node</u>

The block diagram and the timing diagram of a self-routing node is given in figure 5.

The output signal X of the detector circuit is such that it rises just after reception of the <u>first</u> received signal and falls a time δ_G after the end of the <u>last</u> received signal. The rising edge clocks a pair of latches whose outputs G_0 G_1 select the appropriate "elected" input of the multiplexer. In the event of simultaneous arrivals at the node, the encoder circuit chooses one of the inputs on a priority basis, an independent choice is unnecessary because the inputs signals are copies of the same signal (we assume in this section that only one subscriber at a time may transmit).

The guard time δ_G ensures that signals to not propagate unendingly around the network and the delay Δ in the multiplexer input paths avoids shortening of the first pulse due to the input selection delay.

To give an idea of the simplicity of the node, a functional circuit without autotest has been realized using eleven ordinary SSI-MSI TTL circuits.

II.1.2. <u>Connectivity and modularity</u>

As shown in figure 5, the nodes have been provided with four I/O ports. This figure was chosen so that a subscriber (connected to one of the four ports) may have three connections to the network. The network so obtained may thus be three-connected resulting in a sufficient connection reliability for most applications [16] .

However, if a higher connectivity is required, the nodes are completely modular ; two four-port nodes may be connected together to form a single six-port node, three four-port nodes form an eight-port node, etc.

II.1.3. Node operational security

Tolerance of faults and damage is inherent to a network structure but the basic node concept given above is insufficient for three reasons :

- failure of a node leads to the loss of a subscriber : this may not be acceptable for the more critical subscribers,
- system reconfiguration will be more efficient if the total system status may be observed at any instant,
- the nodes must not feature failure modes that are catastrophic to overall network communication.

Two levels of operational security thus appear :

- knowledge of the state of the network which can be achieved with self-checking nodes,
- fault-tolerant nodes for connection of critical subscribers (which could themselves be fault-tolerant).

The second level is obtained simply from the first by virtue of the connectivity modularity of the nodes : associating two self-checking nodes (with self-disconnection in case of failure) leads to a fault-tolerant node with one degree of redundancy.

This solution, which enables a graduation of fault-tolerance via identical elements, was thus prefered to the TMR solution which has also been considered.

The chosen fault-detection method is a duplication and comparison structure. Two elementary nodes as defined in figure 5 are connected in parallel and their outputs are compared with a self-checking comparator using morphic boolean logic [17] [18] [19] [20] (figure 6).

The output signal of the self-checking comparator is processed in order to filter out parasitic errors and upon detection of an out-of-code signal, power to the node is removed.

An original aspect of the self-checking comparator is the method used for ensuring error-detection in the comparator.

The problem is due to the fact that the four outputs $S_i, i \in [0,3]$ of N_1, which are compared with the complemented output $\overline{W}$ of N_2, are in fact four copies of the same signal. Consequently, if two pairs $\{W\ S_i\}$ $\{W\ S_j\}$ are compared by a 1-out-of-2 comparator, the input signals only describe one-half of the comparator input space ($\{01\}$ $\{01\}$ and $\{10\}$ $\{10\}$). The outputs $\{f,g\}$ thus remain fixed at 01 and the comparator is no longer self-checking.

The solution to this problem is to provide another pair of inputs to the comparator which is obtained by a simple division by two of the node output signals. The first comparator thus receives signals that describe all of its input space ($\{01\}\{01\}$; $\{10\}\{01\}$; $\{01\}\{10\}$; $\{10\}\{10\}$) and the effect is cascaded to the other comparators (figure 7).

The outputs of the comparator stages are no longer constant, their complete input space is described and self-checking is thus maintained.

II.1.4. Signal path

When these self-routing nodes are connected together to form a network, signals are propagated from a transmitting subscriber along a tree-like path to every other subscriber in the network (figure 8).

In such a network, signals arriving on the input ports of a particular node may originate from either a subscriber or a neighboring node. Consequently, the maximum delay between the first and last received signals is equal to $2T_p$ where T_p is the maximum node plus link propagation delay. This is due to the fact that the worst case delay corresponds to the "echo" of the "elected" input signal from a neighboring node.

II.1.5. Power distribution

As indicated in paragraph I.2.2., the damage-proof irregular network structure may only be justified if the power distribution system also possesses the same multiple-path properties. Current research is aimed at the solution to this problem with the development of a power distribution node that disconnects short-circuited links.

II.2. Level 2 : a star structure

As mentioned in paragraph I.2.3., the star structure is interesting for interconnecting geographicaly localized subscribers if the central node does not constitute a hard-core.

The realization that we are envisaging is shown in figure 9. It is based on the use of a loosely-coupled pulse transformer.

The aim of the loosely-coupled transformer is to provide a central node that features no failure mode leading to total communication black-out. The transformer is air-cored and due to the low coupling, the short circuit of a winding does not affect communication between the other windings. The star structure so realized may be considered as a protected bus structure where the bus is concentrated at a single point.

The short-circuit independence has been verified both theoretically and in practice. Figure 10 shows the pulse response between two windings of a three-winding loosely-coupled transformer when the third winding is respectively normaly loaded and short-circuited.

In practice, very low coupling coefficients may be obtained and the effect of short-circuits is imperceptible. Of course, the signal attenuation is inversely proportional to the coupling coefficient and the output signal must be amplified and reconstituted. The circuits necessary for this may be powered from the same source as the subscriber and as a consequence, any winding or interface failure may be assimilated to a subscriber failure ; there is thus no single-point failure leading to total communication disruption.

REMARK : The star structure may also be realized using fiber-optic technology using mirror coupling 21 (figure 11).

Such a realization is interesting from the fiber-optic viewpoint since all subscriber-subscriber paths feature the same optical attenuation. However, stuck-at-one failures of the subscriber light emitters are not tolerated as in the loosely-coupled transformer.

III. COMMUNICATION CONTROL

In order to avoid any centralization in our global bus data processing system, it is logical that communication control should also be completely distributed. If the full benefits of decentralization are to be obtained much attention must be paid to the recoverability and the modularity of the control method.

The two most likely candidates for an operationaly secure and modular communication control are :

- contention control [6] - [8] (figure 12),
- decentralized daisy chain [22] (figure 13).

III.1. Communication control on the irregular network

III.1.1. Effect of conflict on the network

Consider the event of two subscribers starting transmissions at times 0 and $0+\delta$ respectively. Figure 14 gives an example of such a situation ; the times marked on the node ports are the times at which signals are received where T is the node plus link propagation time (supposed constant), the dotted port corresponds to the elected input.

Since the nodes are locked onto the elected input from the beginning of the first signal received to the end of the last signal received, all those subscribers to the left of the dotted line receive the message emanating from node 1 and those to the right from node 10. Neither subscriber is conscious that conflict is occuring. Thus, for both of the envisaged communication control systems, remote conflict detection is a necessity.

III.1.2. Contention control

In the contention control system, conflicts must be detected in order to initiate message retransmission. There are two mays to detect conflicts.

Firstly, an acknowledge message may be sent by the destination subscriber. Lack of receipt of an acknowledgement message would mean that the transmission has been perturbed and the message must be retransmitted. This method for contention detection has two disadvantages :

- conflicts are detected only after the end of a message,

- an acknowledgement message means that a broadcast mode is not feasible.

The second way to detect conflicts implies a modification to the nodes in order to detect the fact that different messages are being received (nodes 2, 6, 7 and 8 in figure 14). This may be achieved either by decoding the messages or by simply comparing the inputs of each node with their outputs. The first method leads to considerable complication of the node and the second implies restrictions as to the maximum delay permitted for an inter-node link. A node that detects contention sends a pulse on its ouputs in order that neighboring nodes may also detect the contention. The conflict signal thus "bubbles" out to the sources of the conflicting transmissions.

III.1.3. Daisy chain control

In a daisy chain control system, conflicts must be detected in order to detect disynchronization and to initiate an initialization procedure.

With the irregular network, one way of achieving conflict detection is to include the contents of the counter as a word in each message transmitted (including the synchronization message). If a subscriber detects a difference between its counter and the corresponding word in a received message, it takes the responsability of re-initializing the system. This may be done by sending a signal at a different frequency which the nodes detect as a priority signal that interrupts the elected input and is retransmitted over the ouput ports.

III.2. Communication control on the star structure

In the star structure, conflicts may be detected by a listen-while-talk device that delivers an interrupt to the subscriber if the received and transmitted signals do not concord.

Thus, either of the two envisaged communication control methods is equally applicable.

III.3. Choice of a control method

The choice between contention control and a decentralized daisy chain will be made after a study weighing the advantages and disadvantages of both methods. This study is presently being carried out with the aid of a discrete time simulation using the IBM GPSS simulation program. This simulation will highlight the properties of the control methods concerning :

- their aptitude for real-time control (transmission blocking times),

- their behavior in the presence of subscriber failures (the re-synchronization problem),

- their possibilities for carrying out system reconfiguration (redistribution of system tasks).

CONCLUSION

In this paper, we have given the reasons for the choice of a two-level hierarchical structure for communication between relatively autonomous subscribers carrying out real-time control in an environment where physical damage is a possibility.

The structure chosen for the lower level is a star structure that in our opinion provides the highest security for communication between subscribers in a geographicaly localized cluster.

Geographicaly seperated subscribers or clusters may communicate via an irregular network that tolerates physical damage. The nodes of this network are very simple and carry out signal routing automaticaly. A network structure is also very suitable for fiber-optic technology since all links are point-to-point with no power division.

The very principle of input port election together with node self-checkability leads to a high modularity on both the functional level (connectivity) and the operational security level (fault tolerance) by simple paralleling of the nodes.

The choice of a communication control method and the validation of the system are being carried out at several levels :

- by a discrete-time simulation,

- by the construction of a prototype system,

- by the application of the system to a typical avionic integrated data processing structure carrying out the navigation and guidance functions [23].

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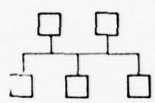
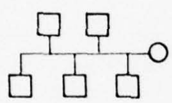
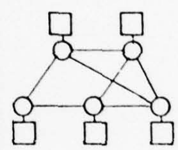
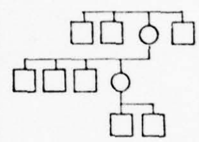
TYPE	GLOBAL BUS	BUS WITH CONTROL SWITCH	NETWORK	BUS WINDOW
FORM				
REMARKS	Complete subscriber homogeneity	The central "switch" may be a master computer, a control room or a mailbox memory	Communication is achieved by message or packet switching	Communication on each bus may be independent or to another hierarchy via the bus "window"
EXAMPLES	ETHERNET [6], DP/M [7], COBUS [8]	OSIRIS [1], SIGMA [9], OAIS [10]	Kent University Network [11]	OSIRIS [1] and DP/M [7] have certain characteristics corresponding to this structure

□ : subscriber

○ : switch or message routing element

FIGURE 1. Distributed control data processing structures.

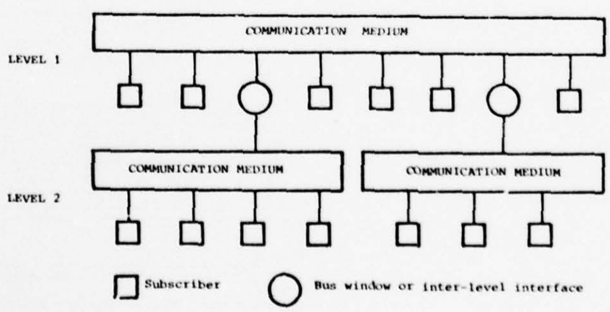


Figure 2. The two-level data processing stucture

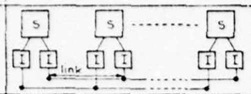
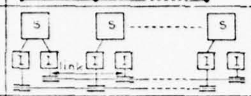
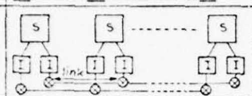
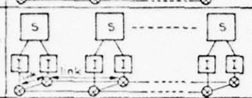
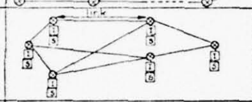
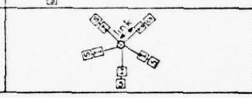
STRUCTURE	ARCHITECTURE	COMMENTS	EFFECT OF RANDOM FAILURES				TOLERATES DAMAGE	APPLICABLE TO DISPERSED SYSTEMS
			STUCK INTERFACE	NODE FAILURE	LINK SHORT CIRCUIT #	LINK OPEN CIRCUIT		
UNPROTECTED BUS		Classical redundant unprotected bus structure	Lose bus	x	Lose bus	Cuts bus in two	NO	YES
PROTECTED BUS		Stuck-at interface failures are isolated from bus	Tolerated	x	Lose bus	Cuts bus in two	NO	YES
NODE BUS		Link short-circuits not propagated	Tolerated	Cuts bus in two	Cuts bus in two		NO	YES
CROSS-STRAPPED BUS		Supplementary paths to tolerate multiple link failures	Tolerated	Extra links tolerate certain multiple node failures	Extra links tolerate certain multiple link failures		NO	YES
IRREGULAR NETWORK		Arbitrary topology. Links are routed seperately	Tolerated	Lose subscriber if latter connected to only one node	Inherently tolerated		YES	YES
STAR		Suitable only for geographicaly localized subscribers	Tolerated	Catastrophic	Lose subscriber		NO	NO

FIGURE 3. Physical implementations of a global communication medium

S : subscriber, I : interface, ═ : passive coupler, ⊗ : isolating node,

x : does not apply to fiber-optic systems.

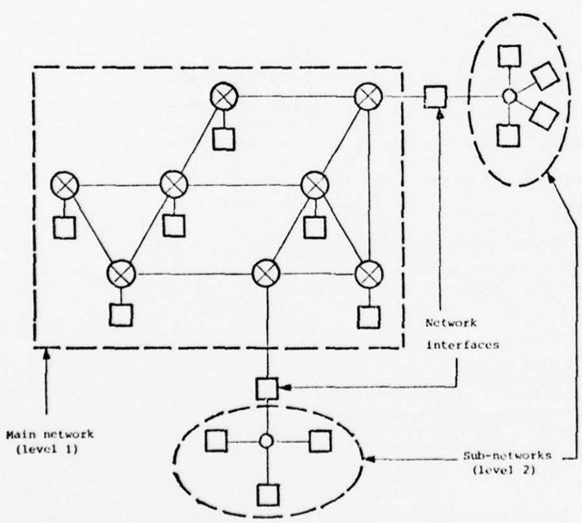


FIGURE 4. Example RHEA topology

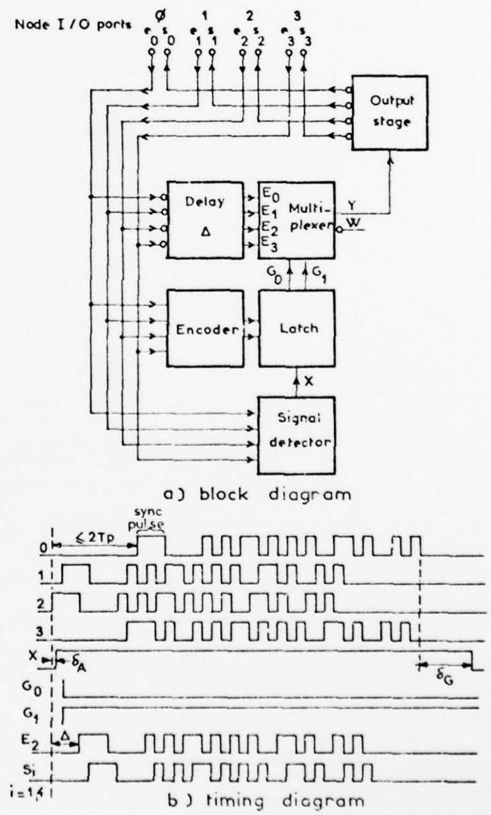
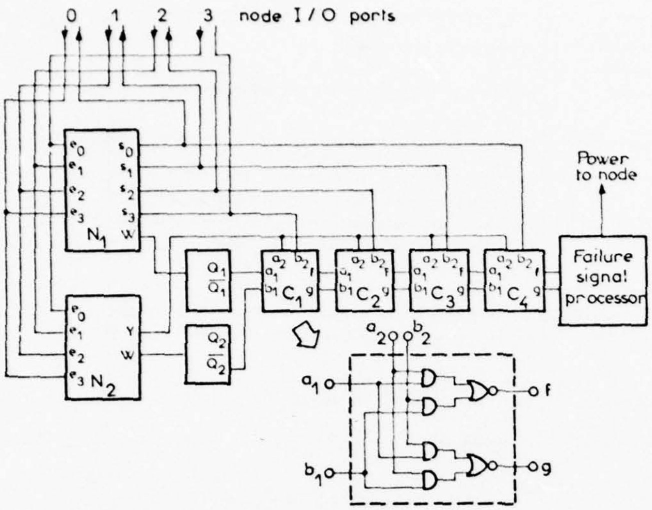


Figure 5. Node operating principle

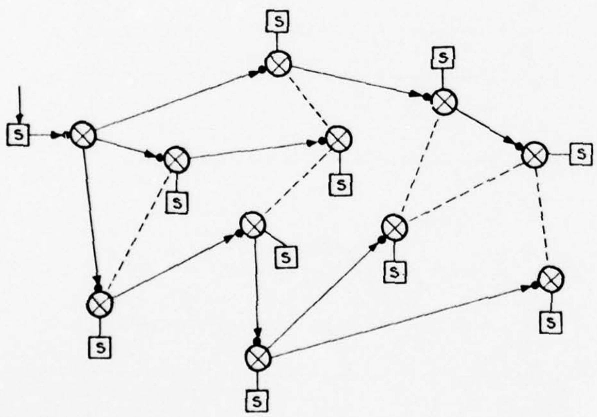


N_i:basic nodes(figure 5) C_i:1-out-of-2 Carter comparator(see insert)

Figure 6 .Self-testing node block diagram

S_i Y	Q_1 $\bar{Q}_2$	f_1 g_1	f_2 g_2	f_3 g_3	f_4 g_4
01	01	10	01	10	01
10	01	01	01	01	01
01	10	01	10	01	10
10	10	10	10	10	10

Figure 7. Evolution of comparator outputs



S Subscriber ⊗ Node —— Active link ---- Idle link

Figure 8. Signal path in irregular network

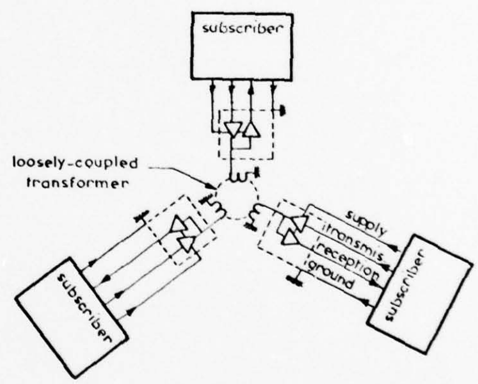


Figure 9 The star structure(configuration shown for 3 subscribers)

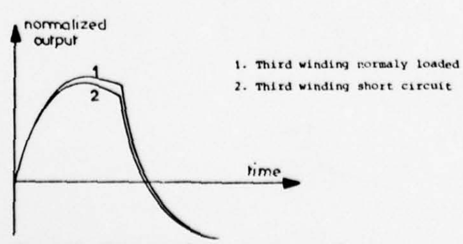


Figure 10. Theoretical pulse response for a coupling coefficient k=0.1

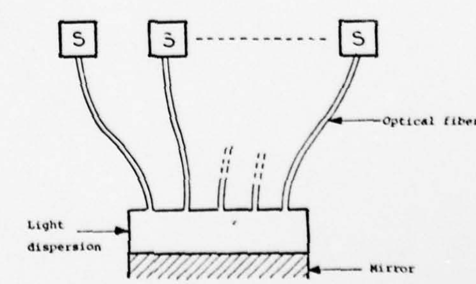


FIGURE 11. Star structure using optical fiber.

PRINCIPLE	COMMENTS
(1) No effort is made to avoid communication conflicts	Communication is stochastic
(2) Conflicts are detected and perturbed messages are retransmitted	Transmission delays are random and may be unbounded
ALGORITHM	**COMMENTS**
(1) A subscriber may only attempt a transmission when it detects a silent communication medium	The probability of conflict is equal to the probability that two or more subscribers attempt to start a transmission in a time interval equal to the communication propagation time
(2) When contention is detected, a subscriber retransmits its message after a randomly selected delay interval	Retransmission intervals must be randomly selected so as to avoid regenerative bursts of retransmissions
(3) A watch-dog in each subscriber times-out excessively long messages	Prevents "hogging" by failed subscribers

FIGURE 12

PRINCIPLE	COMMENTS
(1) Each subscriber is allocated one or more positions in a cyclic transmission frame (2) The transmission frame and the allocated positions are represented in each subscriber by a ROM register which is programmed with ones in the corresponding transmission positions	The order in which subscribers transmit is pre-determined
(3) This ROM register is adressed by a counter that is incremented each time that an end-of-message is detected	Transmission delays are variable but bounded
ALGORITHM	**COMMENTS**
(1) On detection of an end-of-message, each subscriber increments its counter (2) If a subscriber finds that it may transmit (a one in the corresponding ROM position) it sends its message if it has one ready (3) If it has no message ready, it sends a short synchronization message	A spurious counter value may lead to a communication conflict
(4) A watch-dog in each subscriber times-out excessively long messages	Prevents "hogging" by failed subscribers
(5) A time-out device in each subscriber increments the counter if no start-of-message is heard in a certain time interval after the last end-of-message	Protects against failed subscribers and enables possible system extension

FIGURE 13

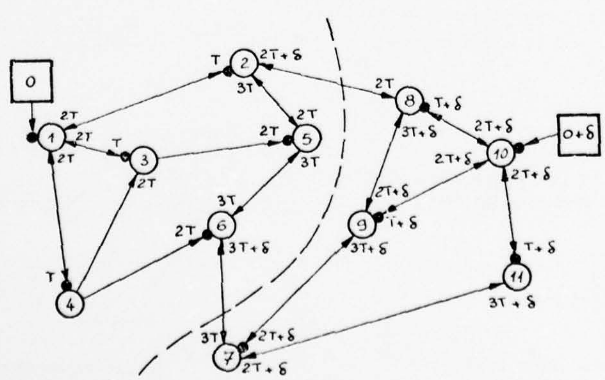


Figure 14. Conflict on the irregular network

DYNAMIC SIMULATION OF A MULTI-SENSOR COMMUNICATION AND NAVIGATION SYSTEM

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SUMMARY:

COMMANDS is a COMMunication And Navigation Dynamic Simulator. This simulator is being used to develop and validate the avionic software contained in Singer-Kearfott's Class II Joint Tactical Information Distribution System (JTIDS) terminals.

Multi-Sensor Communication and Navigation System avionic software is comprised of a real-time airborne operating system (RTAOS), a communication subsystem based on the time division muxiplexed access (TDMA) method, a tactical navigation (TACAN) subsystem, and a relative navigation (REL NAV) subsystem.

Since this avionic software was written in assembly language there existed a very stringent requirement for a dynamic simulator for both development and validation of the avionic software. COMMANDS was developed to meet this requirement for an economical tool that would support the test requirements of both the communication (TDMA) subsystem's high data rates, and the complex computational requirement of the Relative Navigation System. The COMMANDS simulator is resident on a mini-computer, and is physically connected to the JTIDS Class II Operational Flight Program terminal through the operational input/output I/O devices. This allows the simulation of the avionic box which is receiving the same inputs in the laboratory test as it will under flight conditions.

A trajectory is a prescribed sequence of input data which will cause the avionic software to perform the desired function and issue specific responses. Validation of the airborne software consists of exercising a set of trajectories we designed to completely encompass the operational functions deemed testable.

The limitations of the mini-computer are removed by dividing trajectories into a baseline portion, and a real-time closed loop portion. The baseline portion is that unaffected by the response of the airborne software. The baseline trajectory is generated off-line and input to the COMMANDS via magnetic tape. The real-time closed loop portion is dependent on data from the OFP airborne computer. Data transmission acknowledge, built-in test (BIT) wraparound, range error modules, and status processing are examples of items in this class. With the division of trajectory generation responsibility, the mini-computer can readily process the dynamic simulator requirements of COMMANDS.

Most of the data output from the avionic system is accepted by COMMANDS. Some data is processed immediately, but all data is recorded in real time for future data reduction. Data correlation is performed wherein each item from the output tape is compared parameter by parameter to the corresponding item in the "true-world" tape.

This permits a trajectory to be exercised which produces many thousands of data points. These data can subsequently be compared to acceptable limits, and only data exceeding these limits will be typed for analytical investigation. Additionally, the COMMANDS has tested the complex I/O of this terminal since all trajectories transmit and receive data over the operational devices.

The automatic features of this development and validation tool further enhance software reliability. Validation depends on automatic acceptance of data and identification for analysis of all data exceeding the established error bounds. Pre-defined trajectories are certified by the requirement personnel to foster complete testing and assure that trajectories are developed which encompass all functional requirements.

Since the actual I/O is exercised during these tests, software performance in field tests will be the same as in the dynamic simulator. Finally, this method is inexpensive to generate, exercise, and maintain. The combined effort yields a very powerful, real-time dynamic simulator for communication navigation systems.

LIST OF SYMBOLS:

AGC	Automatic Gain Control for a TACAN System
A/J	Anti-jam communication technique
BIT	Built-in Test Software, used to test flight system
C&C	Command and Control
DC	TACAN System Digital Control
DME	Distance Measuring Equipment
D/R	Dead Reckoner Navigator
DU	Display Unit
HELOS	Helicopters
HP21MX	Hewlett-Packard Mini-Computer
I/O	Input or Output over a communication line
MCU	Mode Control Unit
OFP	Operational Flight Program
REL NAV	Relative Navigation, a subsystem for local grid navigation
RF	Radio Frequency Message Processor
RPV	Remotely Piloted Vehicles
RTAOS	Real Time Airborne Operating System
RTE-IIB	Hewlett-Packard Real Time Executive, Version IIB
SAM	Surface-to-Air Missile
TACAN	Tactical Navigation System
TDMA	Time Division Multiplexed Access, a method of communication
TOA	Time of Arrival of communication signal

INTRODUCTION:

COMMANDS is a Communication and Navigation Dynamic Simulator. This simulator has been used to develop and validate the Avionic Software contained in Singer-Kearfott's Class II Joint Tactical Information Distribution System (JTIDS) terminals.

Multi-Sensor Communication and Navigation Avionic Software is comprised of four subsystems. A Real-Time Airborne Operating System (RTAOS), a communication subsystem based on the Time Division Muxiplexed Access (TDMA) method, a TACtical Navigation (TACAN) subsystem, and a Relative Navigation (REL NAV) subsystem. The flight software executes on an SKC3120 which is a 16-bit fractional computer capable of supporting 64K words of instruction/data memory.

Since this avionic software was written in Assembly Language there existed a very stringent requirement for a dynamic simulator for both development and validation of the avionic software. COMMANDS was developed to meet this requirement as an economical tool that would support the test requirements of both the communication (TDMA) subsystem's high data rates, and the complex computational requirement of the Relative Navigation Subsystem.

The COMMANDS simulator is resident on a mini-computer and is physically connected to the JTIDS Class II Operational Flight Program (OFP) terminal software through the operational Input/Output (I/O) devices. This allows the simulation of the avionic box which is receiving the same inputs in the laboratory test as it will under flight conditions. The decision to host COMMANDS on a mini-computer has both advantages and disadvantages. The principal advantage is that the simulator is connected to the software via the operational flight hardware. Effects contributed by delays in transmission of data through the hardware, noise errors in I/O devices, Airborne Computer roundoff and truncation errors are all simultaneously tested within this simulation approach. The sole disadvantage of hosting the simulator on a mini-computer is that the mini-computer is not capable of performing the required computations in real time.

A trajectory is a prescribed sequence of input data which will cause the avionic software to perform the desired function and issue specific responses. Validation of the airborne software consists of exercising a set of trajectories we designed to completely encompass the operational functions deemed testable. Some software is explicitly tested by a trajectory; e.g., the Relay Message function of the Communication subsystem is tested by inputting Relay Messages. Other functions are tested implicitly by this dynamic simulator; e.g., the matrix multiply routine must be correct if the REL NAV trajectories are to be successfully executed, even though the matrix multiply itself is never explicitly tested.

The limitations of the mini-computer are removed by dividing trajectories into a baseline portion, and a real-time closed loop portion. The baseline portion of the trajectory is that portion unaffected by the response of the airborne software. The baseline trajectory is generated off-line and input to the COMMANDS via a 1600bpi 9-track magnetic tape. The real-time closed loop portion consists of the functional capability that is dependent on data from the OFP Airborne Computer. Data transmission acknowledge, built-in test (BIT) wraparound, range error modules, and status processing are examples of items in this class. With the division of trajectory generation responsibility, the mini-computer can readily process the dynamic simulator requirements of COMMANDS.

In addition to off-line baseline trajectory generation, a "true-world" magnetic tape is created. The "true-world" tape is a frame-by-frame prediction of what the correct airborne flight software should generate. The "true-world" tape is the standard to which the output of the OFP software is compared.

Every item of data output from the avionic system is accepted by COMMANDS. Some data is processed immediately, but all data is recorded in real-time for future data reduction. This data reduction consists of two phases. First, the content of messages is printed, status data displayed, and relative navigation information presented in engineering units. Output can be a printout, or a plot of the data. DATA Correlation is the second phase wherein each item from the output tape is compared parameter by parameter to the corresponding item in the "true-world" tape. Associated with each parameter is an expected error range. Parameters which exceed the allowed error range are identified by parameter, frame, and error value. Output of the data correlation phase of this simulator can be a printout, or a plot of the errors discovered. This permits a trajectory to be exercised which produces many thousands of data points while all are thoroughly examined and compared to acceptable limits. Only data exceeding these limits will be typed for analytical investigation.

COMMANDS has been used to develop and validate Multi-Sensor software at Singer-Kearfott which is currently in Advanced Development and Prototype testing. The separation of trajectories into baseline, and real-time functions was previously developed but was not fully applied to communication systems until this project.

Additionally, the COMMANDS has tested the complex I/O of this terminal since all trajectories transmit and receive data over the operational devices.

The COMMANDS simulator will be expanded concurrently with refinements in the functions of the operational flight software. This tool will also be expanded to have additional quasi-real-time data reduction features, that is, expanded ability to display reduced data during quiescent portions of a trajectory.

The automatic features of this development and validation tool further enhance software reliability. Validation depends on automatic acceptance of data and identification for analysis of all data exceeding the established error bounds. Pre-defined trajectories are certified by the requirement personnel to foster complete testing and assure that trajectories are developed which encompass all functional requirements.

Since the actual I/O is exercised during these tests, software performance in field tests will be the same as in the dynamic simulator. Finally, this method is inexpensive to generate, exercise, and maintain. The combined effort yields a very powerful, low cost, real time dynamic simulator for Communication and Navigation systems.

OVERVIEW
OF
COMMANDS

MAIN COMPUTER FACILITY	MINI-COMPUTER	SPECIAL PURPOSE AIRBORNE COMPUTER
DOES OFFLINE GENERATION OF INPUT TRAJECTORY DATA	CLOSED LOOP PORTION	
		CONTAINS FLIGHT SOFTWARE
DOES OFFLINE DATA REDUCTION AND CORRELATION	INTERFACES I/O MODELS	

RELATIVE NAVIGATION SUBSYSTEM

The Relative Navigation (REL NAV) subsystem in the Multi-Sensor Operational Flight Program has been designed to provide the capability for each member of a cooperative community to derive the benefits afforded by a sensor at any other member's location. This is accomplished through the use of a dual-grid navigation technique. In addition to navigation in the conventional geodetic frame, a relative grid is established. The key element in the dual grid concept is the Time Division Multiple Access (TDMA) ranging and communication system. Its member-to-member distance measuring capability provides the precise ranging information necessary to lock each member in a common yet arbitrary relative navigation grid, while its communication capability provides the means for dissemination of sensor data.

The Relative Navigation (REL NAV) capability affords members of the same community the ability to acquire targets and share the target sensor data for all grid members, effect rendezvous, exchange position data and deliver weapons effectively and accurately. Accurate relative navigation within a community implies that there exists azimuthal as well as position and time correlation among the community members. That is, any member possesses the navigation accuracy to touch any other member with an imaginary stick, or perhaps more significantly to touch any arbitrary point in the tactical region that another member is touching. Each member positions himself in the relative grid as well as aligning the axes of the relative grid to a common azimuth.

No community member represents an independent entity, but rather is an integral part of a network of users linked together by the precise RF ranging system. Relative position is known so well that this network of members can be thought of as one hybrid navigation/weapon delivery system with distributed sensors. The single relative grid provides a measurement base in which the precise exchange of target sensor data occurs (e.g., radar target data). For combat considerations this means that target positions as well as navigation sensor data and other points of interest can be exchanged among members of the community without significant loss of precision. This same relative grid which links all cooperative members allows for the distribution of precise geographic position data (e.g., GPS data) as well as the mutual augmentation of different distributed geographic navigation sensors, just as if they were located on the same vehicle (e.g., a doppler on one vehicle being utilized with an inertial system in a second vehicle to provide a hybrid doppler inertial capability). Members carrying different sensors and sensor types, with different error signatures, augment each other to provide a hybrid system which is more accurate than any single system element alone.

The software mechanization for the dual grid navigation consists of a Kalman filter which models those independent states necessary to develop two complete sets of navigation parameters; a relative set and a geographic set. Of utmost importance here is the fact that the pilot does not become involved with any of this data. The seemingly complex decision as to which grid to use for any given problem is transparent to the pilot. He is not required to make real-time selections of position or any other data. The system is programmed to address the selection of which grid information should be utilized. The relative navigation coordinates which are available as filter outputs are utilized for relatively derived target data, whereas the geographic position outputs are used for geographically designated targets.

The concept of having to select from two different sources of position is not new to the weapon delivery software designer. Most tactical aircraft are equipped with both a radar altimeter and a baro-altimeter, one providing relative (ground referenced) altitude, the other providing a measurement of absolute (sea level referenced) altitude. The use of two altimeters with the associated question of which to use at any given time is well understood and the application-related solution taken for granted. Use of the dual grid data is no more complex than this. In a direct air-to-ground attack of a target acquired by the aircraft's own radar, the software establishes its own local relative grid for navigation to determine when to release the weapon. Since it is a relative problem use of geographic position directly is meaningless. In addition, any updates to the geodetic system received during that time would be ignored insofar as the position computation is concerned. Over the short period of time (approximately 5 to 10 seconds) from target designation to weapon release, the integral of dead-reckoner velocity, with its inherent short term stability, represents the best estimate of position change.

This single aircraft example is analogous to the community problem where the target is acquired using the radar on one member's aircraft, and this target must subsequently be attacked by another member aircraft. Using the relative grid coordinates, the first aircraft can accurately transmit the location of the target to the second aircraft, just as though the radar had been mounted on that second aircraft. The second aircraft can then attack the target continuing to navigate using the same common relative grid. Any geodetic navigation position observations processed during this time will not impact the relative navigation solution. This is accomplished by the dual grid capability afforded by the terminal where navigation is essentially independent between the two coordinate frames or grids. Updates to the geographic system will correct the geographic position, but will not cause short term decorrelation of position data in the relative grid frame.

A second example, again beginning with the conventional single aircraft case, can be utilized to demonstrate when position data from the geographic frame would yield better results. A target is designated by a forward observer using geographic map coordinates. An attack on such a geographically designated target should be made based on the aircraft's best available geographic position. All geographic position updates available prior to the attack would be utilized to improve his estimate of geographic position and compute a more accurate weapon release point.

Similarly, if this forward observer information is received by a community member equipped with the dual grid navigation capability of the communication (TDMA) terminal, that member aircraft would also use his best estimate of geographic position. By utilizing data from the geographic grid, he automatically receives the community's best estimate of geographic position, which has been derived using essentially all geographic update information available to each of the community members.

APPLICATIONS

The applications of the TDMA system take advantage of several distinct features which are integral to the communication terminal. Certainly the relative navigation capability, especially with the dual grid navigation just described, is extremely versatile. This capability becomes even more powerful when utilized with the data link of the TDMA system and the inherent time synchronization of all participating community members. The following examples illustrate the use of several of these features.

MISSILE GUIDANCE AND CONTROL

The utilization of a TDMA terminal for missile guidance and control is particularly efficient. There is a significant savings which results from the fact that the basic TDMA link can provide the time of arrival (TOA) signals, which will allow for computation of the target missile (SAM) relative position as well as being used as the link for guidance commands.

A typical scenario is depicted in Figure 1 where the target missile is detected by airborne and seaborne elements which are operating in the grid: 1) The target position is fixed in the grid based on the detection elements, A, B, and C. 2) A TDMA command and control (C&C) member (C) is assigned. 3) Guidance plane information may be developed based on the selected launcher position and an extrapolated target position. 4) The SAM launcher vehicle (B) is selected. 5) Command and guidance data may be transmitted to the SAM from the C&C ship. 6) The SAM will respond to these commands signals and provide TOA signals to the airborne elements. 7) The TOA signals then may be retransmitted to the C&C ship. 8) Upon reception of the SAM TOA signals, the C&C ship develops the SAM position in the TDMA grid.

Based on the position in the grid, new guidance vector commands may be computed and transmitted over the same data link for SAM guidance. If the SAM possesses a terminal sensor, then TDMA guidance need only continue to place the SAM within the acquisition "basket" of the terminal sensor.

Other SAM scenerios are somewhat similar and, in fact, may operate with a passive system aboard the SAM. Then the SAM position may be established by tracking by TDMA community members, as was the target for the scenario presented.

TARGET DETECTION

1) RADAR ILLUMINATION & DETECTION

2) TRANSMIT TARGET MEASUREMENT IN TDMA GRID TO COMMAND AND CONTROL

PRELAUNCH

3) GUIDANCE PLANE IN TDMA COORDINATES BASED UPON EXTRAPOLATED TARGET BY C&C

4) SELECT MISSILE SHIP AND LAUNCHER

POST LAUNCH

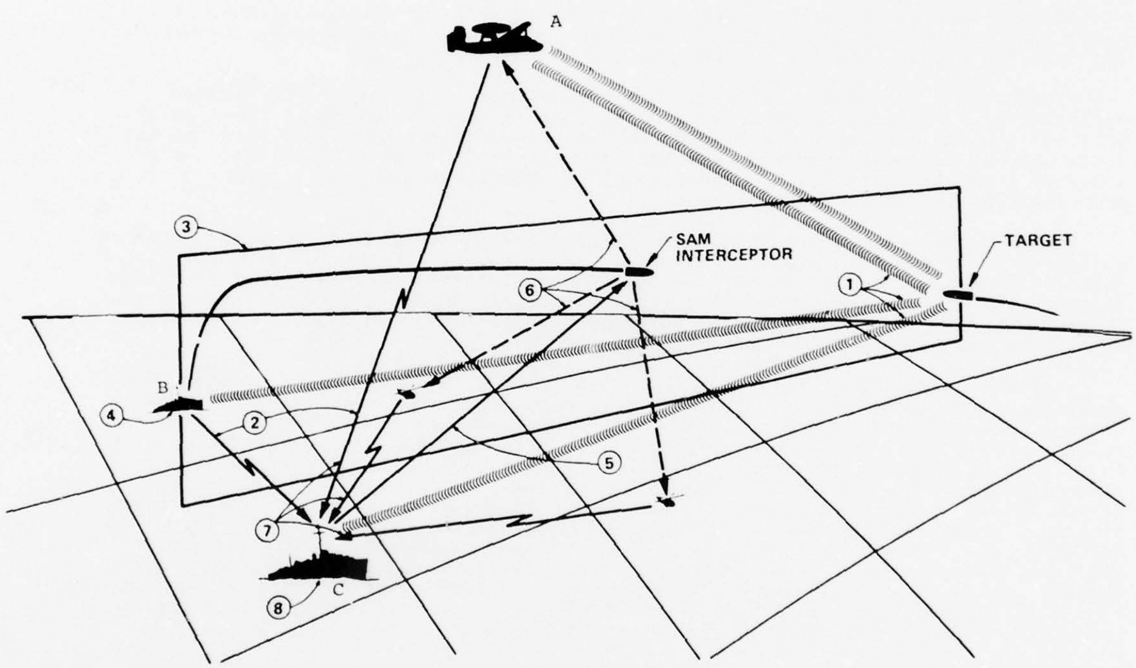
COMMAND AND CONTROL TRANSMISSION TO/FROM SAM

5) COMMAND & GUIDANCE DATA

6) SAM RESPONSE TO COMMAND

7) TRANSMIT TOA SIGNALS TO ASSIGNED TDMA COMMAND & CONTROL MEMBER

8) DEVELOP INTERCEPTOR MISSILE POSITION



MISSILE GUIDANCE AND CONTROL

FIGURE 1

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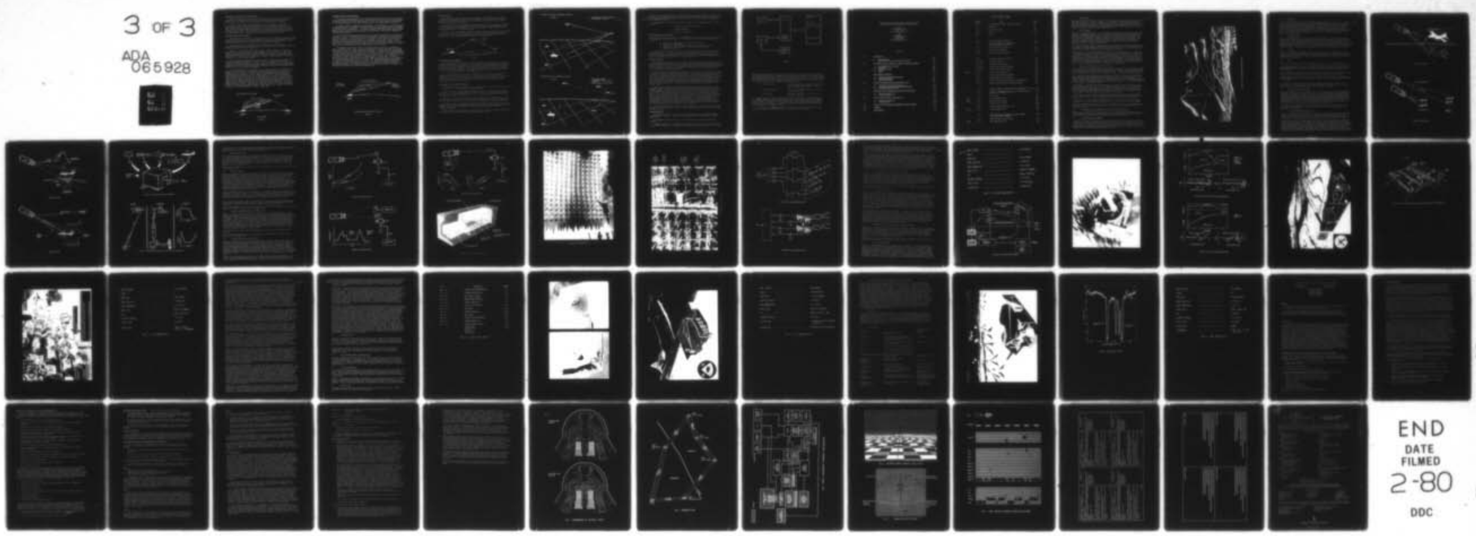

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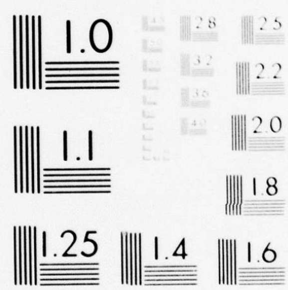


MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

INTELLIGENCE GATHERING AND DISSEMINATION

These activities can be described as requiring signal intercept, selection of significant or correlated signals, providing emitter locations where possible, identification of emitter type, correlating data and reporting to command.

As part of the emitter location process, it is advantageous to utilize the high accuracy relative navigation capability provided by TDMA especially for redundant reception of emitter transmissions and the subsequent correlation of the data. Another extremely important feature in terms of data correlation is provided by the ability to "time tag" data accurately for processing and correlation with data from other sources. The secure data distribution capability of this system is, of course, important for communications with other operational elements.

RPV/DRONE CONTROL AND GUIDANCE

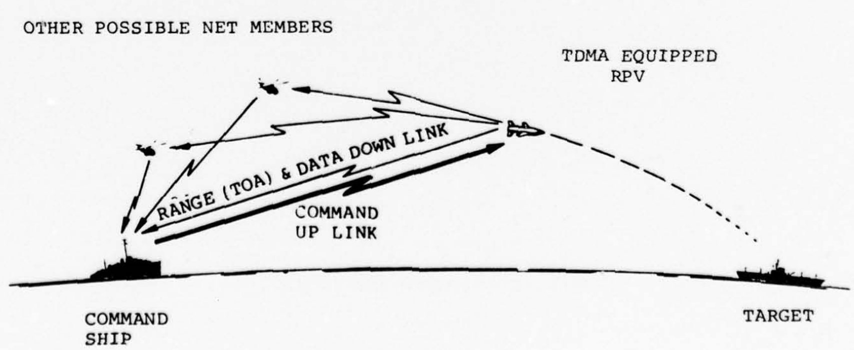
Although missile guidance techniques utilizing Distance Measuring Equipment (DME) and signal Time of Arrival (TOA) measurements are not new, a TDMA compatible terminal offers great simplicity over current systems. Utilizing the TDMA tactical link, a complete dedicated separate guidance link and system is not required.

A TDMA terminal is also applicable to Remotely Piloted Vehicles (RPV). A simplified terminal would provide the A/J data link and relative navigation capability for both the missile and RPV application.

There are a full range of terminal versions to meet a spectrum of vehicle applications. For simple low-cost glide weapons, a basic terminal is ideal as it exhibits a balance between basic guidance features and low cost. For more capable glide weapons and air-to-ground missiles, additional terminal capabilities can be added such as a terminal with an additional receiver, thereby enhancing the anti-jam properties of the basic configuration.

Finally, for more sophisticated vehicles with the capability for self-contained guidance, a different terminal is in order. This is a fully passive terminal (without transmission capability). Since it does not transmit, it is a more secure terminal but it must now synchronize itself passively into the TDMA slot structure. Relative missile location is now computed on board the missile from the TDMA position reports of the other tactical net members.

Typical operation of this terminal is as follows: (Figure 2) A message (for example, an RPV command) is transmitted over a predetermined missile slot. This message, complete with synchronization and header information, is received and recognized by the terminal. A reply signal is assembled containing externally supplied data (targetting, etc.) or a fixed ranging reply. This reply message is then transmitted to the command ship either directly or through a relay aircraft or RPV after a fixed time delay. On the command ship, relative navigation computations are performed upon the receipt of TOA data, thereby placing the RPV in a relative grid. Command information is then uplinked to the RPV in the next chosen TDMA time slot and the process repeated. Note that for navigation, two other net members are shown (Helos). They are not required, however, if relative navigation is not to be performed by TDMA equipment. Also, they are not required if relative navigation is to be performed in a single update mode.



RPV SCENARIO

FIGURE 2

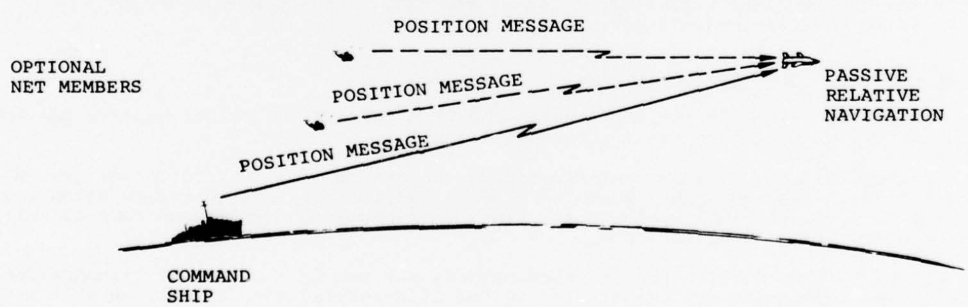
AUTONOMOUS OPTION FOR NAVIGATION

For certain mission applications it may be desirable for the RPV to remain silent until reaching the terminal area. This requirement can be met by utilizing autonomous navigation capability with the TDMA terminal. This is a feature which already exists in the Class 2 terminal, which has been developed by the Singer-Kearfott Division.

A form of relative navigation is performed by the RPV terminal. Once the terminal is synchronized into the TDMA slot structure accurate range measurements are performed in the RPV terminal (passively). Relative navigation processing of these measurements permits the RPV to place itself in a relative tactical grid continuously during flight. Navigation and hence guidance computations are performed in the RPV in a completely self-contained fashion.

There are several reasons, however, why this capability may not be worth the extra complexity. First, the ship must periodically radiate a transmission to the RPV for relative navigation. This radiation may be detected by unfriendly vehicles (provided they are within line-of-sight range). In the simpler active terminal, the RPV must radiate to the ship which then performs the guidance computations. However, this radiation is aimed toward friendly forces and away from unfriendly forces. Since the ship retains the command function and has greater data processing capability, it makes sense to place the navigation and guidance authority here as well.

For comparative purposes, Figure 3 indicates the data flow in an autonomous mode. As shown in this figure, one way transmissions are made from one or more TDMA net members to the RPV. These transmissions consist of normal TDMA Position messages (containing position, velocity, heading, information). The RPV accepts these messages by listening and decoding the data in each assigned vehicle slot, then performs relative navigation computations and locates itself in a relative grid. This location technique is superior to simple multilateration techniques as it is based upon optimally filtering both range (TDMA derived) and dead reckoning derived data. Furthermore, with this technique (unlike multilateration), TDMA range measurements can be processed at various non-synchronous times. It is this fact that permits relative navigation to operate on a "single range update" basis, i.e., updated navigation between only two TDMA net members. With the singular, non-autonomous RPV terminal previously described, navigation and guidance computations are performed aboard the command ship and are based upon TOA multilateration computations.



AUTONOMOUS NAVIGATION FOR THE RPV

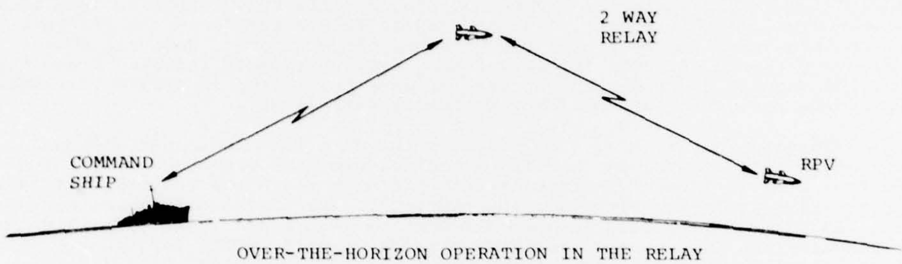
FIGURE 3

RELAY OPERATION

Each airborne terminal contains the provision for relay operation. At TDMA frequencies, transmission is solely by line-of-sight. To achieve over-the-horizon operation with a low flying RPV, a relay operation is probably required. Figure 4 demonstrates this type of operation.

Ranging and sensor data transmitted by the RPV are received by the relay terminal. This data may then be re-transmitted as shown in Figure 4 in a pre-determined relay slot for reception by the command ship. By the same token, a TDMA terminal (e.g., in another RPV) also can be used as a relay.

For the over-the-horizon RPV, TDMA capabilities supplying secure data transmission and navigation seem to be a natural choice. The ability to handle high data rates, multiple users, and supply an inherent flexibility adaptable to different mission scenarios, all seem to enhance TDMA applicability to this problem.



OVER-THE-HORIZON OPERATION IN THE RELAY

FIGURE 4

SEARCH AND RESCUE OPERATIONS

The Search and Rescue Operation, although involving two segments to the operation, requires a high degree of coordination, data transfer and common navigator accuracy for the fulfillment of the mission. It is, in many respects, quite similar to an ASW mission which requires SEARCH and then handoff of a target, for ASW, or of friendly and enemy personnel for RESCUE. It is extremely important for the search vehicles to be able to lock a rescue target in the same grid as that used by the rescue vehicle, if it is not on search during the search operation.

The TDMA equipment may be used to advantage for this mission in that different terminals on various types of vehicles or terminals used in conjunction with a sonobouy will all be inter-operable. This inter-operability of terminals is important and fortuitous since the type of different vehicles used for search and rescue operations may be quite varied depending on the nature of the operation and the availability of vehicles on location.

TARGETTING USING DISTRIBUTED SENSORS

Targetting may be accomplished using the TDMA terminal by either passive monitoring of an emitter or by active radar threat verification.

Figure 5 depicts a Target emitting energy in search of a task force (or ground installation). Airborne TDMA community members receive this energy transmission and time tag it using the TDMA synchronized time. A designated Command and Control unit thus computes the emitter location.

Figure 6 shows a situation in which members may verify a threat by illuminating a target. The reflection may be utilized by the illuminating aircraft (A) as well and by other passive elements (B and C) in order to provide emitter location information.

The utilization of the distributed sensor concept in conjunction with a precisely defined relative grid and synchronized time from the TDMA system provides a wide range of application oriented capabilities, heretofore not available to cooperative community operation.

These concepts may be utilized to provide smaller target acquisition baskets while providing increased survivability by decreasing radar on-time.

TARGETTING THROUGH DISTRIBUTED SENSORS

FIGURE 5

o PASSIVE SHIP FORCE MONITORING
o TOA EMITTER LOCATION

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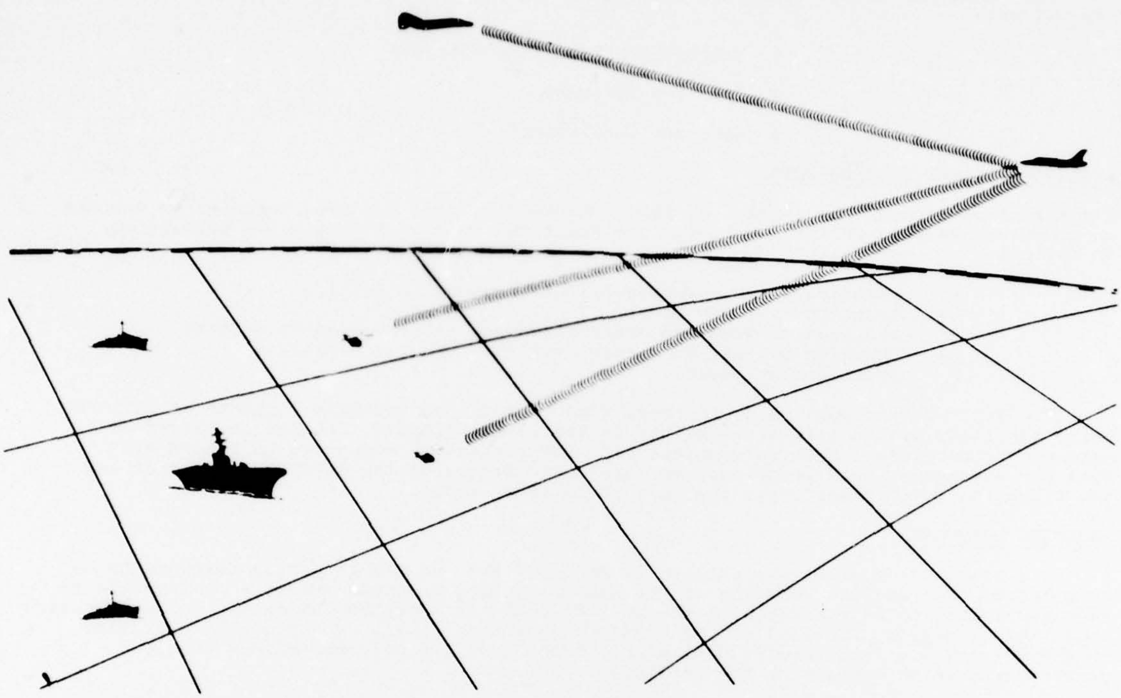
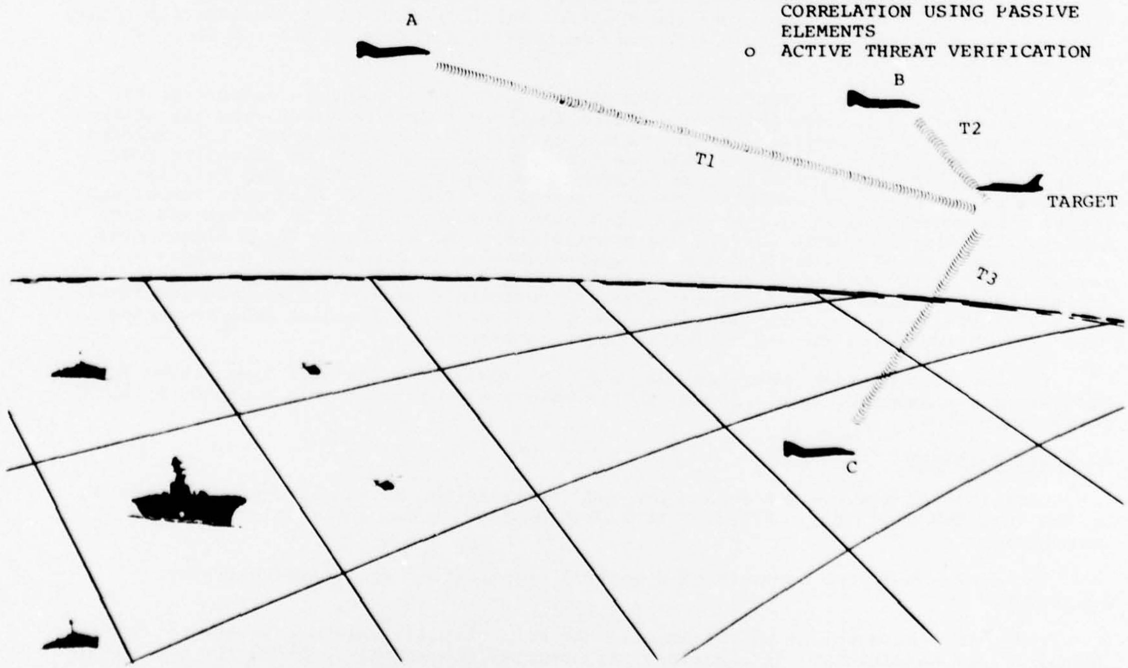


TARGETTING USING DISTRIBUTED SENSORS

FIGURE 6

o LONG RANGE THREAT CORRELATION USING PASSIVE ELEMENTS
o ACTIVE THREAT VERIFICATION



23-12 A system of this sophistication with its high data rate communications and complex state-of-the-art dual grid Kalman filter analytics requires dynamic, automated validation techniques to assure the integrity of the software. This has motivated the development of the COMMANDS facility.

The remainder of this paper details three aspects of the COMMANDS system. These aspects are:

- o Closed-Loop Real Time Software
- o Baseline Software
- o Hardware Configuration

REAL TIME CLOSED LOOP SOFTWARE

The Real Time closed loop portion of the COMMANDS simulator has been designed to execute on a Hewlett-Packard HP21MX computer. The functions of this software are summarized as follows:

a) Execution of Dead Reckoner (D/R) Navigation Model
b) Execution of TACAN Model
c) Execution of TDMA and Radio Frequency (RF) Message Processing
d) Execution of Port Processing for the terminal's message communication ports.

The Input-Output portion of the Real Time Closed Loop Software consists of buffers whose functions are to initialize output to the flight computer; to act as interface between the baseline generated magnetic tape and the flight computer; to record all data for subsequent data reduction; to record the status of the COMMANDS simulator and the Flight Software under test, and to contain error codes.

BASELINE SOFTWARE

The baseline programs are written in ANSI FORTRAN and are primarily designed to be executed on the IBM370. Versions of the data reduction programs are written in Hewlett-Packard FORTRAN to be exercised on the HP-21MX computer operating under the Hewlett-Packard Real Time Executive (RTE-IIB) on the COMMANDS facility itself. The programs operating on the HP-21MX are functionally equivalent to and conditionally portable with their counterparts which operate on the IBM370.

The baseline programs generate a time associated data base for use as input to the Closed Loop Software. The program calculates true geographic and relative grid positions for the test vehicle and other community members. It provides ranges between the OFP vehicle and the members including simulated slot jitter, TOA error and blackout conditions. The program maintains the community time of day and provides for discrete step changes in a large variety of parameters including coarse sync indicators, RTT range and SRN, oscillator and power amplifier ready indicators, geographic update parameters, slot jitter, TOA error, fixed format message headers, initialization data, message repetition rates, internal message fields, polling and machine acknowledge events and fee text message changes.

The program provides simulated fixed format position messages received from the community members mentioned above in addition to fixed format messages from any other community member. In addition, it provides free text received messages. All emulated received messages have associated with the messages error indications including code error counts, erasure count and check on inner code parity. The received Position-messages are also time synchronized as to internal position data, time slot number and range. The program supplies simulated input data from both the fixed format and free text portion including both control and message data, and simulated TACAN signal data representing a range and bearing model implementation. It provides for periodic signal blanking DC, AGC lock/unlock condition and valid/invalid indications. Also implemented is a simulated co-channel signal interference mode. The program simulates (in conjunction with the real time Closed Loop Software) the inertial dead-reckoning unit and provides data for the navigation port interface.

The baseline program generated data base is supplied to the Real Time Closed Loop Software via a magnetic tape. Inputs to the Baseline program are via a standard IBM370 format card deck.

COMMANDS HARDWARE

This section describes the function and interrelation of the hardware components in the COMMANDS simulator. Figure 7 is a diagram showing the interaction of these components.

The COMMANDS system consists of a central computer and associated peripheral equipment.

COMMANDS simulates, in real time, all the data normally received by the OFP under actual flight conditions. In addition, the computer continuously checks the OFP output

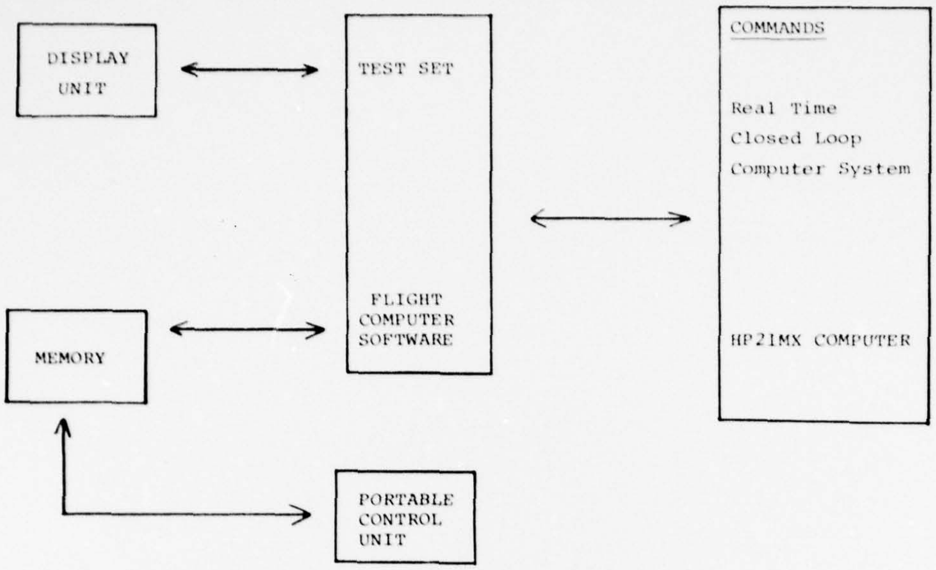


FIGURE 7

data and produces appropriate responses, and/or adjusts the simulated data as required. Actual flight conditions are simulated by eleven test cases which are designed to check all combinations of the major functions to be performed by the OFP. This, then, provides a complete validation of the flight software in the laboratory prior to flight testing. The trajectory driver tape is mounted on one tape drive while the other drive is used for recording the OFP output. Other peripherals used are:

- o CRT I/O Terminal - provides the means for entering some program initialization data, as well as examining data on the OFP output tape.
- o Cartridge Disk Unit - stores the software needed in the operation of the test station.
- o Line Printer - produces a hard copy of the output from the flight computer.

The Operational Flight Software Test Set handles the data flow between the OFP, the COMMANDS simulator, and the Display Unit (DU). The Test Set controls the communications between the OFP and DU and allows them to operate independent of the external test computer. In addition, the Test Set provides controls that simulate the Mode Control Unit (MCU) functions. The Test Set provides all necessary clocks and a real-time interrupt to all external devices.

In conclusion the COMMANDS system offers a reliable low cost mechanization that provides for the thorough verification of the Airborne Flight Software.

RADIO FREQUENCY (RF) HOMING MISSILE GUIDANCE & CONTROL SIMULATION TECHNIQUES, FACILITIES, AND EXPERIENCES

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CONTENTS

LIST OF FIGURES & TABLES

1.0 INTRODUCTION

This paper describes the basic RF homing missile guidance simulation problem and solutions that have evolved over the past 20 years. Starting from simple component transfer function measurements in the late 1940's and early 1950's, the evolution of RF homing guidance simulation techniques is followed to present day hardware-in-the-loop facilities at Boeing, the United States Army's Radio Frequency Simulation System (RFSS), and the United States Navy's Central Target Simulator (CTS).

1.1 Why Simulate?

Simulation is performed because it is an effective management and engineering tool for analysis, development and test. Missile guidance systems have become too complex to be adequately developed and evaluated by reliance on limited analytical simulations and sparse flight test programs. Managers are expected to make difficult decisions with increasing risk based upon decreasing technical data at key program milestones. Yet program costs continue to climb and accelerate, further complicating the manager's decisions dilemma. Hardware-in-the-loop missile guidance simulations can provide meaningful answers to tough system questions.

There is also a large financial advantage in using simulation over actual flight tests; for example the Army recently performed 3000 simulated flights in three weeks at a cost of $100,000. Comparable actual missile flights could have cost at least $140,000,000 for the missiles alone and taken at least two years to perform.

Technically such a simulation capability provides a means to perform; (1) a reduction in development time and failures for new guidance and counter-guidance system designs, (2) preflight simulation to ensure that the proposed missile flight test is within the capability of the weapon system, (3) post flight simulation to exactly characterize a flight anomaly, and (4) simulate the effect of aging components to determine the performance of the "wooden" missile as a function of storage time, (5) a thorough mapping of the missile performance envelope in a controlled environment, (6) support throughout the life cycle of the missile system. Significantly hareware-in-the-loop simulations bridge the gap between analytical simulation and flight testing.

1.2 Major RF Guidance Simulation Facilities

The RF homing missile guidance and control simulation problem is to realistically create an RF target and background environment, subject the RF homer to this environment, close a missile guidance loop around this RF homer, perform real-time hardware-in-the-loop giudance tests which result in miss distances the same as actual missile test flights.

Early simulation techniques were done on a piecemeal basis. Tests were run on motors, wind tunnel tests were performed for aerodynamic information, open loop RF homer tests were performed, measurements were made of targets and all were then mathematically modeled; these models were then put in an analog computer and a missile-target engagement scenario flown. The primary output was the system miss distance. This was characteristic of the period up to the middle 1960's.

In the middle of the 1960's Boeing developed it's Terminal Guidance Laboratory (TGL). This laboratory employs a 25' X 25' X 50' anechoic chamber, a 16' X 16' electronically steerable target array at one end of the chamber and a full scale hydraulic flight table to hold the missile homer at the other end of the chamber. This laboratory was used to make direct comparisons between actual missile flights and simulated flights. The results demonstrated the validity of this type of simulation.

In 1975 the U. S. Army's Radio Frequency Simulation System (RFSS) became operational, and the characteristics of this facility are described along with some representative tests. This facility is the most sophisticated of it's kind in the free world.

In 1980 the U. S. Naval Research Laboratory will bring on-line their Central Target Simulator (CTS), and the general characteristics of this facility will be described.

Finally, a look will be taken at the future including the exciting new radiometric and millimeter RF homing simulator area which is important for new applications such as anti-tank missiles.

2.0 RF HOMING MISSILE GUIDANCE AND CONTROL SIMULATION PROBLEMS

Modern day RF homing missiles must operate in an electromagnetic environment characterized by maneuvering targets, multiple targets, background clutter, multipath and ECM. Figure 2.0-1 illustrates this for an active air-air RF homing missile.

2.1 Single Maneuvering Target

This target situation is characterized by rapidly changing target kinematics and radar signature. As the target maneuvers prior to intercept, it will present a different aspect to the RF homer and it's basic signature will change. The signature will undergo wide variations in amplitude due to the changing aspect of the target with respect to the RF homer. Target angular glint, amplitude scintillation, and radar crosssection all change in a dynamic manner during the target maneuver.

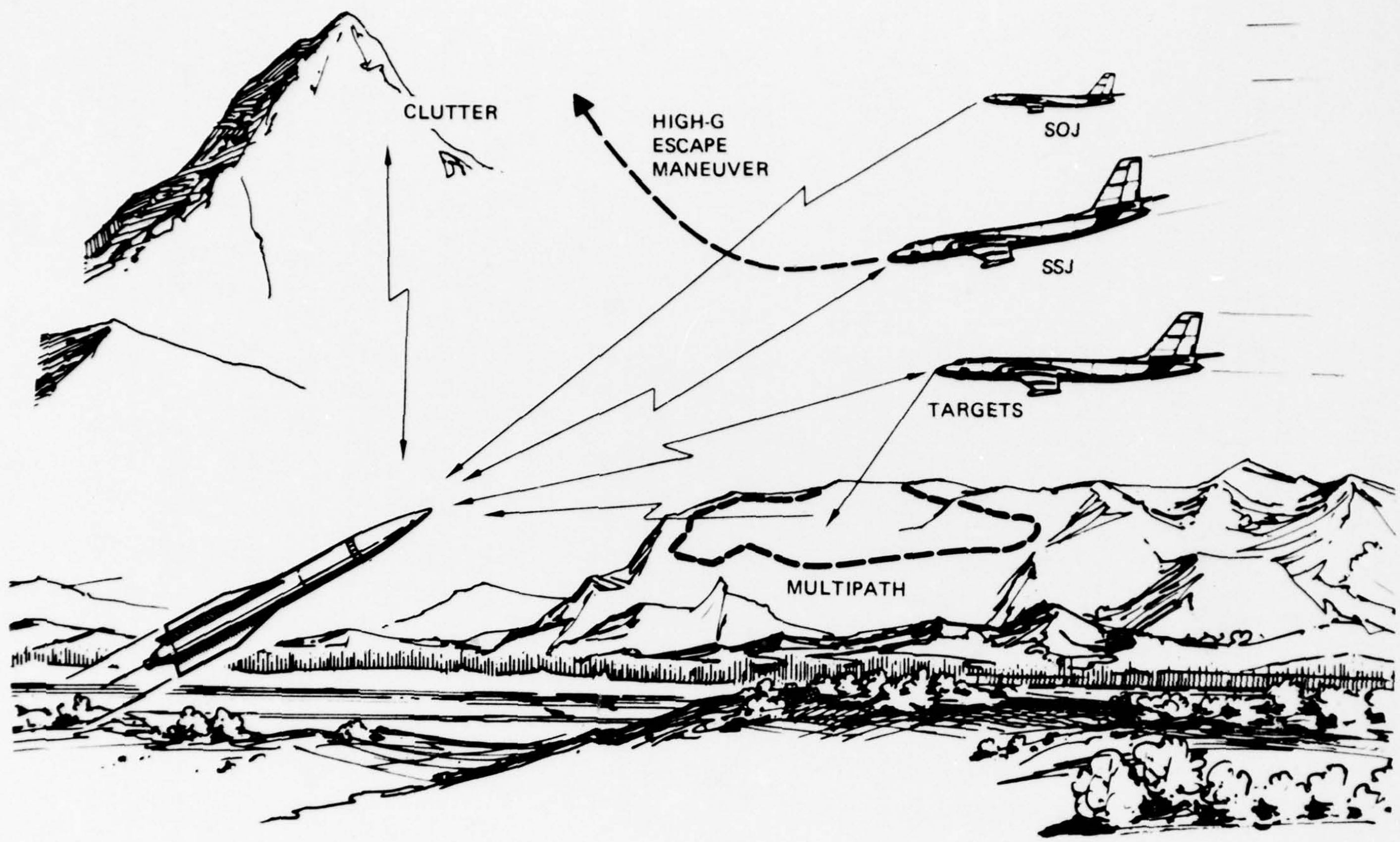

Figure 2.0-1. RF Homing Missile G&C Simulation Problem

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● PLAN VIEW

2.2 Multipath

A low altitude target gives rise to an image phenomena shown in Figure 2.2-1. Some of the RF guidance signal energy bounces off the target onto the sea (or ground) and gives rise to an apparent target flying under the sea at a distance equal to the real aircraft's distance above the sea. The guidance problem occurs when the missile has to choose one target or the other. This transition occurs when the missile closes in range to the point where both targets are no longer in the homer's antenna beam, and the missile has to choose one target or the other. This is a very difficult choice and is frequently made so late in the intercept that the miss distance is significantly greater than that for a single target. 24-5

2.3 Multiple Targets

Two or more targets in close formation have a vastly different target signature than a single target. At ranges where the two targets are not resolved by the homer's antenna (Figure 2.3-1), the apparent target centroid or homing point can be well outside an imaginary sphere containing both targets. This is due to the individual signals from each target interfering with each other in the homer's signal processing system. As the targets cannot keep perfect formation their return signals alternately add and subtract as the targets "jockey". This gives rise to large angular glint and amplitude scintillation.

2.4 Clutter

Figure 2.4-1 shows both sidelobe clutter and main beam clutter. Clutter is a much stronger signal than the target, and will mask the target unless some method of clutter suppression is employed in the homer. Range, angle, and velocity gating are the common methods employed. Thus the clutter environment has extensions in both space and time which must be simulated.

2.5 ECM

Figure 2.5-1 shows the ECM situation typical of today's missile-target encounters.

The self-screening of jammer (SSJ) is effective when the target is within the main beam of the homer antenna. The target generates deceptive signals (by various means) which make the homer believe there are other targets displaced in angle or range from the actual target.

The self-screening jammer makes the deceptive targets stronger in signal amplitude than the real target return and thus make them appear more attractive to the homer.

Another general type of ECM is denial ECM in which a standoff jammer (SOJ) generates signals which are much stronger than the real target return which hides the real target from the homer.

In summary, the accurate dynamic simulation of these environments in space, time and frequency is the central RF homer missile guidance and control problem. The next section covers the early attempts at simulating these environments.

3.0 Early Simulation Techniques

Early simulation techniques (circa 1945 to 1965) are characterized by the methodology depicted in Figure 3.0-1. Motor tests were run and from the test resutls mathematical models were developed of the missile propulsion performance; wind tunnel tests were performed and mathematical models were developed of the aerodynamic performance of the missile; homer tests were performed against simulated targets and mathematical models developed for the homer acquisition and homer tracking performance; and target radar signatures were measured. All of these mathematical models were then programmed in an analog computer. The primary result was miss distance and the sensitivity of miss distance to various system performance parameters was measured.

To illustrate the development of the mathematical models and the techniques used in experimental measurements, the following sections will cover the development of the math models of the homer.

3.1 Single Non-Maneuvering Targets

Figure 3.1-1 shows the three stages in developing the mathematical models of the homer during the early days of simulation. A scenario was postulated in which the missile closes on the target in range. The missile homer's ability to acquire and track the target were statistically measured and performance models developed from the data.

The stationary horn was placed in the far field of the homer's antenna, a transponder connected to the horn, and adjustments made to the signal strengths so that specific ranges could be simulated. The output of the homer was recorded, and the data analyzed. The detection probability of the homer was then modeled as a function of the strength of the signal return from the transponder to the homer as shown in the figure. In addition, the ability of the homer to track the target was measured as a function of range and a model developed. The dotted line in the figure indicates that this portion of the model was based upon other experiments which measured the effective angular glint

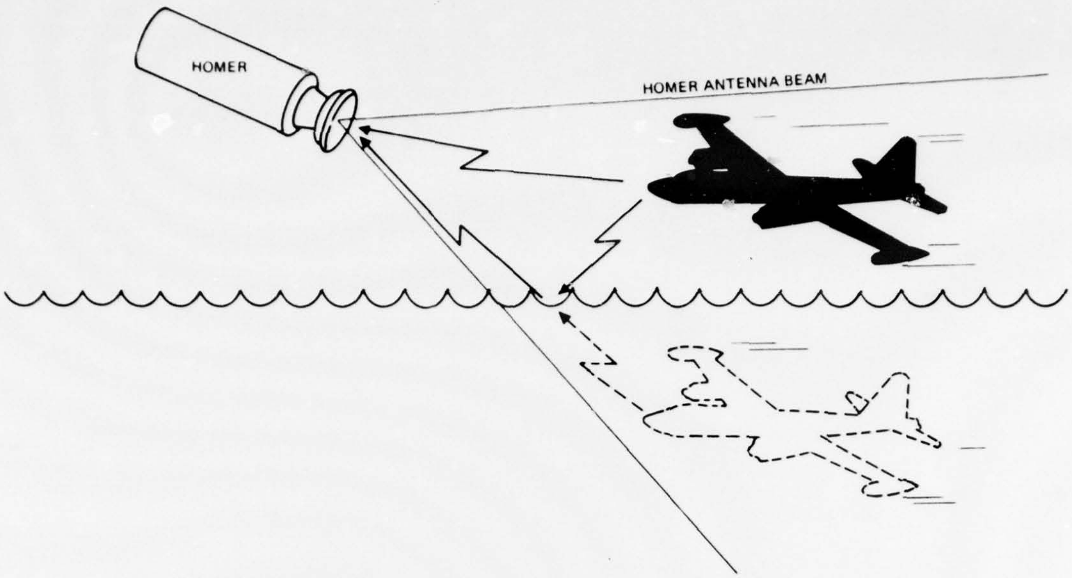

Figure 2.2-1. Multipath

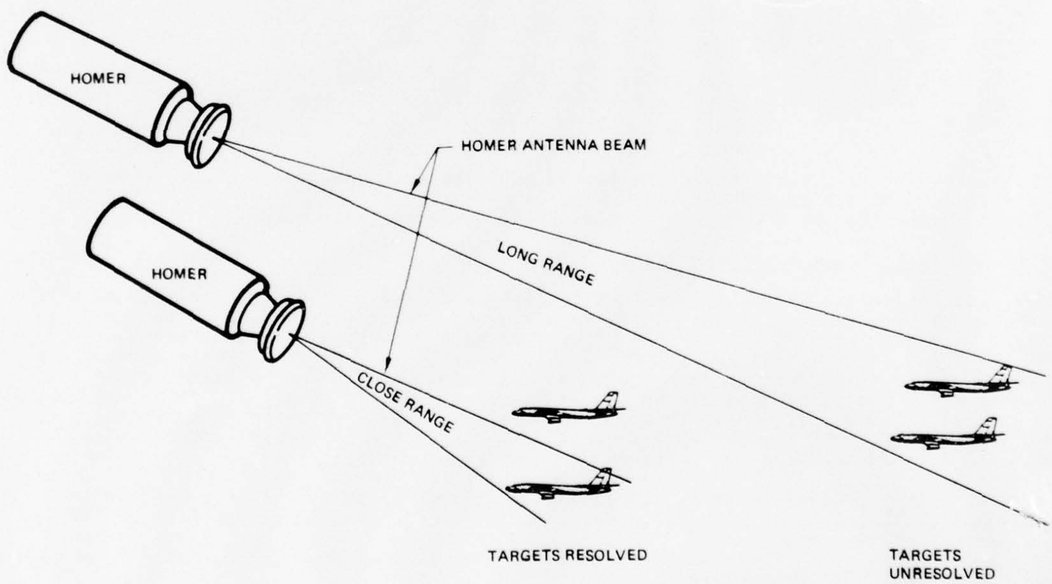

Figure 2.3-1. Multiple Targets

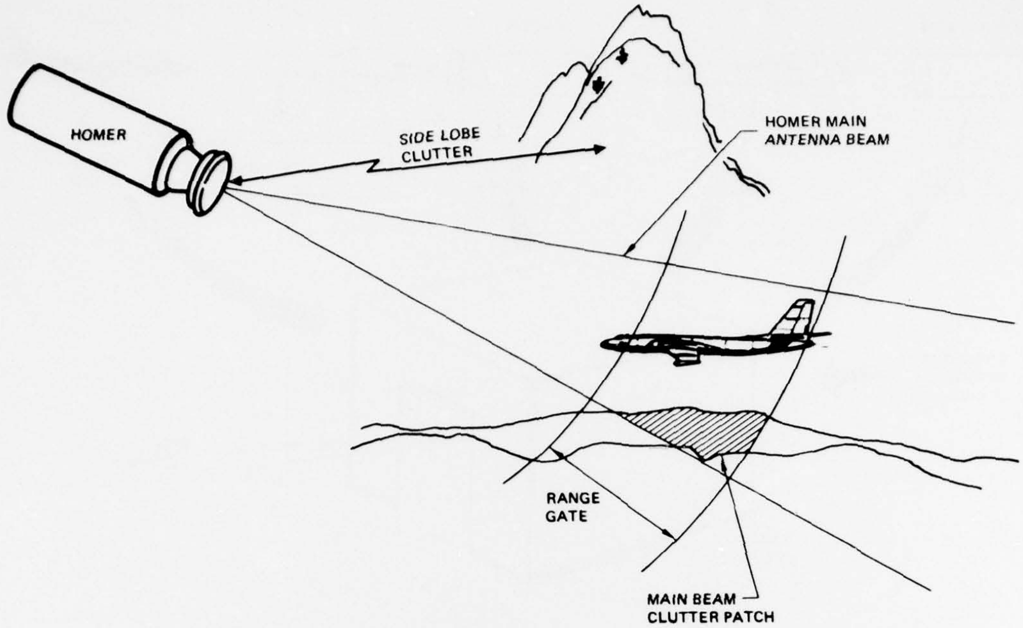

Figure 2.4-1. Clutter

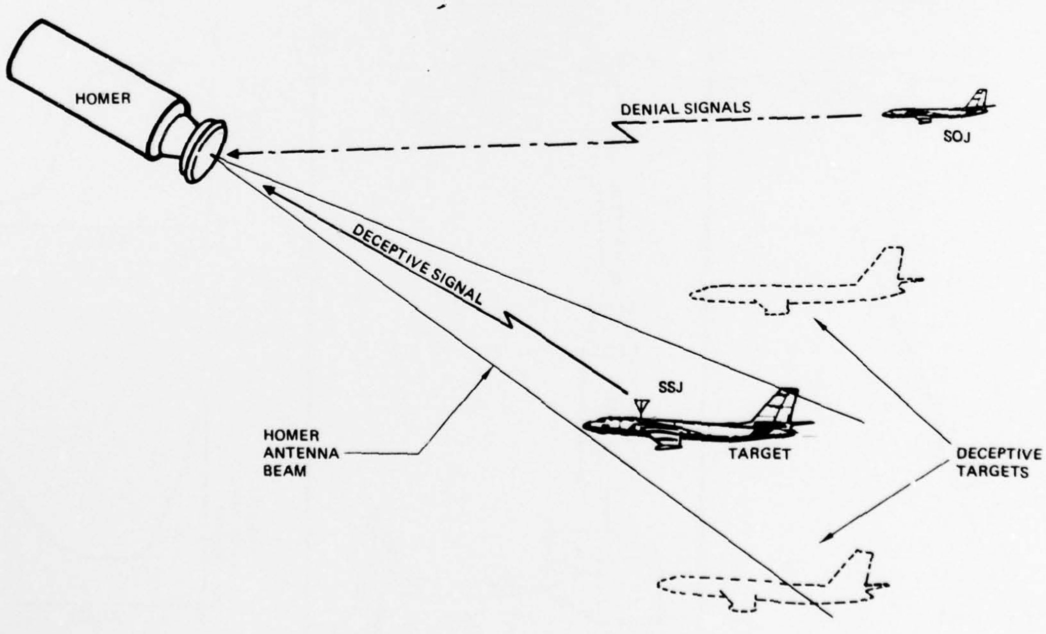

Figure 2.5-1. ECM

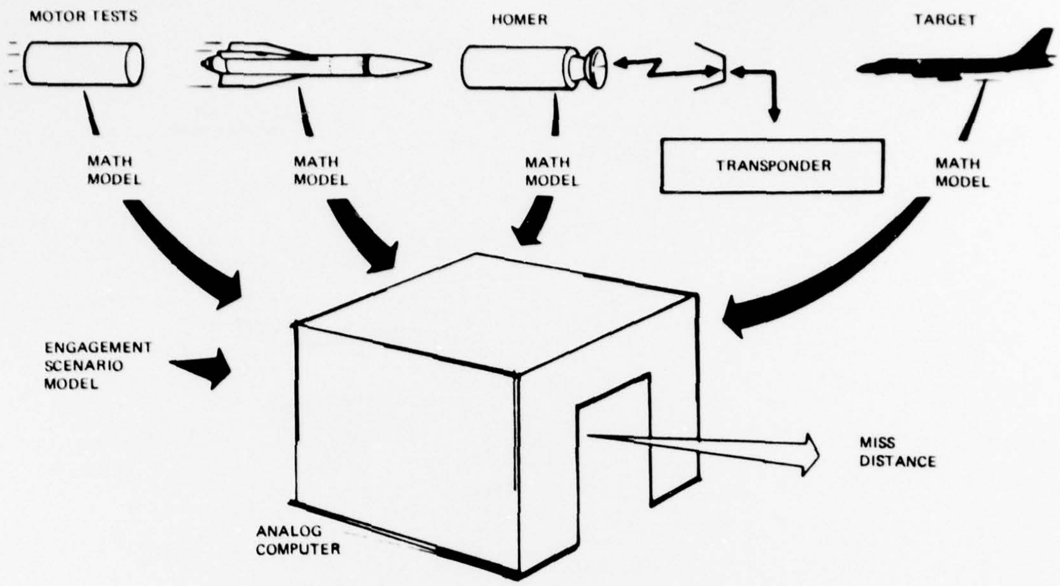

Figure 3.0-1. Early Simulation Techniques (1945-1965)

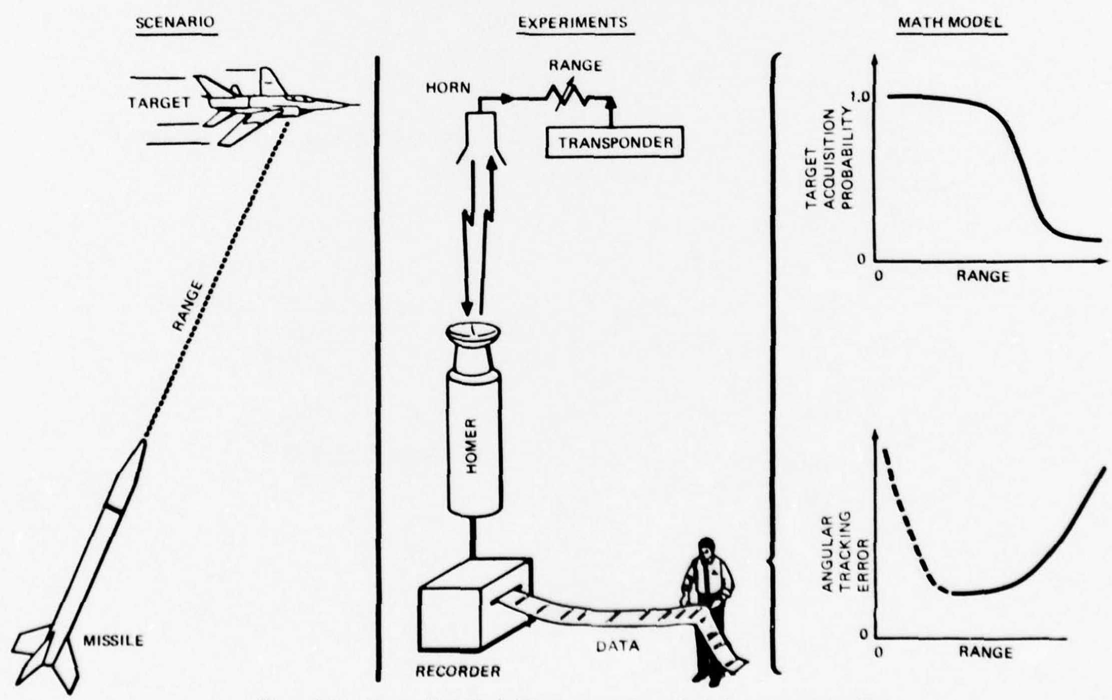

Figure 3.1-1. Homer Math Model Development for a Single Nonmaneuvering Target

of the target. This technique proved to be adequate as long as the target was relatively non-maneuvering. If the target maneuvered, however, the data for the mathematical model needed to be modified as shown in the following section.

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3.2 Maneuvering Targets

Early methods used to characterize maneuvering targets were very crude at best. Figure 3.2-1 shows how experimental data was gathered for this situation. A horn was mounted on a boom and rotated at different velocities. The range to the target was changed by means of the range adjustment shown. For specific rotational speeds of the horn, the tracking error of the homer was measured, and the data plotted as shown in the figure. This was a crude simulation of a high-G escape maneuver but did indicate how well the homer could track a moving target.

Other situations important to simulate are the multipath and close spaced targets discussed in the next section.

3.3 Multiple Targets

Figure 3.3-1 shows an early day dual-target simulation. The second target was simulated by the addition of a fixed horn near the path of the moving horn. This simulated the effect of the homer closing on two close spaced targets. As the homer closed on the targets they approach the angular extent of the homer's antenna beam (Figure 2.3-1). This effect was simulated by moving the targets apart rather than moving the homer in on the targets. The specific tracking error shown in the plot was a function of how fast the moving horn moved past the stationary horn (which simulated how fast the homer closed on the targets), and the beamwidth of the homer. The homer had a decision to make when the horns approached the main beamwidth of the homer. Sometimes it would choose the fixed horn, sometimes it would choose the moving horn. The resulting tracking error then would depend on which one was chosen,and the resulting gyrations of the homer were random and varied from flight to flight.

If more than two targets were to be simulated, a technique called signal injection was used which is shown in Figure 3.3-2. The homer is tracking a moving target while a signal generator injects many targets or interfering signals into the homer. This was used to measure the performance of the homer in a high signal density environment. Both target acquisition probability and tracking error as functions of range were measured.

Early simulation techniques were characterized by open-loop non-real time experimental measurements; mathematical models were developed; from the experimental data an analog computer was then used to "fly" scenarios, and miss distance was computed.

With the development of large scale electronically controllable antenna arrays and anechoic chambers the situation radically changed about 1965.

4.0 BOEING'S TERMINAL GUIDANCE LABORATORY (TGL)

Figure 4.0-1 shows the terminal guidance laboratory (TGL) developed by Boeing. This laboratory was placed in use in 1967. It consists of a large anechoic material-lined chamber with a large electronically steerable target array at one end of the chamber and a hydraulic table (onto which the actual missile homer is mounted) at the other end of the chamber. There is a separate shielded room in which the RF and ECM signals are generated. These signals are coupled into the target array which then broadcasts them to the missile under test. The digital computer controls the position and amplitude of the signals from the array. Target, aero, and propulsion models are also resident in the digital computer in addition to target scenario. The digital computer "flys" the missile and the targets in real-time while simultaneously controlling the electronic steerable array which broadcasts the RF signals

Thus, for the first time a real-time-hardware-in-the-loop guidance simulation was performed.

One of the key hardware items in this simulation is the electronic sterrable array which will be described next.

4.1 Electronically Controllable Array

Figure 4.1 shows two views of the terminal guidance laboratory (TGL) target array. It consists of 256 antennas mounted on one foot centers in a 16' X 16' array. Solid state switches are used to connect the RF signals to those antennas which are to broadcast to the other end of the chamber. The RF signals from the RF generation and ECM equipment are conducted through semi-rigid coaxial waveguides to the individual antennas. Figure 4.1-2 shows how the array is programmed. At any instant of time one target is broadcast through four adjacent antennas. The computer uses look-up tables to determine the control signals for the solid state switches so that the desired A, B, C, D antennas radiate the target. This selection process provides coarse target positioning on the array.

Figure 4.1-2(b) shows how the target is positioned within the quad of antennas. The array is carefully phase-adjusted so that all the signals radiating from the antennas are of equal phase as shown in the figure. Computer controlled attenuators are then

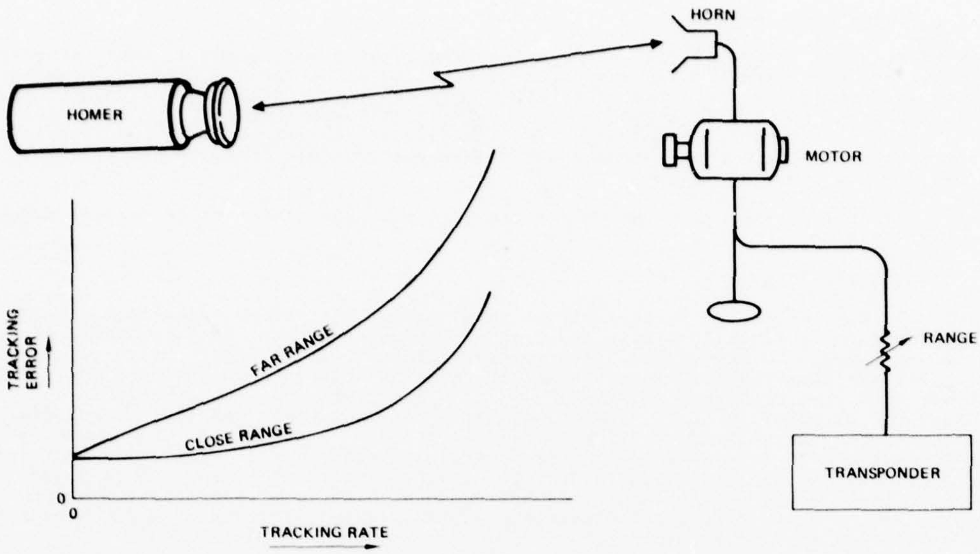


Figure 3.2-1. Moving Target Simulation

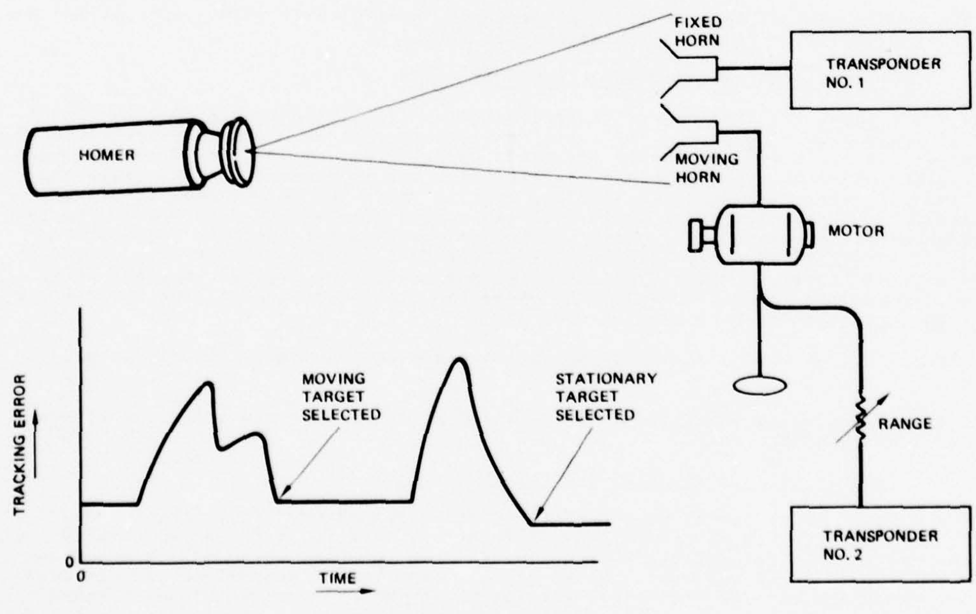


Figure 3.3-1. Dual Target Simulation

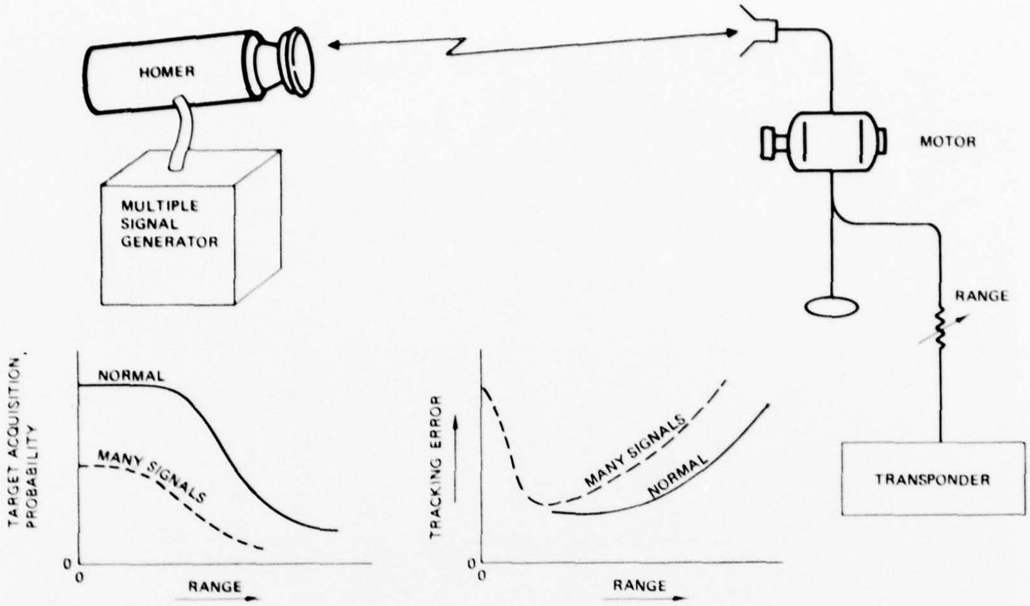


Figure 3.3-2. Multiple Signal Simulation

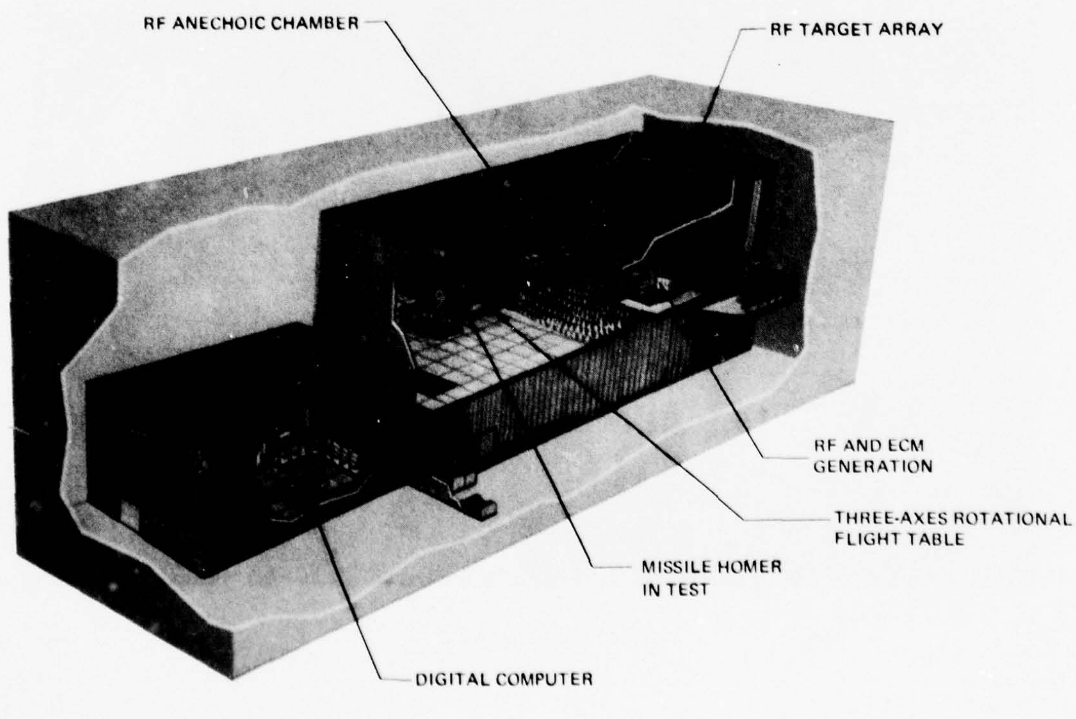


Figure 4.0-1. Terminal Guidance Laboratory (TGL)

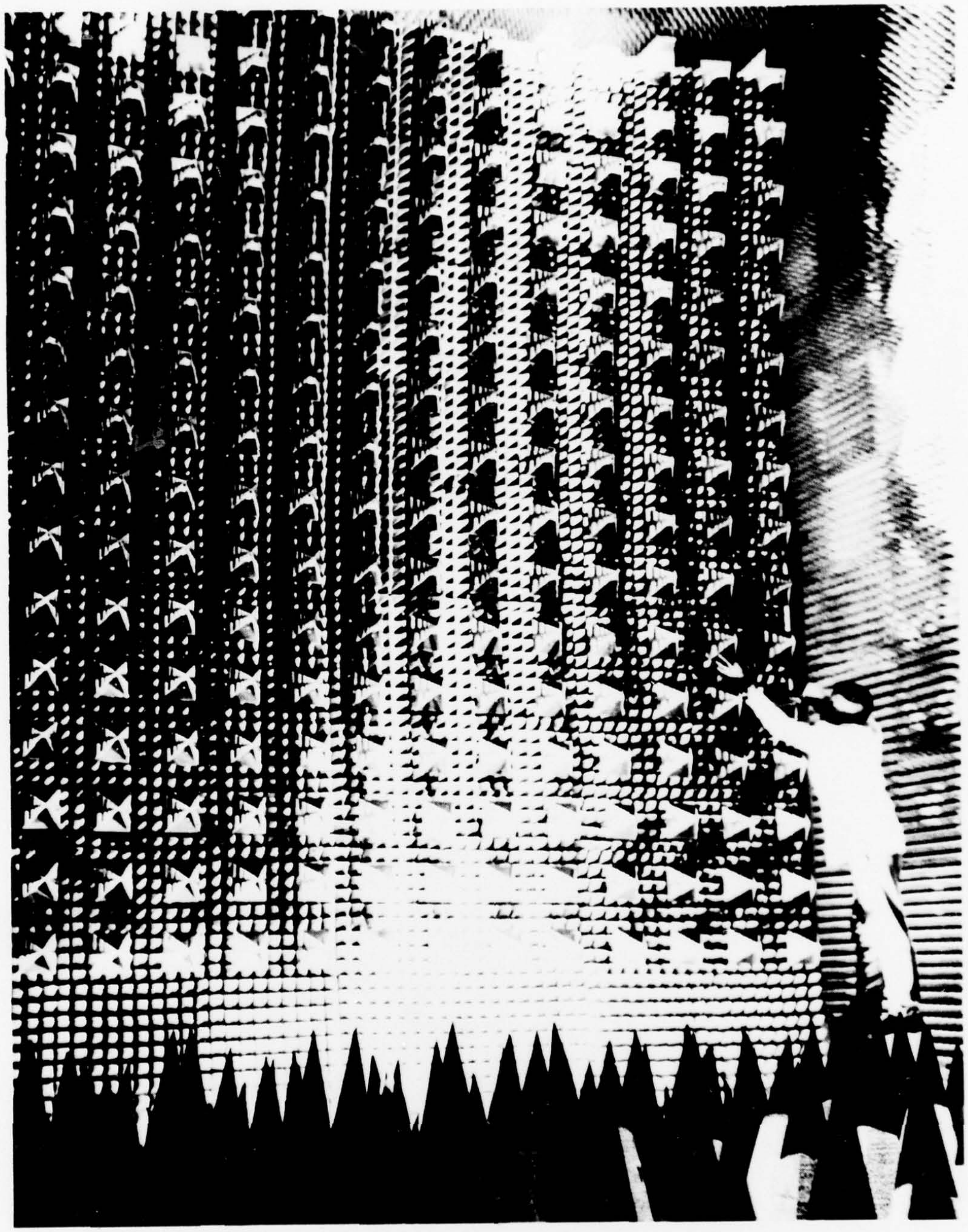


Figure 4.1-1(a). TGL Array (Front)

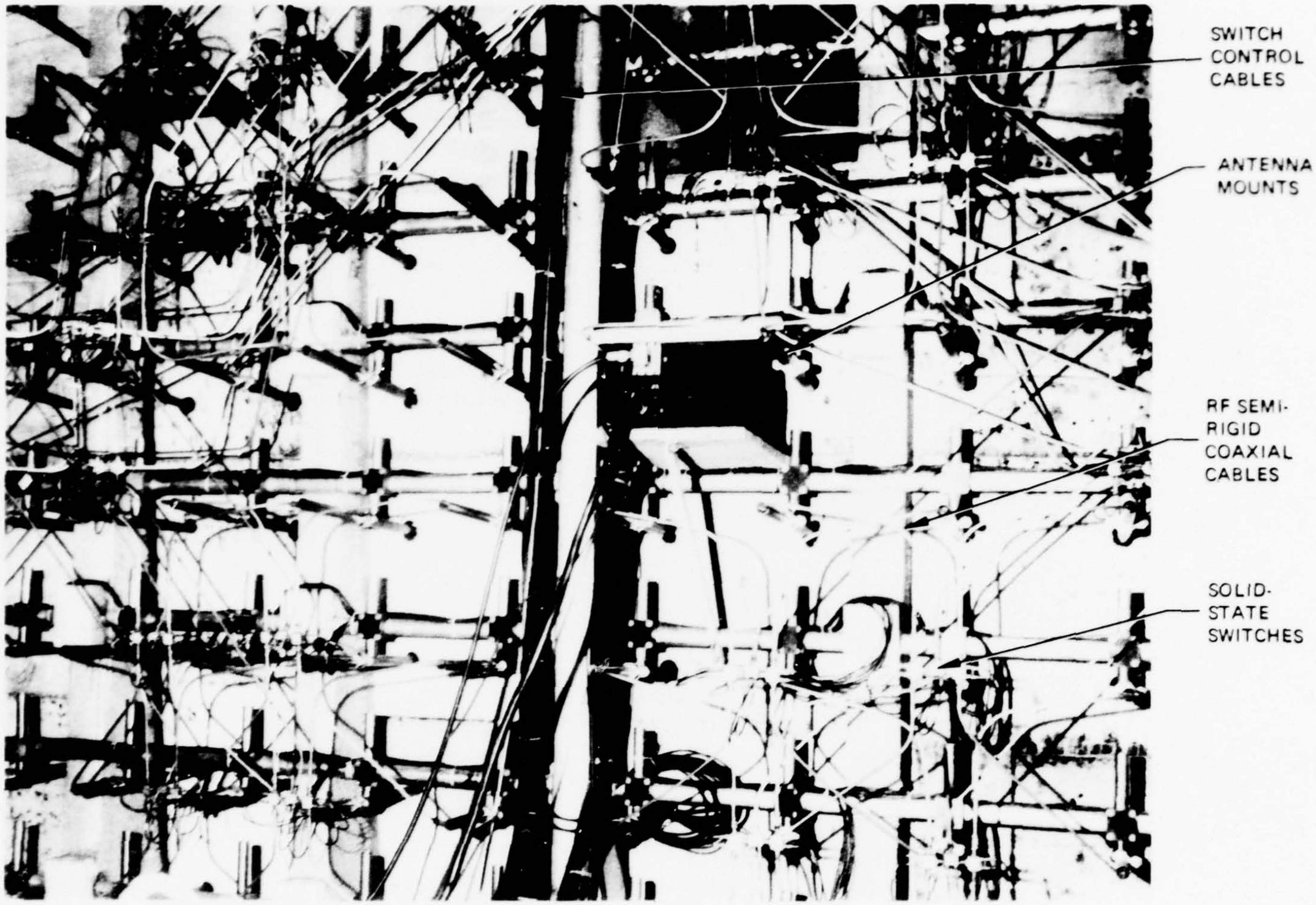


Figure 4.1-1(b). TGL Array (Rear)

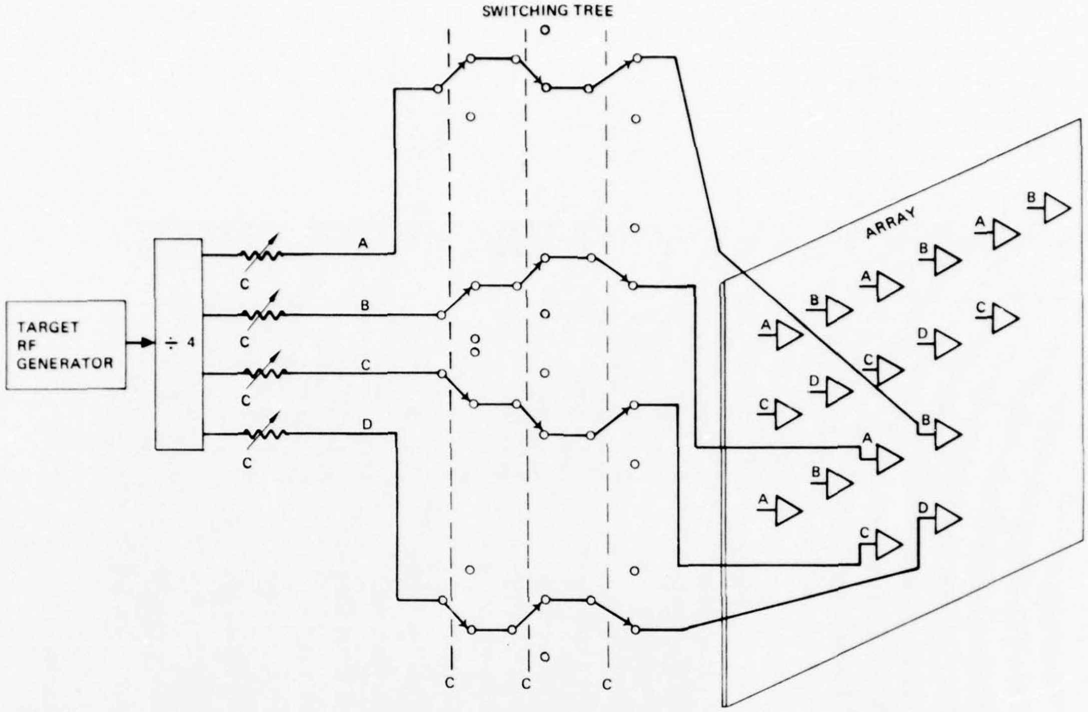


Figure 4.1-2(a). Target Coarse-Position Control

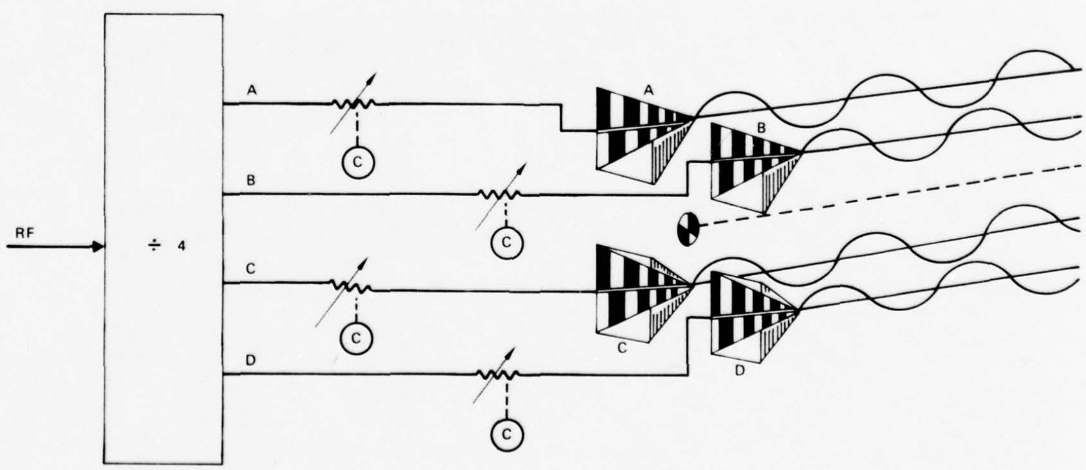


Figure 4.1-2(b). Target Fine-Position Control

adjusted to control the relative amplitude of the signals radiated from the four antennas, and for the situation shown places the apparent phase center of the target in the center of the quad of antennas. By properly adjusting the signal amplitudes, the target is moved to any desired position within the quad of antennas at very fast update rates. Table 4.1-1 gives the resulting characteristics of the array. 24-15

The basic accuracy of the system in angular position is 5 milliradians which is a function of the spacing of the antennas, the distance between the array and the homer, and the accuracy of the relative signal amplitudes radiated from the antennas. The system can generate two independent targets and the array can be updated every two milliseconds. Since it is electronically controlled, the target's path on the array can be altered extremely fast so there is actually no practical limit to the target's velocity and maneuvering capability. The target size is a function of the power that can be generated and broadcast from the array. The TGL can simulate targets from small missiles to B-52 type bombers, and can generate the two basic types of ECM discussed previously. The frequency coverage of the TGL is from 2 to 12 GHz. The field of view is 30 X 30 degrees. The polarization is linearily fixed, but can be manually adjusted.

4.2 Closed Loop Real Time Guidance Simulation

Figure 4.2-1 shows how the TGL is used in a closed-loop guidance simulation. The guidance loop consists of the following elements: the target array, the signals broadcast from the array to the homer, the steering signal output of the homer, the autopilot, the elevon servos (these two blocks can either be in software or hardware depending on the requirements of the simulation and the status of the hardware), the missile propulsion, the missile aerodynamics, and the relative kinematics. These elements comprise a closed-loop real-time hardware-in-the-loop digital guidance simulation which operates from missile firing through the fuzing function.

This system was put together in 1968 and actual missile hardware brought in and tested as discussed in the next section.

4.3 Simulation Verification

Verification of this type of simulation was performed with actual missile hardware shown in Figure 4.3-1. Many tests were performed including maneuvering targets, multiple targets, ECM'ing targets, etc. Figure 4.3-2 shows a comparison between an actual missile flight and a simulated flight for a high-G target escape maneuver. The top of the figure shows the missile and target cross-range trajectory profiles in a plan view. The profile view shows the missile and target elevation trajectories. The situation depicted is a constant altitude target which pulls a 5.5 G escape maneuver 6.4 seconds prior to intercept. Typical data is shown comparing actual flight test data to simulated data. The actual flight test data is the telemetered data from the missile, and the simulated flight test data is from the equivalent outputs of the simulation. Notice that the major characteristics of the simulated flight are remarkably similar to those of the actual flight.

Another simulation verification test is shown in Figure 4.3-3. The targets are two co-altitude aircraft with constant separation. As the missile closes on the target formation it has to make a decision as to which target to choose. In the actual flight test, the missile first chose one, then the other and finally back to the first one. The simulated data shows a similar situation occurred. No two runs appeared exactly the same because of the dynamic interaction between the guidance system and the homer's target choice. The homer did not always choose the same target. After several years of using this simulation the Army decided to build the Radio Frequency Simulation System (RFSS) which is discussed in the next section.

5.0 MIRADCOM'S RADIO FREQUENCY SIMULATION SYSTEM (RFSS)

Figure 5.0-1 shows the Army's Radio Frequency Simulation System (RFSS) which is larger than the TGL. It consists of a larger anechoic chamber, a larger array, and capabilities beyond those inherent in Boeing's TGL. It can simulate four simultaneous targets and it has an ECM array embedded within the normal array which generates standoff jamming signals. It has a large computer capability and a hydraulic table which can handle missile up to and including the latest high performance missiles. The RF generation equipment is much more sophisticated than the TGL.

The RFSS is located in building 5400, in the Francis J. McMorrow missile laboratory at Redstone Arsenal, Huntsville, Alabama as shown in Figure 5.0-2.

5.1 RFSS Characteristics

Figure 5.1-1 shows how the RFSS array components are mounted on a steel hemispherical surface. Since there are four simultaneous targets there are four independent RF paths for the targets, and since it also employs polarization diversity, each path is again doubled for a total of 8 independent RF signal paths. There is a total of 12,000 RF components on the back of the array, 534 antennas, 16 ECM antennas. Table 5.1-1 gives the characteristics of the system. The target accuracy is an order of magnitude better than that of the TGL. It has the capability of four simultaneous targets that can each be updated in one microsecond. The frequency coverage extends up to 18 GHz. The field of view is a 45° solid cone. Polarization is a significant advancement over the TGL. It

24-16

TARGET ACCURACY	5 MILLIRADIANS
TARGETS	2
UPDATE RATE	2 MILLISECONDS
TARGET VELOCITIES	0 TO MACH 50
TARGET MANEUVERING	0 TO 5,000 G's
TARGET SIZES	MISSILES TO BOMBERS
ECM	DENIAL, DECEPTIVE
FREQUENCY COVERAGE	2 TO 12 GHz
FIELD OF VIEW	30 X 30 DEGREES
POLARIZATION	LINEAR FIXED

TABLE 4.1-1 TGL ARRAY CHARACTERISTICS

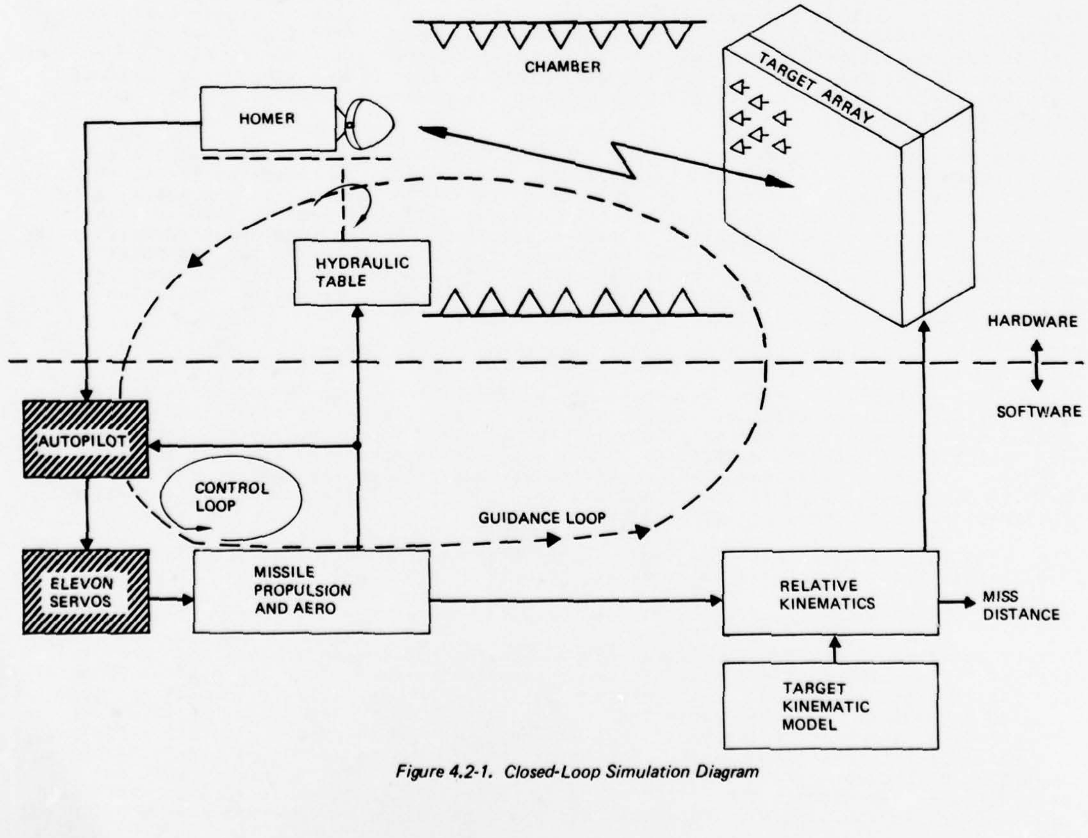


Figure 4.2-1. Closed-Loop Simulation Diagram

Figure 4.3-1. Homer Hardware in Simulation

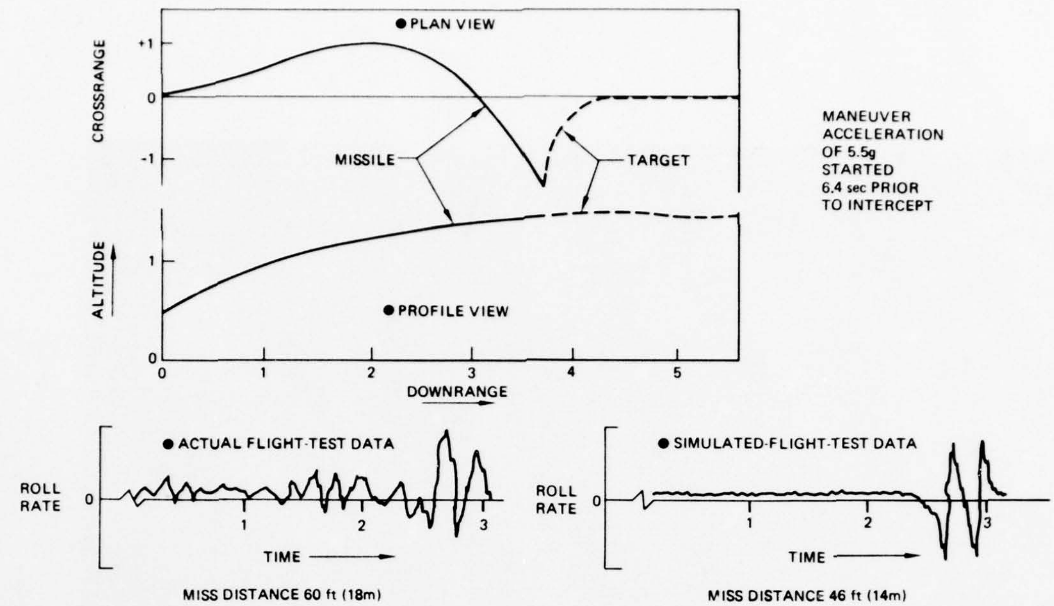


Figure 4.3-2. Simulation Verification—High-G Escape Maneuver

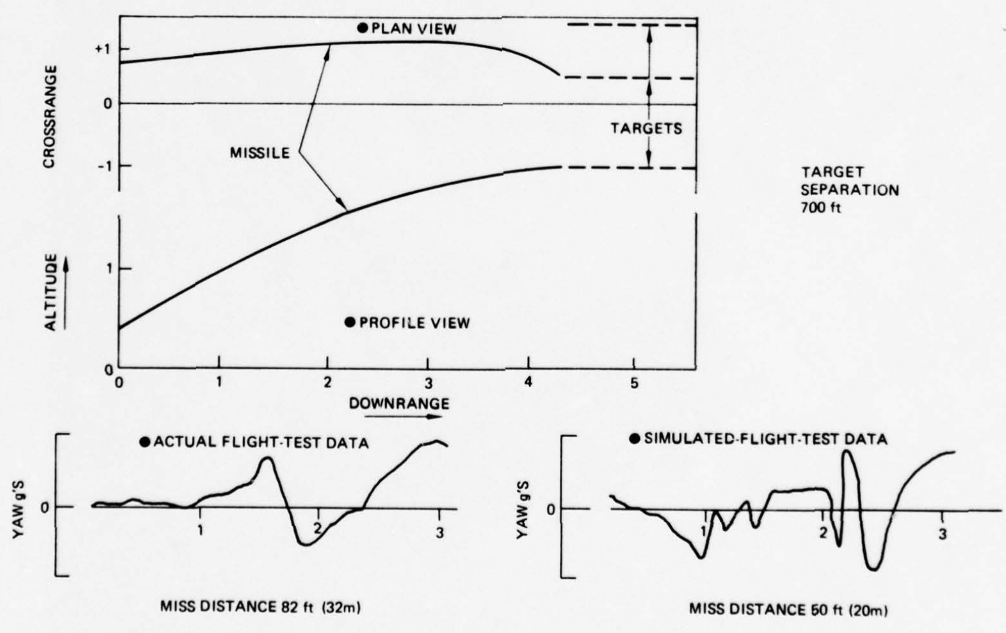


Figure 4.3-3. Simulation Verification—Multiple Targets

of the signal return from the transponder to the homer as shown in the figure. In addition, the ability of the homer to track the target was measured as a function of range and a model developed. The dotted line in the figure indicates that this portion of the model was based upon other experiments which measured the effective angular glint

Figure 5.0-1. Radio-Frequency Simulation System (MIRADCOM, U.S.A.)

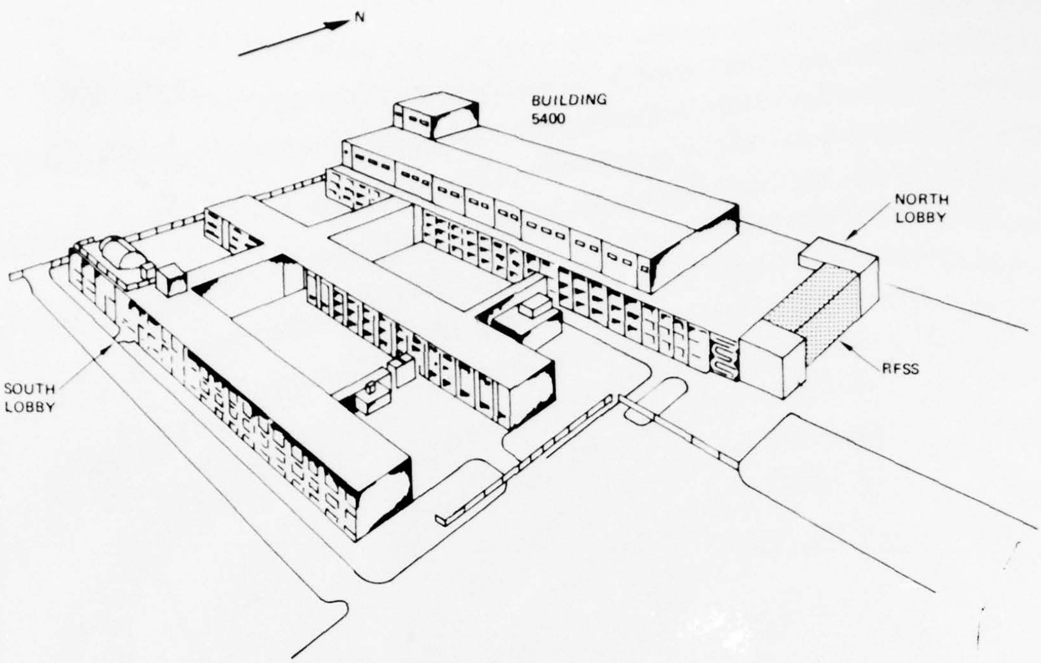


Figure 5.0-2. Francis J. McMorrow Missile Laboratory, Redstone Arsenal, Huntsville, Alabama U.S.A.

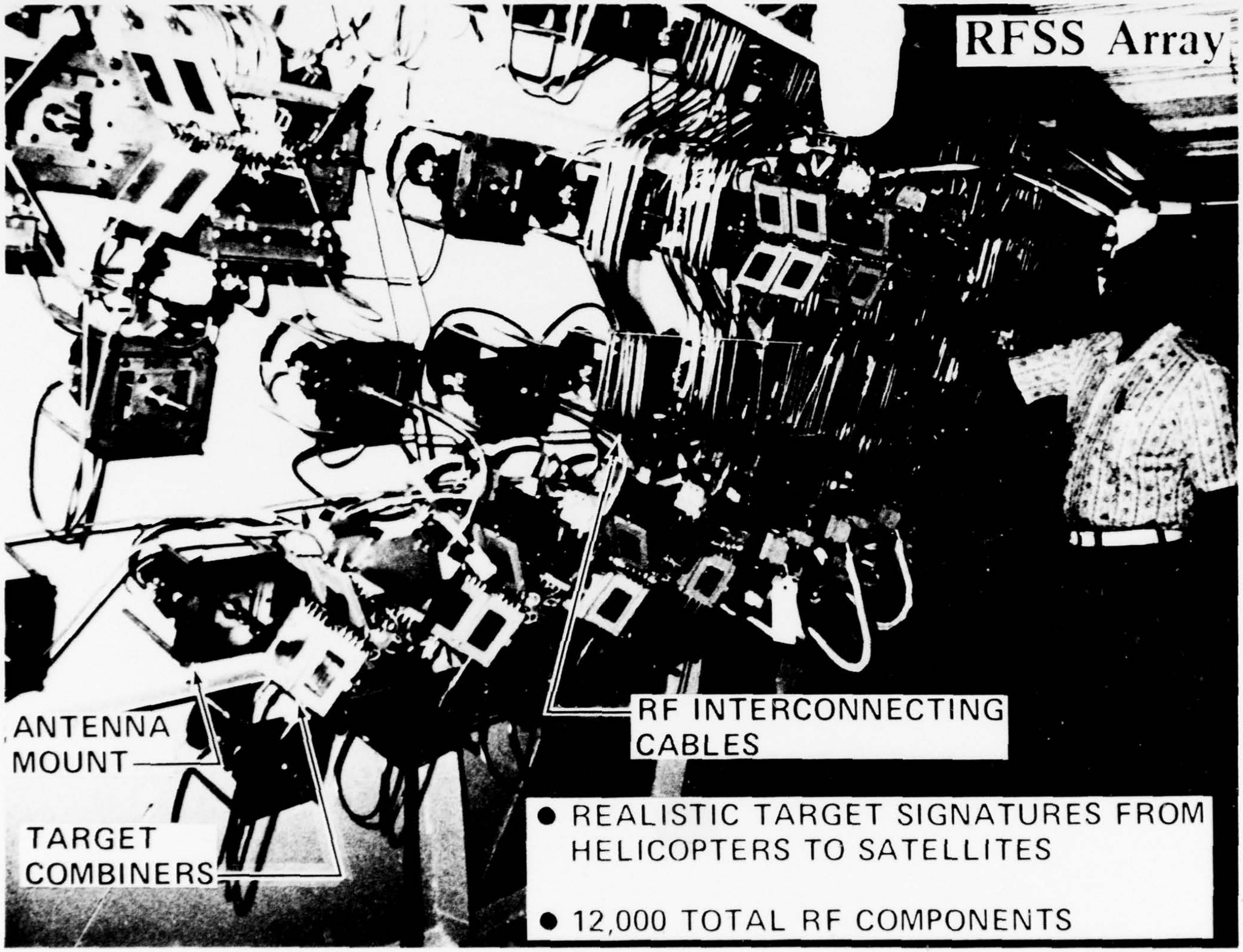


Figure 5.1-1. RFSS Array

TARGET ACCURACY	------------------------------------	0.3 MILLIRADIAN
TARGETS	------------------------------------	4
UPDATE RATE	------------------------------------	1 MILLISECOND
TARGET VELOCITIES	------------------------------------	0 TO MACH 100
TARGET MANEUVERING	------------------------------------	0 TO 5,000 G's
TARGET SIZES	------------------------------------	MISSILES TO BOMBERS
ECM	------------------------------------	DENIAL, DECEPTIVE
FREQUENCY COVERAGE	------------------------------------	2 TO 18 GHz
FIELD OF VIEW	------------------------------------	42 DEGREES
POLARIZATION	------------------------------------	LINEAR, CIRCULAR, ELLIPTICAL - PROGRAMMABLE

TABLE 5.1-1 RFSS CHARACTERISTICS

is electronically controlled so that the target polarization can be programmed or altered during the course of the simulation and therefore polarization diversity can be simulated.

The RFSS was designed to perform real time, hardware-in-the-loop simulations of active, passive, semiactive, command, beam rider and track via missile guidance in surface-to-air, air-to-air, air-to-surface or surface-to-surface modes. To date semiactive and passive missile guidance simulations have been performed and the first active seeker simulation will be accomplished during the summer and early fall of this year. In the active missile simulation, the seeker will transmit RF signals through the anechoic chamber to the array where they will be received. The RFSS RF generation system will process these signals and provide desired target signals to the array for display and retransmission down the chamber to the active seeker mounted on the three-axis flight table. The flight table provides the dynamic rotational pitch, roll and yaw forces on the seeker. The guidance loop will be closed through the autopilot and control systems modeled in the central hybrid computer complex. Open-loop simulation is an important tool and always is a precursor of closed-loop simulation. In open-loop simulation, the RF loop is closed through the target and seeker, but the guidance loop is open. A static open-loop test example is determination of AGC or doppler tracking loop response. An example of a dynamic open-loop test would be a seeker head response test with the seeker mounted on the flight table.

Since errors (associated with modeling nonlinear processes) and choices are avoided by incorporating actual seeker hardware into the simulation, it becomes extremely important that the RF environment be represented realistically as the seekers are stimulated at their operating wavelengths. The RF environment is defined to include target cross-section, glint, scintillation, ECM, clutter, multipath, and propagation anomalies. Continuous improvement in the ability of the RFSS to provide a high fidelity RF environment is a priority endeavor. Degrees of freedom required include when and where signals appear in the seeker field of view, frequency, power and polarization of the electromagnetic wave. Consideration is given to high and low resolution target modeling.

Three kinds of target models have been employed in the RFSS to the present time. The first target model is referred to as a point target or "flying sphere" model. In this model the signature is constant and appears at the center of gravity along the target trajectory. The power level is modulated only by range closure. This model is often employed during the early stages of a simulation for integration and checkout purposes and is used to establish baseline guidance performance data in a benign environment.

The second target model is called the empirical model because it is based upon measured data (static or dynamic) of a scale or full size target. The data are normally pre-processed to characterize the "bright spot wander" of the target signature and is sometimes referred to as the low frequency model. Statistical fluctuations are applied to the empirical model to incorporate fine detail or high frequency characterization of the target signature. This action results in the empirical-statistical model. For example, scintillation might be represented with a Rayleigh distribution and glint might be described with a Gaussian distribution. The simulation designer is free to characterize the fluctuations employing stochastic models of his choosing in accordance with his insight and understanding of the target behavior.

There is a strong motivation to employ deterministic models to represent RF distributed sources (time, space, frequency) such as extended targets, distributed clutter and diffuse multipath. This follows from a lack of a technical consensus concerning the statistical behavior and representation of these elements of the RF environment. Furthermore, it is very difficult for an analyst to pose a hypothesis, design and conduct an experiment to test the hypothesis, and perform the analysis to verify or modify the hypothesis without the means to probe the target characterization in a deterministic, controlled fashion. The RFSS will have completed a major step in this direction when its distributed source generator (DSGS) becomes operational late this fiscal year. The DSGS will permit implementation of a fourth type of target model, the deterministic multiple scatterer target model.

In this target model scatterers are located in target coordinates and are assigned independent gain and phase values. In real time, individual range computations are made for each scatterer and a vectoral combination is performed by range resolution cell. This complex summation is employed to control the generation of RF signals and their placement upon the target array. An extended target signal in angle and range is therefore displayed to the seeker by creating power density spectra at the seeker aperture where a realistic response will be provided by the missile hardware. By making changes to scatterer values and placement, multiple runs can be performed to develop data for statistical analysis. This approach allows an opportunity to gain new insight into the nature and behavior of a particular target. Significantly also, verification of this modeling technique enables relatively easy development of models for other target types quickly without resorting to measurement programs or overly simplified statistical models. Comparisons between empirical data and deterministic data provide an opportunity to correlate the two models.

Modeling of distributed clutter and diffuse multipath employ similar software and hardware techniques as the deterministic target model. Real world clutter and multipath create asymmetrical power density spectra which do not conform to familiar probability density functions. Tailored deterministic modeling techniques to provide control over the time, spatial and frequency characterization of these distributed phenomena such as those

Unit 4

...control interface which is driven from Unit 1 providing

24-24 described in the extended target model can provide realistic seeker performance evaluation. This approach is more desirable than the sampling approach whereby one or two "tones" or lines are generated and modulated in frequency and amplitude to determine the seeker response. The distributed source modeling approach is virtually required for high resolution seekers.

Simulation in an electronic warfare environment has proven to be an important application of the RFSS. A number of different jammers have been incorporated into the simulations performed to date. These have included denial and deception jammers under development and in operation. Use of real jammer hardware functioning in a realistic, dynamic environment against real seeker hardware has added a new degree of credibility and utility to hardware-in-the-loop simulations. Integration of the jammers into the simulation is easily done. ECM techniques are quickly developed and optimized; operational techniques are quickly verified or modified. ECCM needs are identified, defined, implemented and verified effectively on a quick turnaround basis. Statistical simulation data are collected to design ECM flight tests and serve as a guide during the flight test for interpretation purposes. Flight test results are then used to validate the simulation upon proper correlation.

Accomplishment of a simulation task occurs in five phases of activity. In sequence the phases are: coordination and planning, development, integration and checkout, simulation, and documentation and analysis. During the first phase agreement is reached on the simulation requirements and objectives resulting in definition of the costs and schedules. Thereupon development of the simulation is initiated where any required facility modifications are performed, the missile and RF environmental models are developed, software is written, missile hardware interfaces are designed and fabricated, test plans and procedures are prepared and the hardware systems are calibrated prior to delivery of the missile hardware. Integration of the missile hardware with the RFSS subsystems is then accomplished and checkout proceeds to verify everything is operating properly. At this point the simulation is actually performed and data are collected. Finally the results and data are processed, analyzed and documented in a simulation report. The level of effort and time required to perform the development phase can vary greatly depending upon a number of factors. Initial development of the missile models and the missile hardware interfaces is a complex and time consuming effort. Depending upon the simulation requirements, modifications to the facility can be involved. Development of new RF environmental models can be an extensive task. Reactivation of a simulation for additional test entries is usually characterized by a greatly reduced development effort and schedule. The duration of the simulation phase is determined by the scope of the requirements as defined in the test matrix.

5.2 Testing in RFSS

Since it was dedicated in November 1975 there have been extensive missile evaluations performed. Some of the tests are shown in Table 5.2-1. Figure 5.2-1 shows a missile in test. The RFSS has been used to evaluate the HAWK, Standard Arm, advanced sensors, Navy systems, foreign systems. In all, at least 8,000 simulations have been performed with a cost advantage of at least 1,000 to 1.

It is possible with this system to use the flight test to validate the simulation rather than vice versa. This system can be used to explore all the performance capability of the system and then use the actual flight tests to spot check the simulation results.

6.0 NRL'S CENTRAL TARGET SIMULATOR (CTS)

Figure 6.0-1 shows an artist's view of the Central Target Simulator being developed by the Naval Research Lab in Washington, D.C. This laboratory will be used to evaluate EW systems effectiveness in defending carriers against enemy missiles. The simulator generates carrier radar crosssection and EW signals, radiates these signals to a missile homer and in a closed-loop real-time situation evaluates the miss distance for various EW techniques.

6.1 CTS Characteristics

This system has a thousand antennas, and radiates signals over the 8 to 18 GHz band. One of the most significant capabilities of this system is the ability to generate high power. Several orders of magnitude higher power are being generated in this system to simulate the effects of the much larger target radar crosssection and electronic warfare signal powers.

Table 6.1-1 summarizes the general characteristics. The target accuracy is 1 milliradian which is compatible with the state-of-the-art for generating these high powers. The number of targets are 28 including ECM. Update rates of as fast as 1.5 microseconds are provided. Notice that target sizes show a significant difference over previous systems.

6.2 Tests in CTS

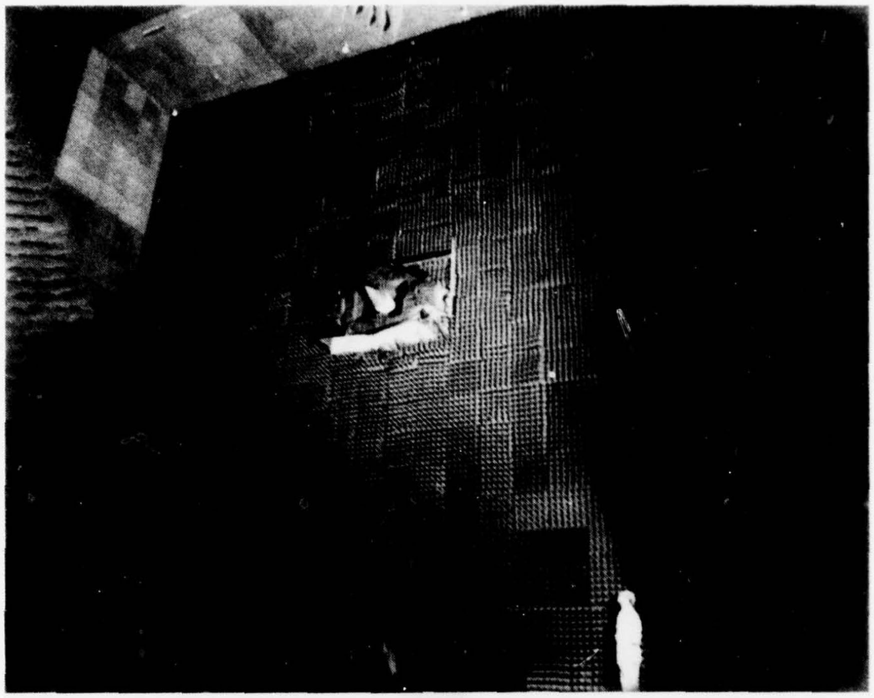
The Navy plans to bring their system on-line in the 1978-1980 time period. Initial testing will be done on high priority EW systems.

Question 1. Did you find operation of these push buttons Difficult? Acceptable? or Easy?

DATES	SIMULATION	NUMBER
NOV. 1975	IMPROVED HAWK DEMONSTRATION	100
JAN. 1976	STANDARD ARM/ARM DECOY	200
MAR. 1976	IBM ADVANCED SENSOR	150
JUNE 1976	IMPROVED HAWK NAVY/AFWL	350
OCT. 1976	IMPROVED HAWK ECM-1	50
OCT. 1976	TRI-FAST VERIFICATION	—
NOV. 1976	GROUP WORK	2,000
FEB. 1977	IMPROVED HAWK ECM-2	1,000
MAR. 1977	BASIC HAWK	1,500
MAY 1977	WEAPON SYSTEM DEVELOPMENT	2,500
AUG. 1977	IMPROVED HAWK ECM-3	700
SEPT. 1977	IMPROVED HAWK-AIR FORCE	130
	PATRIOT ARM MISSILE TESTS TRI-FAST GENERIC ARM	8,680

TABLE 5.2-1 MISSILE TESTS IN RFSS

5. Simulation of failures has not been as successful as hoped for. The main



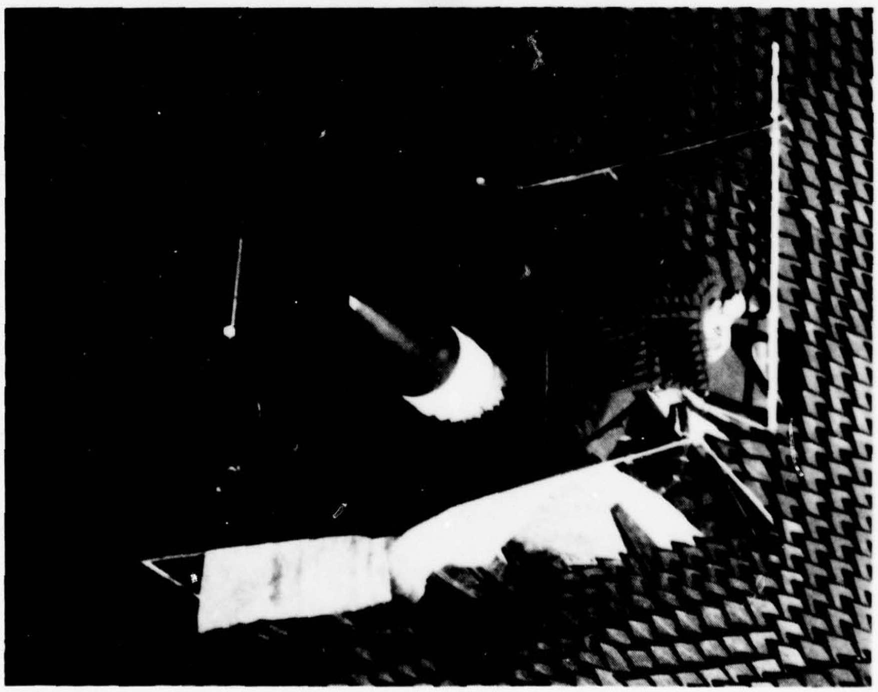


Figure 5.2-1. Missile Test in RFSS

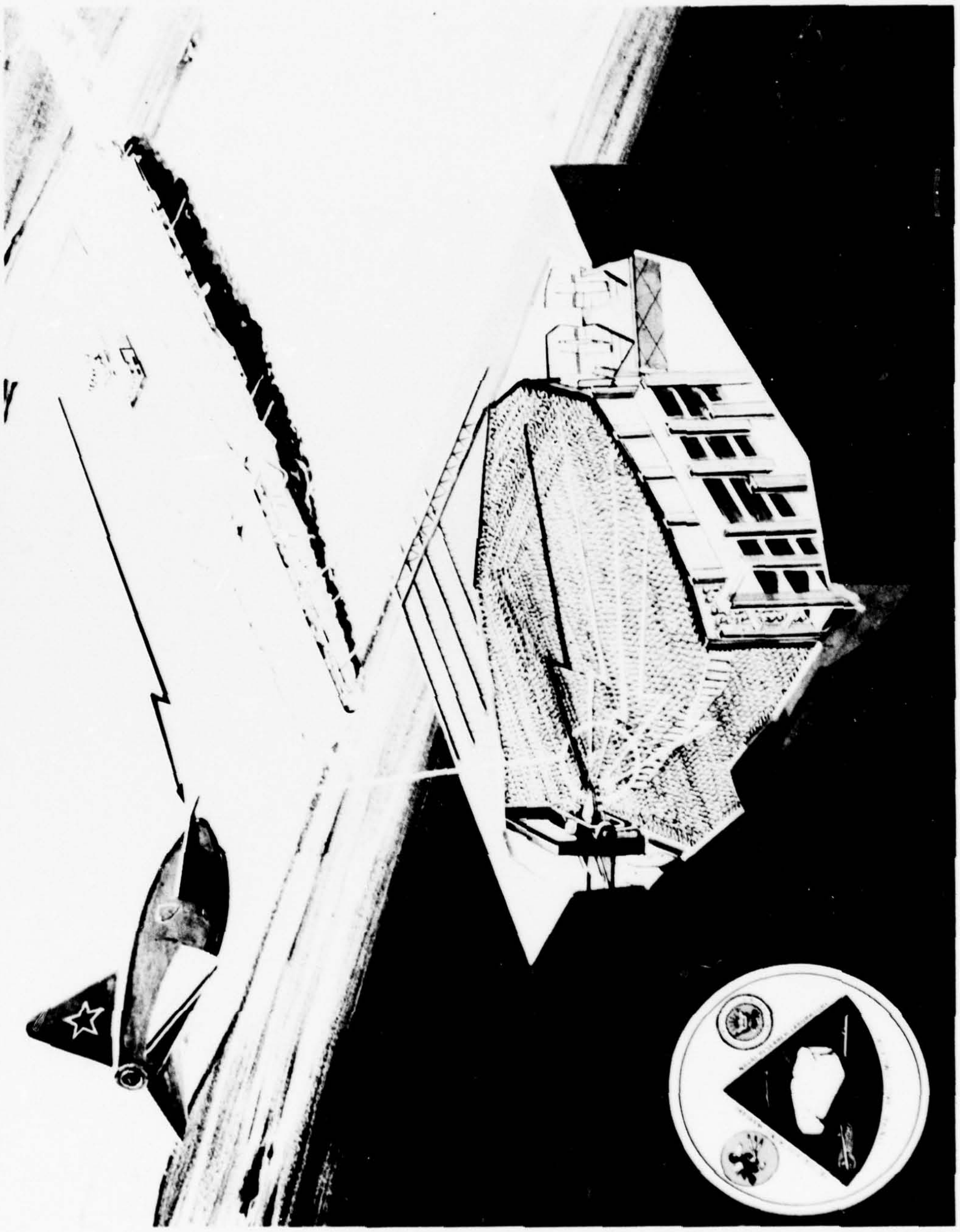


Figure 6.Q-1. Central Target Simulator (CTS) (Naval Research Laboratory)

TARGET ACCURACY	1 MILLIRADIAN
TARGETS	28 INCLUDING ECM
UPDATE RATE	1.75 MILLISECONDS
TARGET VELOCITIES	0 TO MACH 50
TARGET MANEUVERING	0 TO 5,000 G's
TARGET SIZES	MISSILES TO CARRIERS
ECM	DENIAL, DECEPTIVE, CHAFF
FREQUENCY COVERAGE	8 TO 18 GHz
FIELD OF VIEW	80 DEGREES AZIMUTH, 20 DEGREES ELEVATION
POLARIZATIONS	HORIZONTAL, VERTICAL, CIRCULAR

TABLE 6.1-1 CTS CHARACTERISTICS

7.0 Boeing's Millimeter Wave Engagement Simulator (MWES)

Figure 7.0-1 shows an artist conception of the simulator that is being developed by Boeing to operate in the millimeter region of the RF spectrum. The primary intent is to evaluate missiles homing against tanks. An electronically controllable array broadcasts signals to an actual missile under test. Instead of an anechoic chamber, however, a metal lined room will be used to reflect all the background radiation to a sloping front, and thence to the sky as a heat sink. The missile will be mounted in the center of the cold spot and will be looking at the array at the other end of the chamber. The feasibility model of this system was built by Boeing in 1976. Some aircraft tests were made with a radiometer looking at land/water areas in the Puget Sound area. The radiometer was then placed in the simulator and the array programmed to simulate the same flight. A comparison of the data is shown in Figure 7.1-1. Excellent agreement was obtained thus proving the feasibility of the concept. During 1978 Boeing will construct a full scale capital facility with the characteristics shown in Table 7.1-1. The target accuracy again will be in the order of 1 milliradian with up to 100 targets. The array itself has an update rate of 150 microseconds. The targets encompass trucks, tanks, lakes and deceptive ECM. The frequency coverage is from 18 to 300 GHz, and the array will provide a 30 degree field of view. In addition to the normal target signatures, environments such as smoke, haze, fog, rain and dust can be programmed from the array to simulate actual battlefield conditions.

8.0 SUMMARY

This paper has discussed the development of RF homing missile guidance and controls simulations starting from the early 1940's up to the present date covering the frequency range from 2 to 300 GHz. This is an area that is expanding in importance and with the high cost of missile flight tests is proving to be a very cost effective method to develop new missile systems. The military agencies are recognizing these facts and are posturing future missile programs to utilize hardware-in-the-loop real-time closed-loop simulations in place of missile flight tests to provide performance data over a very wide spectrum of battle scenarios.

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RFSS	Brochure Maurice Belrose (205)876-2592	
CTS	Brochure Gene Erickson (202)767-2208	
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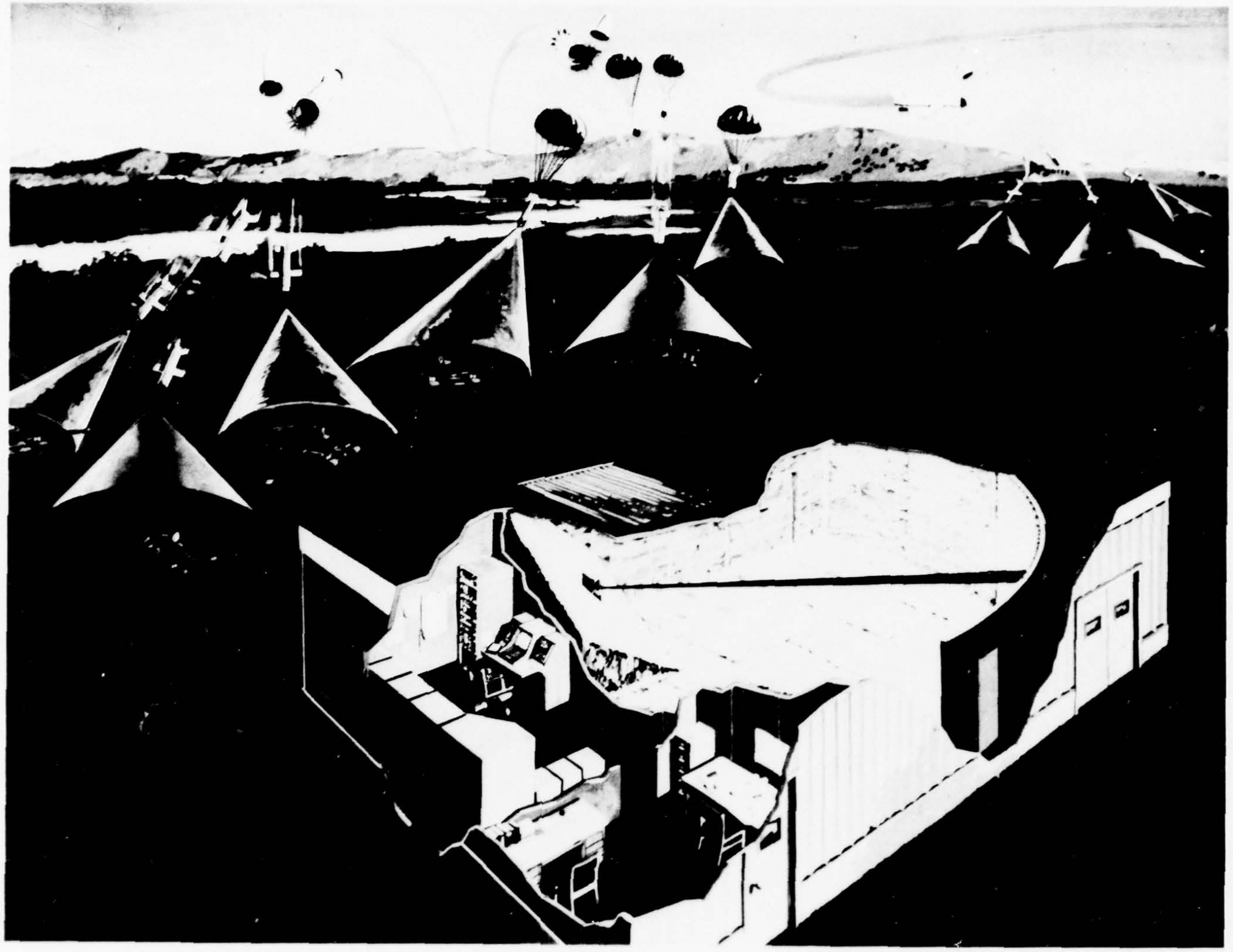

Figure 7.0-1. Millimeter Wave Engagement Simulator (MWES) (Boeing Aerospace Company)

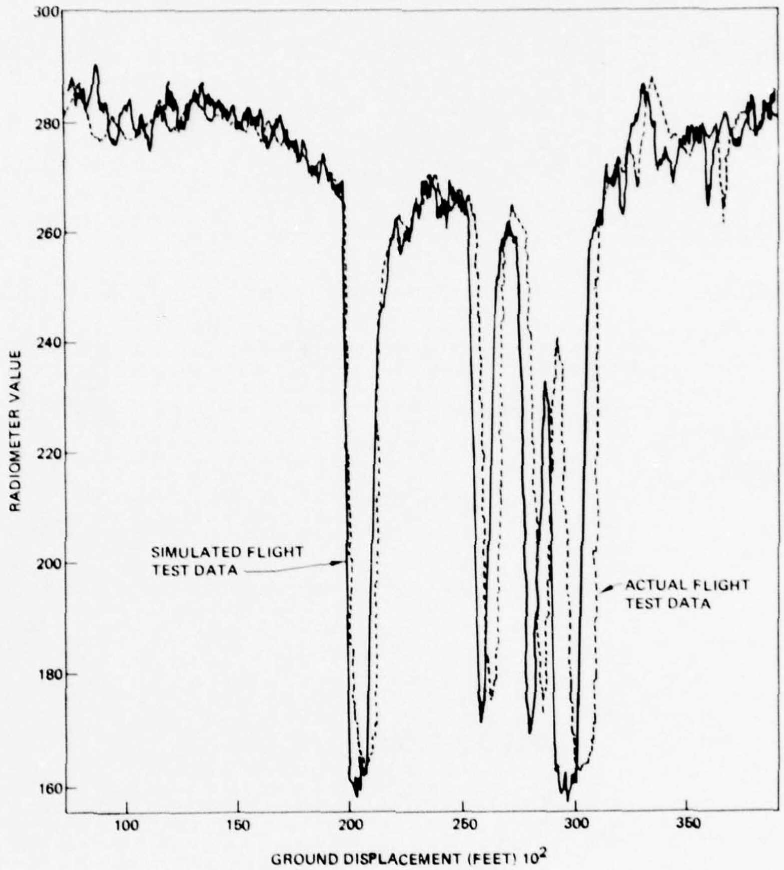

Figure 7.1-1. MWES Simulation Verification

TARGET ACCURACY	----------------------------------	1 MILLIRADIAN
TARGETS	----------------------------------	100
UPDATE RATE	----------------------------------	150 MICROSECONDS
TARGET VELOCITIES	----------------------------------	0 TO MACH 10
TARGET MANEUVERING	----------------------------------	0 TO 5 G's
TARGET SIZES	----------------------------------	MORTAR, TANKS, LAKES
ECM	----------------------------------	DECEPTIVE
FREQUENCY COVERAGE	----------------------------------	18 TO 300 GHz
FIELD OF VIEW	----------------------------------	30 DEGREES
POLARIZATION	----------------------------------	RANDOM
ENVIRONMENTS	----------------------------------	SMOKE, HAZE, FOG, RAIN, DUST, ETC.

TABLE 7.1-1 MWES CHARACTERISTICS

MISSION SIMULATION AS AN AID TO DISPLAY ASSESSMENT.

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Summary

Advances in computer and display technology have created a situation where drastic changes in aircraft cockpit layout are possible.

In recent years investigations have been undertaken at B.Ae. at Warton involving full mission simulation in an advanced cockpit environment. This paper discusses the philosophies and methods adopted and the hardware required for such simulation and also indicates areas where problems have been encountered.

Introduction

In recent years considerable advances have been made in the miniaturisation of computers and have led to a situation where a large amount of information could be made available to the pilot, firstly if it is shown to be useful to the pilot, and secondly, if space can be provided to display such extra information. The provision of space for this extra information is a major problem as the physical size of the cockpit instrumentation panels, in the new generation of high speed military combat aircraft is tending to be reduced rather than increased. This reduction in panel size is particularly evident in aircraft which have reclining seats (Fig. 1). This results in a need to re-assess the information traditionally displayed, compare its value with information which could now be made vailable, and consider in what way the traditional cockpit concept can be modified to accommodate the extra information in the reduced area. The ground based research simulator provides an excellent tool for such investigations. Firstly, the simulator enables examination of potentially dangerous flight conditions in the total safety of the simulator and, secondly, it allows the use of general purpose digital computers, which offer great flexibility, and also the use of hardware of a far less stringent standard than that required for inflight testing. The latter results in considerable cost savings. We will describe the manner in which we have approached this form of simulation at British Aerospace Warton.

Requirements of a Simulation for the purpose of Displays Assessment

In order to enable a successful assessment to be made the simulation must perform certain functions. Basically it must create a representative environment whether it be a low-level attack situation or a high altitude combat engagement. In order to achieve this it is necessary to have:-

- Handling and performance characteristics appropriate to the type of aircraft under consideration.

- Realistic cockpit geometry.

and a resulting realistic level of pilot work load.

Having obtained a representive environment, provision must be made for examination of the displays with regard to the following:-

- Physical positions of the displays within the cockpit.

- Ease of interaction between the pilot and the display.

- Display content.

- Method of presentation.

- Interaction between displays

and the effect of a display on the pilots performance.

Philosophy Adopted

As we are concerned with basic research we are in many instances taking a first look at new concepts, some will show promise others will not. It is therefore essential to approach such assessments in as flexible a manner as possible, being prepared to change both the direction of the investigations and/or the types of display under assessment, as a result of initial testing. With this in mind it is desirable to use high speed general purpose, digital computers wherever practical, and provide software in a high level language whenever changes are thought likely. This enables software modification with minimum delays.

While wishing to have flexibility we must inevitably impose some restrictions upon ourselves. It can be tempting to construct a completely new cockpit totally reliant on electronic displays but this bold step may well lead to a situation where, cause and effect can no longer be readily identified because of the numerous possibilities which are present. A further possibility is that pilots become overwhelmed by the number of new concepts confronting them, and pilot alienation may result. It appears far more reasonable to consider a step by step approach by which pilots are introduced to the change from electromechanical instruments to totally electronic displays in certain areas of the cockpit, while retaining traditional instruments in the remainder. Typically this would result in basic flying instruments being unchanged, while systems information for engines/fuel/navigation, and weapons are presented electronically, by various methods.

Having imposed this restriction it would appear prudent to restrict the dimensions of the new cockpit to that of an existing airframe so that the natural progression from simulation to full inflight testing can be achieved more readily.

The choice of suitable subjects is important. Ideally we require pilots who are currently flying the particular aircraft chosen for the simulation and who are proficient in the operation of any actual aircraft hardware, such as head up displays, or navigation systems which are included. Failure to obtain such subjects leads to excessively long familiarisation periods being necessary prior to undertaking the intended display assessments. Flying clothing is also important. This is generally cumbersome but must be worn if a realistic environment is to be recreated, and if assessments of display management are to be meaningful. Possibly the most important factor in relation to the subjects is that the simulation engineer must develop a successful rapport between himself and the subjects, so as to encourage the subject to become actively involved in the experiment, rather than merely performing set tasks to order. Comment, criticism and suggestions should be solicited at all times and carefully constructed questionnaires can provide an ideal basis for this. It is particularly convenient to have two subjects available at the same time since this allows each pilot to rest between missions without loss of simulator utilisation. Subjects should be prevented from forming preconceptions about the experiment by restricting the amount of information available before the official briefing to a minimum.

Presentation of the briefing material in written form ensures as far as possible that each pilot receives an identical brief. During any verbal briefing it is important that the engineer involved avoids any temptation to express his own opinions on topics intended for assessment.

Associated with the brief is a period of familiarisation with both the simulation and the displays. Half a day could be sufficient in some cases although this time must be extended if required in order that the effect of the learning process on experimental results is minimised.

Typically a program for display assessments could be made up of the following phases:

1. Initial definition during which time types of display, sizes of display, display content and scenarios are established. (6 months).

2. A Build program during which the integrated simulation is constructed (9 months)

3. A series of Shakedown Experiments using test pilots from both the company and the armed forces (2 months).

4. A modification period during which shortcomings identified in the shakedown experiments are eliminated (2 months).

5. A formal experiment using front line Squadron pilots. (3 months).

6. A reporting and presentation phase (2 months).

Simulation Techniques for a Particular Application

Having established our requirements and the philosophy with which to meet them we are now in a position to consider a detailed breakdown of the simulation. Let us assume the following scenario which might apply to any typical low-level strike mission Fig 2.

- Starting conditions - over base 450 knots/500 feet.
- Follow H.U.D. director to the first waypoint which is still within friendly territory.
- Check position as required on the navigation displays.
- Check fuel state and engine parameters on appropriate displays.
- Approaching first waypoint - check navigation system for drift and update with correct position data if required.
- Confirm satisfactory timing and proceed to next waypoint accelerating to 550 knots and settling to 200 feet as the FEBA is crossed.
- The waypoint is a target, so arm the weapons system and select a suitable weapon delivery display on the HUD.
- Deliver the weapons and make the remaining weapons safe.
- Carry out attacks on the two subsequent waypoints using appropriate weapon delivery HUD modes.

Having completed the hostile phase of the mission:

- Return to friendly territory crossing the FEBA at the correct position and time.
- Decellerate to 450 knots and take a navigation fix on the next waypoint.
- Update the system if necessary.
- Return to base after making the weapons system safe.
- Mission ends over base.

Probably the largest division which has to be resolved is that between hardware and software. It has already been mentioned that flexibility must be maintained in certain areas and this implies software. Indeed in our experience, it requires high level software written in a language such as FORTRAN if rapid and reliable changes are to be effected during the development phase. It is equally as important to avoid a software overhead which could be resolved with hardware whilst taking into account any software/hardware interfacing which may be required. Based on the above mission definition, the tasks allocated to software could therefore consist of:

1. Head down display generation
2. The aircraft simulation including engine performance and fuel consumption.
3. Navigation calculations.
4. Weapons system computations.

The remaining requirements listed below are largely met by dedicated hardware.

1. Cockpit instrumentation.
2. Outside world display
3. Head-up display symbology generation.

It is clear that the software listed above could pose problems of compatibility such as update rate and C.P.U. requirements if an attempt was made to run it in a single large processor. A more satisfactory solution is to employ more than one small processor which can be allocated each to its specific task. Sub division of software in this way also has advantages in program development and testing aspects.

A final point to consider when optimising software distribution is the quantity of data transfer between processors associated with each configuration.

Let us now consider each of the software tasks in turn:

Head Down Display Generation This could be one of the following.

1. System Management displays - these will be manually selectable and can vary in form from a simple system status presentation to a detailed subsystem analysis of the form necessary in the event of a systems malfunction. These displays are likely to be both alphanumeric and graphical in form.

2. Navigational displays - these can range from simple stylised moving map displays to complex mission overviews overlayed with current status related to points within the mission profile.

To exercise subselections of the system management displays it is necessary to include software for the simulation of non-catastrophic malfunctions during the mission.

Aircraft Simulation

A re-appraisal of the mission definition implies the requirement for an aircraft simulation covering a considerable restricted flight envelope. Also the limited aircraft manoeuvres encountered in such a mission as this allow the use of a simplified set of equations of motion. Both these considerations alleviate to a large degree the software effort needed to achieve an adequate aircraft representation.

Navigation Calculations and Weapon System Computation

These are closely allied and generally favour inclusion within the same processor. Calculations performed here have outlets in a number of other areas notably:

- Navigation display and weapons management displays which have already been mentioned.
- The head up display in both navigation and weapon delivery modes.
- The outside world display for feature representation on the ground.

Facilities can be included within this software for simulating drift within the navigation system and its removal by manual updates.

A typical hardware committment to meet these requirements is shown in fig 3. The items can be grouped into four main units which are detailed as follows:-

Unit 1

PDP15/76 Unichannel min-computer with
a) 32K words core, 20 A/D converters and 32 D/A converters.
b) PDP11/05 with 2 disk drives and 8K local memory.
c) VP15 Storage display and interface.

The PDP11/05 shares memory with the PDP15 and performs peripheral processing thus freeing the PDP15 for user computations which are specifically navigation and weapon aiming calculations as well as parameter calculations for supplying two displays generated elsewhere. The Storage display is used for plotting mission parameters e.g. fuel usage as a function of time into the mission for comparison on the same set of axes with the predicted parameter behaviour. The display is directed to a scan converter for conversion from a cursive signal to a raster video signal suitable for introduction into the video mixer/switcher.

Unit 2

This consists of identical hardware to Unit 1 but its main function is the simulation of the aircraft equations of motion and generation of analogue signals for driving cockpit flying instrumentation. Interprocessor communication is provided by a digital link between units 1 and 2. The PDP15 also produces analogue outputs suitable for driving the outside world display. This display is a self-contained computer generated, six degrees of freedom synthetic terrain generator which produces a display shown typically in fig 4 and consists of a matrix of 200 feet square coloured fields. At appropriate times, conspicuously coloured squares are introduced onto the terrain to coincide with targets or waypoint emergence as required. The synthetic terrain is presented to the pilot via a suitably collimated colour monitor situated in the cockpit.

Unit 3

PDP-9 mini-computer system with 16K words of memory and 24 A/D converters. This machine is interfaced to a cursive refresh display system and is used for drawing the stylised moving map (fig 5) as well as all other dynamically updated displays as and when required. This cursive output is also converted to a raster signal via a scan converter and fed to the mixer/switcher.

techniques as the deterministic target model. Real world clutter and multipath create asymmetrical power density spectra which do not conform to familiar probability density functions. Tailored deterministic modeling techniques to provide control over the time, spatial and frequency characterization of these distributed phenomena such as those

Unit 4

Unit 4 is the sense/control interface which is driven from Unit 1 providing digital interfacing throughout the system as follows:-

a. Remote HUD mode selection as required e.g. for automatic switching from "general navigation" mode to "close navigation" mode as a waypoint is approached. This type of switching descrete is required since the HUD is an aircraft unit which has to be driven via its airborne interface.

b. All cockpit switch and button selections including weapons selections, weapon releases, pilot display mode selection, switching sequences for failure rectification, navigation control unit selections and manual head-up display selections.

c. Remote video signal switching as required for display on the cockpit head-down monitor. Switching is required since video signals from more than one source were available for this monitor.

The remainder of the equipment consists largely of laboratory monitors to enable the personnel conducting the experiment to remain informed as to the status of the mission and aircraft. These displays are repeaters of the cockpit head-down display, the outside world display and the head-up display.

In the particular experiment described here, two cockpits were available one of which had a conventional cockpit fit typical of a current fighter aircraft while the other was fitted with the displays layout described previously. Either could be linked to the computer system enabling piloting assessment to be made of a conventional cockpit vs an advanced displays cockpit in quick succession. This was a very desirable situation.

Data Acquisition

In this type of experiment there are two main sources of measurement: numerical data and pilot opinion. With the advent of the digital computer, numerical data has become easy to obtain and easy to submit to detailed numerical analysis techniques. This has had an unfortunate effect in that before the digital computer era, recording was expensive and engineers were severely restricted in the number of variables which could be recorded. This led to very careful thought as to the most important quantities to store. Nowadays it is all too easy to store vast quantities of results from which, unfortunately in many instances, very few if any conclusions can be drawn. In our experience the types of numerical data worth recording fall into two categories. Firstly, data which gives a clear cut measure of pilot performance under comparative conditions - this might be the accuracy of weapon delivery with differing weapons displays and control laws, or the navigational accuracy with similarly varying display aids. The second form of data is that which relates pilot selections and operations to other mission events on a time basis. Typically this involves monitoring time keeping in relation to estimated times of arrival at waypoints and target areas and also checking whether or not system selections are correctly made at various stages of the mission, since it is futile carrying out an excellent attack profile if you forget to select the LATE ARM switch to ON. A typical data set of this form is shown in (fig 6).

Pilot opinion may well be the best source of information in these types of assessment but the collection of such opinion requires careful planning. In our experience questionnaire methods have proven successful and it is appropriate to explain our approach to using such techniques.

In order to obtain a comprehensive questionnaire we first identify those areas of the cockpit displays on which we wish to concentrate and allocate a separate section within the questionnaire to each topic. Within each subdivision detailed questions are constructed. Care is taken to ensure that questions are non ambiguous, do not tent to bias the answer, and are presented in a logical progression. For example do not ask whether a display is "noisy", as you may get an answer relating to loudness or the level of interference, also, do not ask for comments on display conetent before you have established whether or not the pilot can see the display. A further essential is the provision of space, adjacent to each question, for the pilot to add comments. Having constructed the questionnaire it should be used during the shakedown experiment so that possible trouble areas can be identified and eliminated.

In this type of assessment it is advantageous to present the questionnaire during the initial briefing for the experiment, as this allows the subject to identify the topics on which the questionnaire is based. It is preferable to allow the subject to complete the questionnaire unaccompanied and then have a debriefing so that the subject can clarify any queries relating to the questions, and the engineer can ensure that he understands the answers. The following examples demonstrate the need for such debriefing.

The Navy plans to bring their system on-line in the 1978-1980 time period. Initial testing will be done on high priority EW systems.

Question 1. Did you find operation of these push buttons Difficult? Acceptable? or Easy?

Answer 1. Difficult

Comment "Being small I had difficulty"

Is this a comment on pilot stature or button size?

Question 2. Was the size of the display Too Large? About Right? or Too Small?

Answer 2 Too Large.

Additional question. How large should the display be?

Additional answer. As large as possible ! ! !

One final important apsect of presenting a questionnaire is to guarantee secrecy of the subject's identity when presenting results, an assurance which in many circumstances allows more freedom of expression to the participants.

<u>Discussion of Results</u>

1. Simulations of the form described have enabled useful assessments to be made of new display concepts, display contents, display management and cockpit layouts.

2. Where comparison of two standards of cockpit is required, it is desirable to have both types of cockpit available at the same time. This may appear extravagant but the advantages are considerable. If one cockpit is to be modified to two different equipment fits, it implies an interval between assessments of the two cockpits of possibly several months. This is undesirable on two counts. First, it is unreasonable to expect a pilot to make a comparison between simulated flights under varying conditions which take place months apart. Second, it is our experience that from a programme management aspect it is easier to obtain subjects for a single period of several consecutive days than to obtain a given subject on a series of days at large intervals. With two cockpits the same mission can be flown in both types of cockpits with a delay determined only by the need for the pilot to rest between "flights", and the questionnaire can be completed while the detail of each option is fresh in the pilots mind.

3. Decisions on the inclusion of actual aircraft hardware must be made with care. In one experiment, we included two main items: the Navigation Control Unit, and the Head Up Display, and decided to replace the Moving Map Display by a simulated map. The Head Up Display proved to be totally satisfactory while the remaining two items presented problems of a differing nature. The Navigation Control Unit, while totally without criticism from the pilot, proved expensive to interface both from complexity of computation and manpower required to carry out the interface. In the event a much simpler stylised simulated unit could have been substituted (and will be in future work). The reverse type of decision was taken in relation to the provision of a Moving Map. A simulated moving map, electronically generated on a raster monitor was substituted for the actual aircraft hardware. The degree of complexity of information provided on the simulated map was restricted by computer speed, core size and lack of a small colour monitor. The lack of detail resulted in unanimous criticism by all participating pilots, and therefore for future investigations we have to consider whether to install the actual hardware-with considerable interface problems - or invest in a complex electronic colour graphics system to create a realistic map.

4. The use of a synthetic electronically generated outside world proved acceptable for this type of simulation, particularly as the mission was under flight director control.

Comment was received to the effect that the display had some shortcomings, in particular:

- flat fields - resulting in an easier height control task than in real life.

- consistently good weather conditions.

and very obvious targets and waypoints.

The first of these comments relating to height keeping is undoubtedly true but the pilots task was to some extent increased by the fact that flight director error signal was recorded as a measure of flying accuracy. Weather conditions could be varied but only in terms of cloud base and visibility and these once set were constant for a mission. The third point - very obvious targets - was intentional, since we did not wish to get involved in the arguments relating to target recognition which we have previously experienced.

5. Simulation of failures has not been as successful as hoped for. The main problem here is one of creating the 'life' and death' atmosphere of the inflight situation, in a fixed base simulator. In general our policy has been one of inserting relatively simple failures which, while not jeopardising the mission, have necessitated the use of the displays under assessment. The failures have been inserted at points in the mission having different workload levels. It has been found that while in conditions of low workload, the pilot would carry out the correct procedures. In conditions of high work load, the pilots, perhaps aware of the safety of the simulator, would leave the fault until his immediate task such as weapon delivery had been completed.

To date, the answer to this problem has not been found, but as investigations move towards cockpits with totally electronic displays, on a very limited number of display surfaces, it is likely that in the event of a failure occurring, the information relating to the failure will be automatically overwritten on a particular multi moded display resulting in the loss of the previously selected information and thereby preventing the pilot from ignoring the failure - even in the simulator.

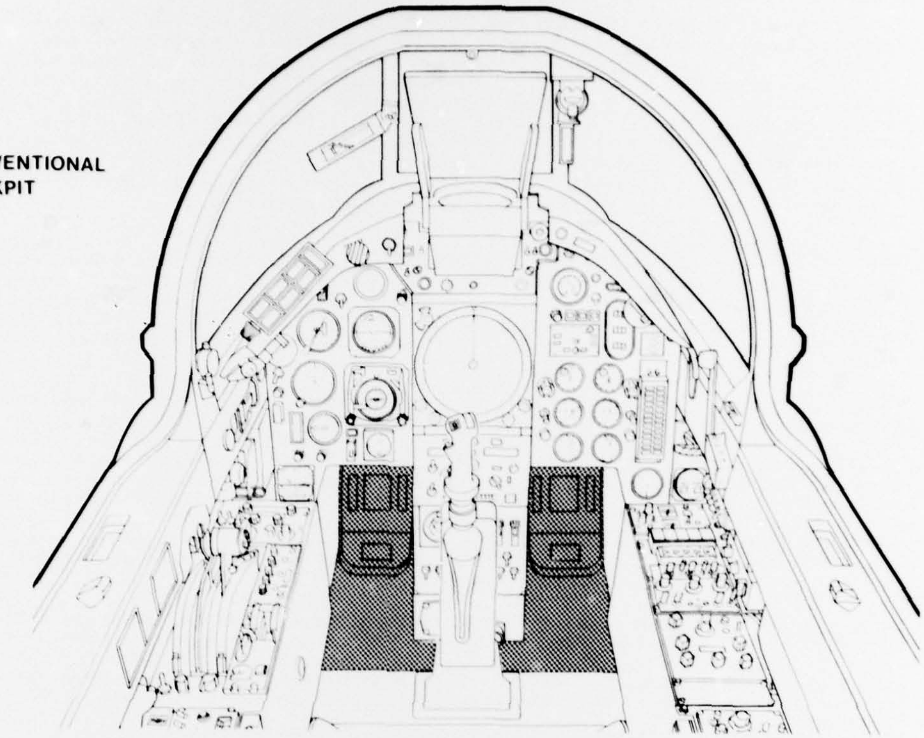
6. Light omitting diodes are becoming popular in new cockpit display concepts, LED technology having reached a state where more or less all colours are available, although in order to obtain the necessary brightness levels for some colours, overheating problems must be resolved. Given the need to simulate such a display, the construction of an L.E.D. panel can be readily undertaken. However, once fabricated, the display format cannot be readily varied. If the assessment is concerned with the display content and concept, as is our case, then it should be considered whether it is better to simulate the L.E.D. display by means of a general purpose colour c.r.t. display thereby allowing pre-storage of various formats with on-line selection capability.

7. One question inevitably asked in this form of experiment is "how does the workload content of the simulator mission compare with that of the real world in-flight situation?" It is our experience that quantitative measures are difficult to obtain since the only methods generally considered to be consistently reliable are of a bio-medical nature and are likely to alienate the subject if not the engineer.

Finally it must be emphasised that conclusions drawn from this type of simulation relate only to human factors aspects and engineering aspects require separate study.

25-8

CONVENTIONAL
COCKPIT



RECLINING SEAT
COCKPIT

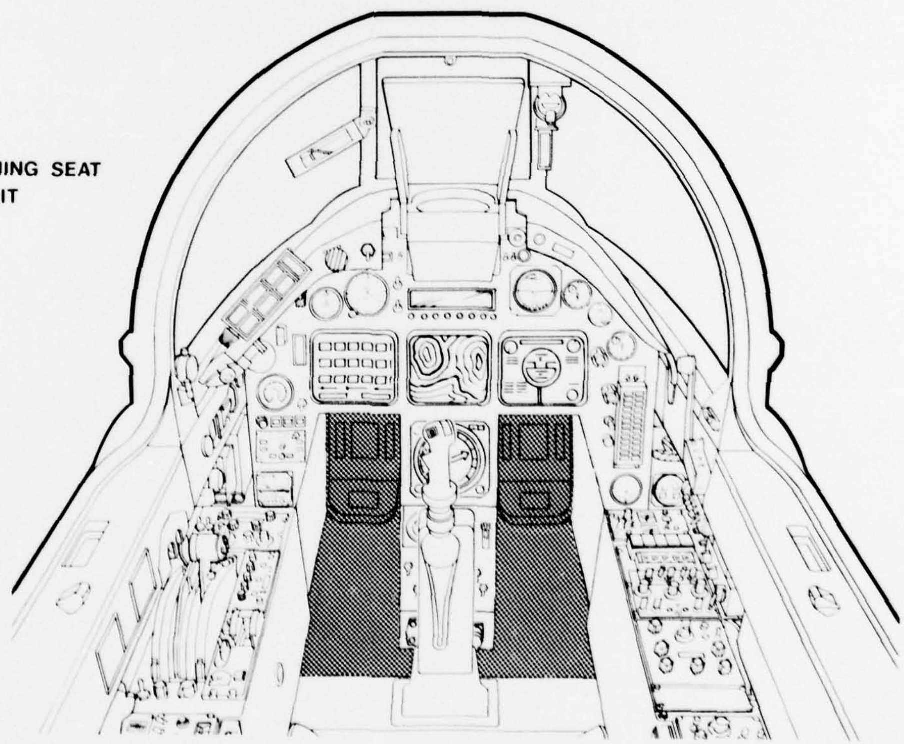


Fig. 1 COMPARISON OF COCKPIT TYPES

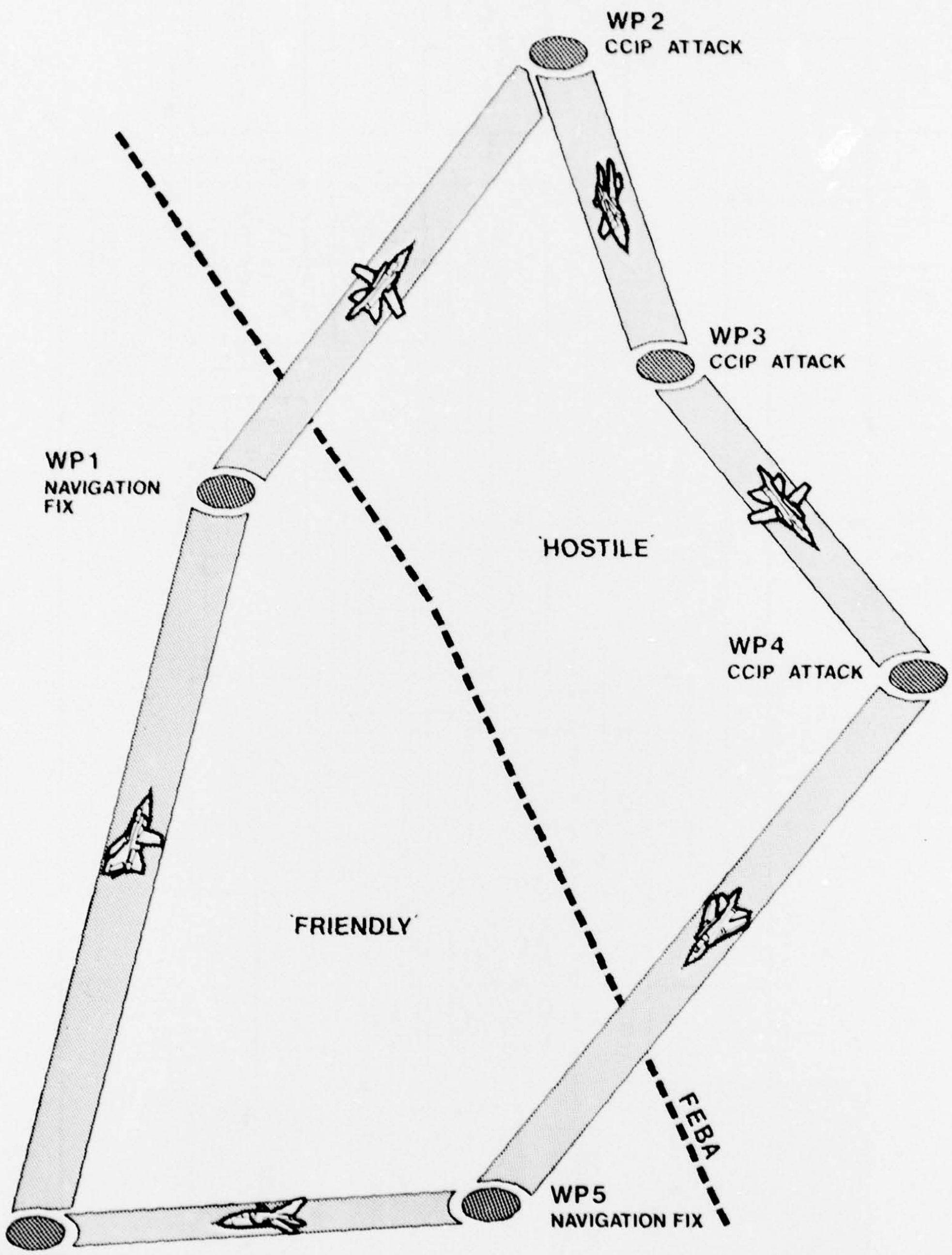


Fig. 2 MISSION PLAN

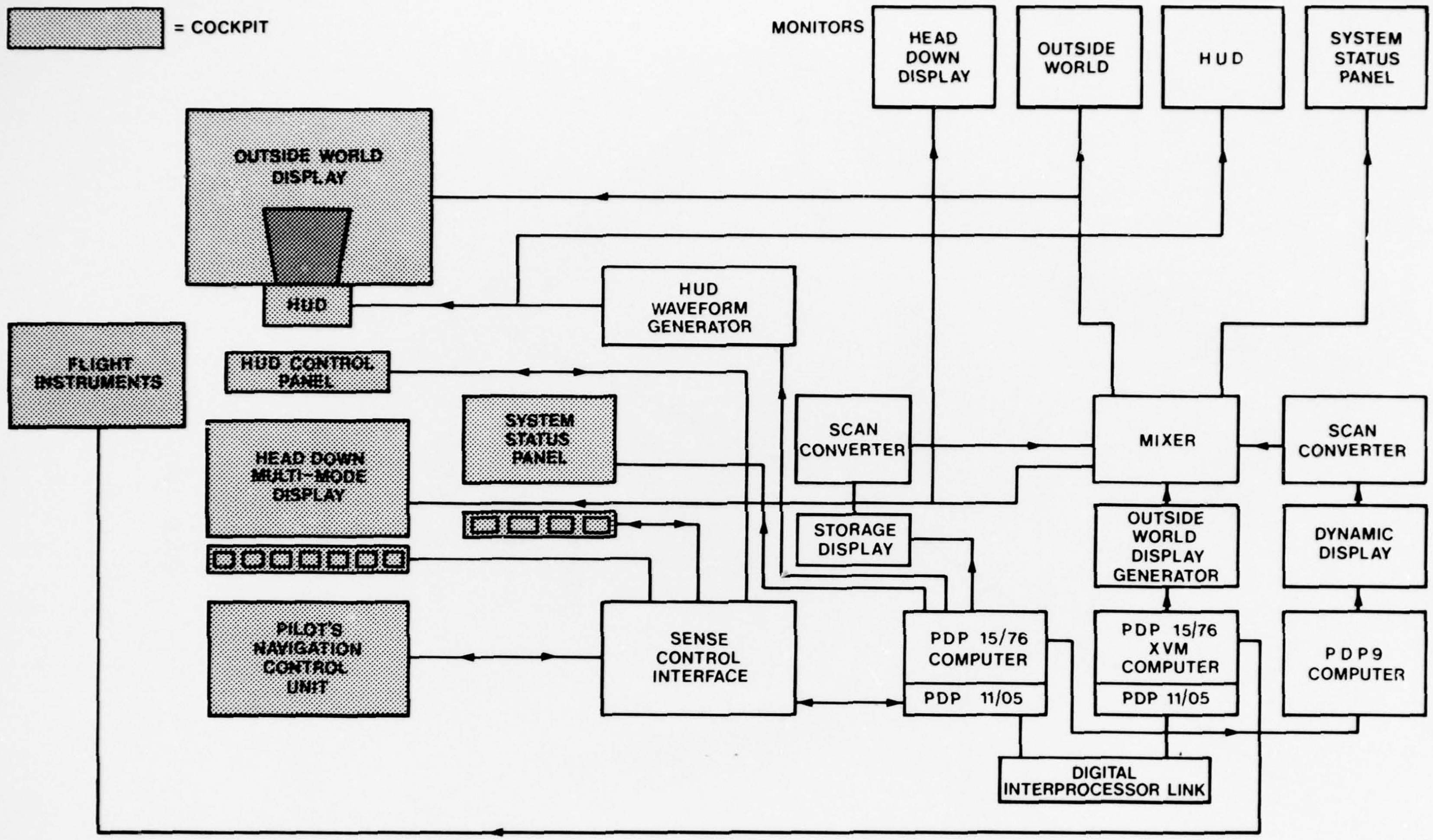


Fig. 3 DISPLAY ASSESSMENT HARDWARE CONFIGURATION

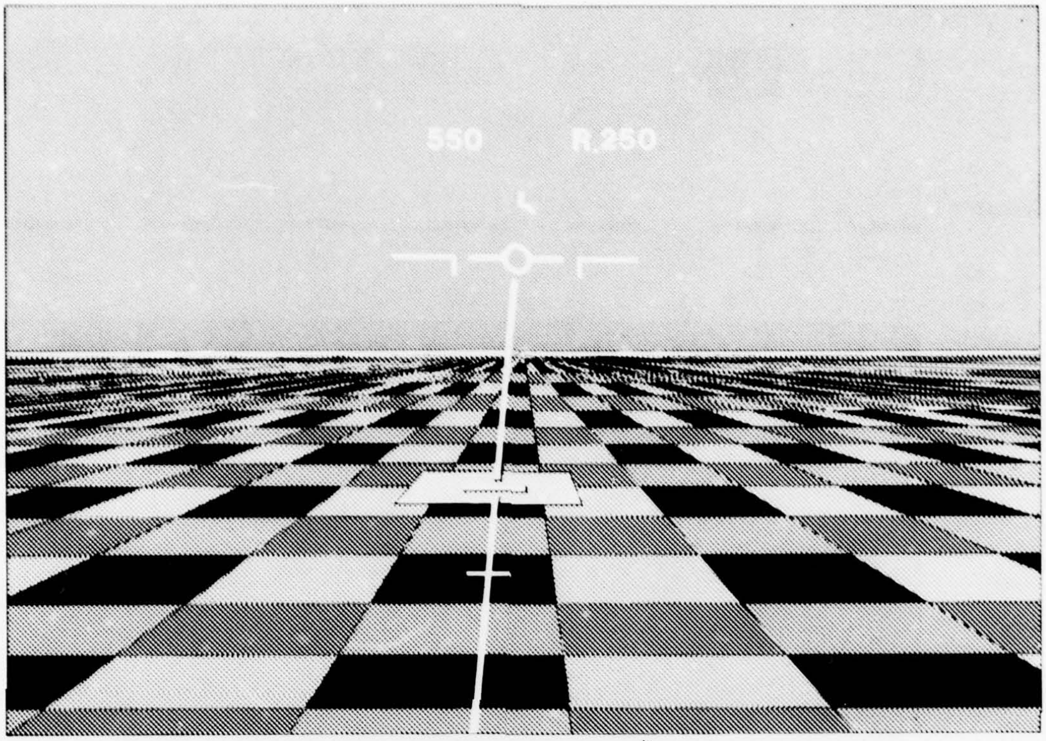


Fig. 4 OUTSIDE WORLD DISPLAY WITH H.U.D.

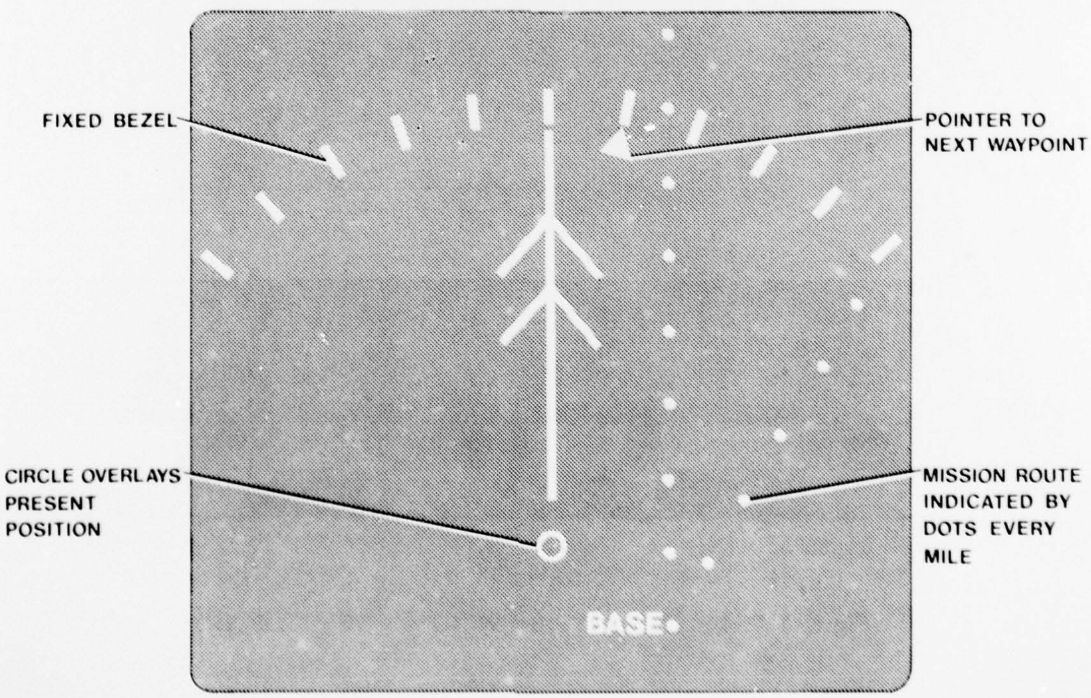


Fig. 5 SIMULATED MAP DISPLAY

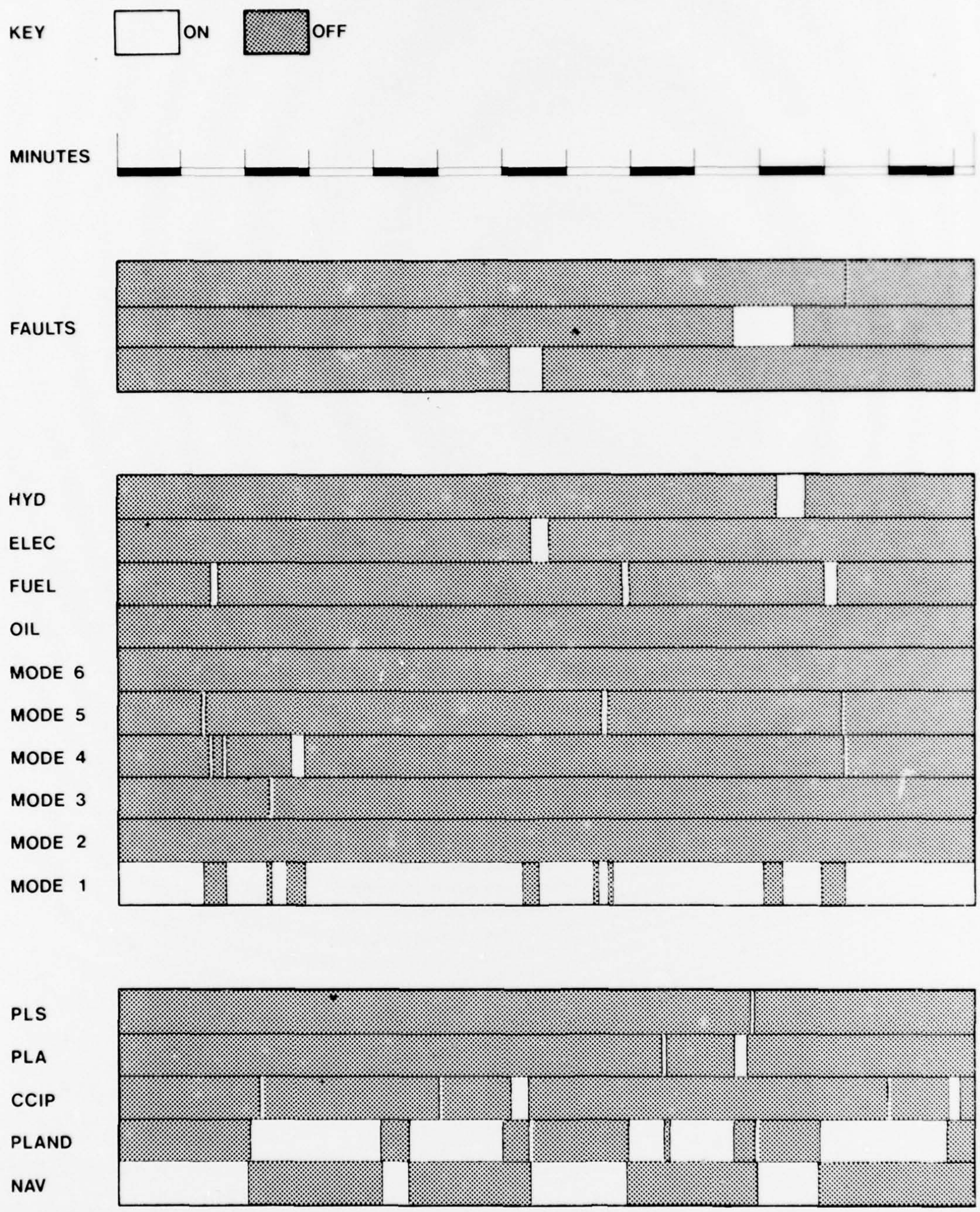


Fig. 6 TIME HISTORY SHOWING MODE SELECTIONS

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- Interaction between displays

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Government Reports Announcements (GRA)
published by the National Technical
Information Services, Springfield
Virginia 22161, USA

Printed by Technical Editing and Reproduction Ltd
Harford House, 7–9 Charlotte St, London W1P 1HD

ISBN 92-835-1303-7